

T.R.
GEBZE TECHNICAL UNIVERSITY
GRADUATE SCHOOL

PRELIMINARY TURBOFAN ENGINE DESIGN, WEIGHT & COST
ESTIMATION

MELİH İÇİNSEL

A THESIS OF MASTER OF SCIENCE
DEPARTMENT OF ENERGY TECHNOLOGIES
APPLIED PROPULSION SYSTEM DESIGN AND ENGINEERING
FOR AEROSPACE TECHNOLOGIES PROGRAM

ADVISOR: ASSIST. PROF. DR. FATİH USTA

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GEBZE TEKNİK ÜNİVERSİTESİ
LİSANSÜSTÜ EĞİTİM ENSTİTÜSÜ

TURBOFAN MOTOR ÖN TASARIMI, AĞIRLIK & MALİYET
YAKLAŞIMI

MELİH İÇİNSEL

YÜKSEK LİSANS TEZİ
ENERJİ TEKNOLOJİLERİ ANABİLİM DALI
HAVACILIK VE UZAY TEKNOLOJİLERİNDE
UYGULAMALI İTKİ SİSTEMİ TASARIM
MÜHENDİSLİĞİ PROGRAMI

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TEMMUZ 2025

MASTER of SCIENCE JURY APPROVAL FORM

A thesis submitted by Melih İÇİNSEL, defended on 25/07/2025 before the jury formed with the 30/06/2025 date and 2025/34 numbered decision of the GTU Graduate Administration Board, has been accepted as a MASTER of SCIENCE thesis in the Department of Energy Technologies, Applied Propulsion System Design and Engineering Program for Aerospace Technologies.

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ABSTRACT

In the conceptual design phase of aero-engines, three parameters are crucial when making the design decisions: performance, weight and cost. To decrease the number of iterations between conceptual and detailed design, it is necessary to estimate cost and weight more accurately while selecting performance cycle at the conceptual design phase.

This study aims to extend the preliminary design process of a performance cycle with a more accurate 1-D flow path and a possible layout. With these detailed inputs in the conceptual design, weight estimation tools can be more accurate and useful which enables a more precise cost estimation. As an overall approach, a performance cycle based on more accurate weight and cost estimation at conceptual design phase expected to bring down the cost and shorten the project time.

First, critical points in the flight envelope are determined considering the role of the airframe. Multi-point design optimization method (MPDOM) is utilized with these points by selecting each point as an individual engine design point for each cycle, and rest of them as an off-design point. Each design point's performance parameters are optimized while meeting the thrust and SFC requirements of the design point and minimizing the SFC of an off-design point. All cycles' outputs are compared and then, engine design point is determined according to the fulfillment of the requirements.

After determining the engine design point, several candidate cycles are obtained by examining the critical performance parameters, turbine inlet temperature (TIT), high pressure compressor pressure ratio (HPC PR), bypass ratio (BPR) and engine inlet corrected mass flow rate. 1-D flow paths of each candidate cycle are generated with cost and weight estimation. One of the candidate cycles is selected as the conceptual design of the engine considering the fulfillment of the requirements while minimizing cost and weight. This methodology is modeled and integrated into Excel with the usage of NASA WATE estimation for weight.

Methodology and the results are compared with existing mixed flow low bypass turbofan engines, and the results are discussed. This method is aimed at shortening the time of iterations between conceptual and detailed design.

Keywords: Turbofan Preliminary Design, Performance, Weight and Cost, Multi-point design optimization method (MPDOM), Flow Path.

ÖZET

Uçak motorlarının konsept tasarım evresindeki tasarım kararlarında üç parametre kritik öneme sahiptir; performans, ağırlık ve maliyet. Konsept dizayn aşamasındaki performans çevrimi seçimi esnasında ağırlık ve maliyet tahminlerini daha doğru yapabilmek, konsept ve detay tasarım aşamaları arasındaki iterasyonları azaltmak için büyük önem arz etmektedir.

Bu çalışma ön tasarım aşamasında performans çevrimini daha doğru 1-B akış yolu ve olası bir yerleşim ile genişletmeyi hedeflemektedir. Konsept aşamasında elde edilebilecek bu ayrıntılı girdilerle ağırlık yaklaşımının daha doğru yapılabilmesi sağlanabilmekte ve bu sayede daha hassas bir maliyet hesabı yapılabilmektedir. Genel yaklaşım ele alındığında konsept tasarım aşamasında daha doğru ağırlık ve maliyet yaklaşımları ile beslenen bir performans çevriminin proje vaktini kısaltması ve maliyeti düşürmesi beklenmektedir.

Öncelikle uçuş zarfındaki kritik noktalar hava aracının rolü göz önüne alınarak belirlenmektedir. Bu kritik noktalardan biri motor tasarım olarak seçilip diğer noktalar tasarım dışı noktalar olarak değerlendirilerek yapılan yaklaşım, her kritik nokta için tekrar yaparak çoklu nokta tasarım optimizasyon metodu (ÇNTOM) uygulanmaktadır. İtke ve Özgül Yakıt Tüketimi (ÖYT) isterlerini sağlarken ve tasarım dışı noktadaki Özgül Yakıt Tüketimi (ÖYT) minimize edilirken her tasarım noktasındaki performans parametreleri optimize edilmektedir. Bütün çevrimlerin çıktıları karşılaştırılmaktadır ve akabinde isterlerin sağlanma durumu göz önünde bulundurularak motor tasarım noktası seçilmektedir.

Motor tasarım noktası seçildikten sonra, birkaç aday çevrim kritik performans parametreleri olan türbin giriş sıcaklığı (TGS), yüksek basınç kompresör basınç oranı (YBK BO), bypass oranı (BPO) ve düzeltilmiş motor giriş debisi incelenerek elde edilmektedir. Her aday çevrim için 1-B akış yolu oluşturulmakta olup maliyet ve ağırlık yaklaşımı yapılmaktadır. İsterleri sağlama durumu göz önünde bulundurularak maliyeti ve ağırlığı minimize edilmiş bir çevrim aday çevrimler arasından seçilmektedir. Bu yöntem, ağırlık yaklaşımı için NASA WATE methodunun kullanımı ile Excel'de modellendirilip entegre edilmiştir.

Yöntem ve sonuçlar, piyasada olan karışık akışlı, düşük bypass oranlı turbofan motorları ile karşılaştırılıp sonuçları tartışılmıştır. Bu yöntem ile konsept ve detay tasarım arasındaki iterasyon süresinin kısaltılması beklenmektedir.

Anahtar Kelimeler: Turbofan Ön Tasarımı, Performans, Ağırlık ve Maliyet, Çoklu nokta tasarım optimizasyon metodu (ÇNTOM), Akış Yolu.

ACKNOWLEDGEMENTS

I would like to express my deepest gratitude to my supervisor, Dr. Fatih USTA, for their unwavering support, guidance, and expertise throughout the process of researching and writing this thesis. His insightful feedback, patience, and encouragement have been invaluable in shaping the direction and quality of this work. I am also equally grateful to my co-supervisors, Emrah GÜLLÜ and Gökhan ARAN, for their valuable contributions, feedback, and support. Their expertise and perspective have greatly enriched this thesis and enhanced its overall depth and rigor. I am truly grateful to my colleagues, Mustafa Burak KEÇEBAŞ and Mehmet Yaşar TEKİN, for their contribution to enhance the thesis and their support.

To my beloved spouse, Öykü SEVER İÇİNSEL, your unwavering support, understanding, and love throughout the course of my master's studies and the writing of this thesis. Your encouragement during long nights, understanding during stressful times, and believe in me even when I doubted myself have been invaluable. This achievement would not have been possible without you by my side, and I am forever thankful for your presence in my life.

I would like to extend my sincere gratitude to my mentor, Gürhan KUZGUN, whose encouragement was instrumental in my decision to pursue this master's degree. His support went beyond professional guidance; his belief in my capabilities gave me the confidence to take this important step. I am truly grateful for his mentorship, which has had a lasting impact on both my academic journey and personal growth.

I would also like to thank my colleagues and fellow master's students, Mehmet Enes YALÇIN and Alper YILMAZ, for sharing this journey with me. Their support, friendship, and encouragement made this journey much more enjoyable and meaningful. I am grateful for the support and friendship we built along the way.

I extend my heartfelt appreciation to all my colleagues, Mahmut Onat AYTAÇ, Metin Deniz YILDIZ, Hande ÖZCAN UZUN and Kerem SELVİ who have supported and encouraged me along this academic journey.

Finally, I would like to express my gratitude to "TEI - Tusaş Engine Industries Inc." for providing the resources and environment necessary for me to pursue my academic aspirations. The opportunities and support offered by the institution have been instrumental in the completion of this thesis.

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LIST OF SYMBOLS AND ABBREVIATIONS

W	: Actual mass flow
A	: Area
a	: Axial clearance
C	: Blade chord
H	: Blade height
U	: Blade velocity
b	: Bypass
c	: Core
ρ	: Density
D	: Diameter
d	: Diffuser length
η	: Efficiency
ΔH	: Enthalpy difference
e	: Exit
ψ	: Flow coefficient
R	: Gas constant
C_p	: Heat capacity at constant pressure
0	: Inlet
L	: Length
M	: Mach Number
$\dot{m}$	: Mass flow rate
ΔR	: Radius difference
ref	: Reference
R	: Rotor
h	: Specific enthalpy
s	: Specific entropy
γ	: Specific heat ratio
$V_{splitter}$	: Splitter height
N	: Spool speed
s	: Static
010	: Station 010
020	: Station 020
021	: Station 021
025	: Station 025
030	: Station 030
041	: Station 041
042	: Station 042
043	: Station 043
045	: Station 045
049	: Station 049
050	: Station 050
060	: Station 060
064	: Station 064
070	: Station 070
090	: Station 090
130	: Station 130
160	: Station 160

S	: Stator
β	: Swirl angle
W	: Swirl velocity
t	: Total
P	: Total Pressure
T	: Total Temperature
V	: Velocity
ϕ	: Work coefficient
AB	: Afterburner
Wet	: Afterburner operation
AR	: Aspect Ratio
BPR	: Bypass Ratio
CEA	: Chemical Equilibrium with Applications
CR	: Clearance Ratio
CC	: Combustion Chamber
CD	: Discharge Coefficient
FPR	: Fan Pressure Ratio
far	: Fuel-to-air ratio
CFG	: Gross Thrust Coefficient
HP	: High Pressure
HPC	: High Pressure Compressor
HPT	: High Pressure Turbine
IGV	: Inlet Guide Vane
ISA	: International Standard Atmosphere
LP	: Low Pressure
LPT	: Low Pressure Turbine
LHV	: Lower heating value
MFR	: Mass Flow Rate
MOPEDS	: Modular Performance and Engine Design System
MoM	: Momentum
MPDOM	: Multi Point Design Optimization Method
FN	: Net Thrust
Dry	: Non-afterburner operation
NPR	: Nozzle Pressure Ratio
1-D	: One Dimensional
OPR	: Overall Pressure Ratio
PTO	: Power take-off
PR	: Pressure Ratio
R&D	: Research & Development
Spool	: Rotating assembly of compressor-shaft- turbine
SL	: Sea Level
SLS	: Sea Level Static
SAS	: Secondary Air System
SFC	: Specific Fuel Consumption
TRLs	: Technology Readiness Levels
TIT	: Turbine Inlet Temperature
TMF	: Turbine Mid Frame
TRF	: Turbine Rear Frame
VIGV	: Variable Inlet Guide Vane
VSV	: Variable Stator Vane

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1. INTRODUCTION

The aviation industry has evolved quite fast in the early 20th century. In the early days of engine powered flights, piston engines with propellers were equipped in the aircraft which is still one of the best choices for low-speed low altitudes flights. As the demand for high-speed flights increased in World War II, revolutionary technology that offers higher speed and altitudes emerged as turbojet engines. This propulsion system changed the principle of the aviation engines from stroke-based systems to continuous flow systems. After this leap in technology, turbofan engines started to show itself in the field with similar thrust-to-weight ratio with turbojets with better fuel efficiency meaning that better mission range for aircrafts.

Turbofan engines can be divided into several categories which are, high or low bypass ratio, mixed or unmixed nozzle, twin-spool or three-spool, geared or direct drive, afterburner or not.

High bypass ratio offers greater fuel efficiency over low bypass ratio for the same inlet mass flow but less thrust. For commercial aircraft where figure of merits based on the efficiency of the flight high bypass ratio engines are preferred. On the other hand, for the military aircraft thrust is crucial for maneuverability, payload and speed. All those figures of merits point to a turbojet engine; fuel efficiency still has almost the same importance as the thrust for the range of the aircraft. Therefore, low-bypass ratio turbofan engines are preferred for military aircraft and UAV.

In the unmixed nozzle configuration, the bypass stream and core stream do not mix in the engine until its exit. Therefore, high velocity core stream and low velocity bypass stream are separated at the exit and high velocity stream which creates a lot of noise being in the center of the exit mass offers a noise suppression with low velocity bypass stream surrounding it. Noise management is important for commercial aircraft that are meant to be around the cities and carry people from one point to another. Unmixed nozzle also offers better nozzle performance for certain regions which can adjust for cruise and take off conditions where commercial aircraft engines work under for most of their life. For the military aircraft, where thrust efficiency is more important, mixed nozzles offer better performance at supersonic flights. Also, for afterburner operation

clean air without combustion gases coming from bypass stream is valuable and mixing them with core flow to heat up a higher portion of a mass flow is crucial for the additional thrust required from the afterburner. Mixed flow turbofan meaning either with a forced mixer structure or a free mixing without a structure allows the mixing of bypass and core streams at the exit of the core and bypass regions. Mixer structure decided with the trade-off between the thrust it provides versus the additional weight it introduces to the engine.

Spool number selection is a fundamental decision that affects the complexity of the engine, weight and maintainability for better performance, efficiency and fuel consumption. With three-spool structure higher OPR can be achieved with better efficiency and operability with the tailored shaft speed for different stages of the compression system rather than dividing the stages into two shafts. Three-spool configurations are more complex and expensive than the two-spool configurations and their advantages stood up in the commercial engines with high thrust requirements whereas most of the commercial and military engines are suitable for the twin-spool configuration.

For the three-spool configurations, geared turbofans are also a consideration for a tailored shaft speed for intermediate compression system with gearbox rather than additional spool. This configuration puts more difficulty on the high-pressure turbine for higher OPR with the lack of intermediate-pressure turbine. The decision lies in the layout of the engine that is studied at different OEMs. Each OEM has its engine layout that comes from their experience.

For the afterburner selection aircraft's requirements is the key whether its mission profile requires fast responses like interception missions or short take-off from military bases or aircraft carriers.

In this thesis study, low bypass mixed flow turbofan engine preliminary design is studied with performance cycle approach, weight and cost estimation to shorten the design iterations between conceptual design and detailed design phase of the aero engines where most of the critical decisions are made that affects the project and need of a change of a concept in the later stages of the developments greatly costs to the engine program. After the short literature review, the requirement of the engine is examined which is crucial for starting the engine preliminary design. Preliminary

design of the turbofan engine is examined in two parts where the thermodynamic cycle is built from scratch at the chapter 4 and one-dimensional sizing is explained in the **Chapter 5**. With 1-D layout weight and cost estimation applied to the candidate cycles for the requirements and possible engine configurations are examined in the chapter 6. Results are discussed in the **Chapter 7** and the conclusion of the study with prospective future works are considered in the **Chapter 8**.



2. LITERATURE REVIEW

2.1. Preliminary Design of Turbofan

In the preliminary design phase of a turbofan engine, the goal is to meet the airframe mission requirements of thrust and SFC with weight and dimensional envelope constraints of the airframe for the engine [1]. For the thrust and SFC part of the requirements Brayton cycle is used with modifications for turbofans [3], [4], [5]. There are four main parameters that are key to selecting the thermodynamic cycle: Bypass ratio (BPR), turbine inlet temperature (TIT), overall pressure ratio (OPR) and fan pressure ratio (FPR) [1], [2], [3], [4]. Thrust and SFC carpet plots or maps are estimated using parametric models or tools like MOPEDS [7]. In these plots, often one of the off-design points is also examined besides the on-design point. Off-design behavior of the engine is estimated with empirical or scaled design maps for the compressor and fan [2], [3], [4]. For turbines design can be estimated choked from idle to max power therefore, focus is more on the stage loading for simple efficiency estimation and cooling requirements for the high TIT [4], [5]. Nozzle configuration and whether there should be a mixer or not is considered according to the LO requirements and flight envelope [1]. At this point of the thermodynamic cycle in the preliminary design, 1D flow models and mean-line analysis are suited [2], [5]. With the generated 1D flow path weight estimation can be performed for the trade of studies between BPR vs SFC, OPR vs TIT [1], [2], [3].

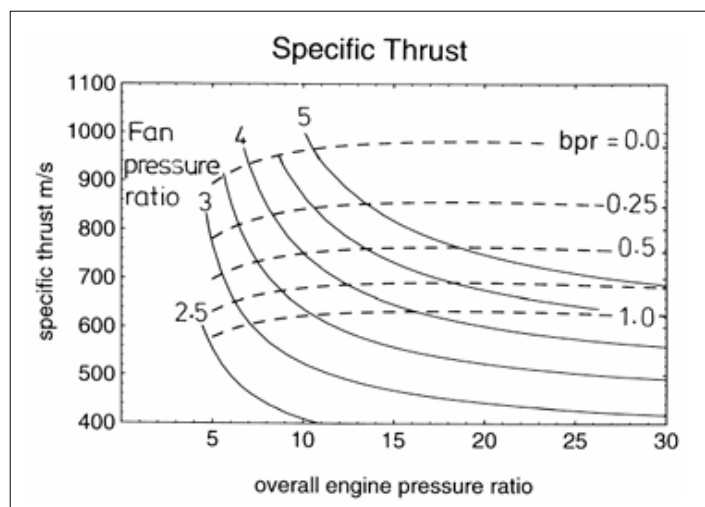


Figure 2.1. Main Parameters Thermodynamic Cycle [3].

2.2. Weight Estimation

In the literature several methods have been developed to estimate engine weight based on key thermodynamic parameters. An empirical approach using regression analysis on a database of modern civil and military engines estimates engine module weights, such as the fan, compressor, turbine, and combustor, based on inputs BPR, OPR, FPR, TIT with an accuracy of approximately $\pm 5\text{--}10\%$ [8]. Another approach with a semi-empirical modular estimation method tailored for advanced turbofan architectures, including geared fans and high-performance cores using same inputs with the addition of spool speeds. This method incorporates technological scaling factors to account for advancements in materials, cooling techniques, and integrated components, resulting in component level weight estimates with an error margin of $\pm 8\%$ compared to existing engines database [9]. A more classical method provides analytical relationships for estimating engine weight and dimensions using Thrust, MFR, and PR as inputs. Though based on legacy engine data, this method remains useful for early conceptual design due to its simplicity and applicability to both large and small gas turbine engines, typically offering accuracy within $\pm 10\text{--}15\%$ [10].

2.3. Cost Estimation

Several comprehensive cost-estimating frameworks have been developed to support early turbofan engine design. One of the earliest structured methodologies for gas turbine engine cost estimation, combining technological classification with analytical cost drivers such as thrust class, development stage, and production quantity. This approach distinguishes between non-recurring costs (e.g., R&D, tooling) and recurring costs (e.g., manufacturing, assembly), making it useful for both military and commercial engine programs [11]. More refined framework developed by integrating cost improvement curves, technology readiness levels (TRLs), and economic factors affecting engine acquisition and sustainment. This method also addresses uncertainty in early-stage cost estimates and supports trade-off analyses across engine configurations and mission profiles [12]. More modernized approach tailored for conceptual design, providing integrated estimation formulas for engine weight, cost, and development time. This methodology uses design-point parameters, such as BPR, FPR, and TIT to estimate both unit cost and total program cost, offering valuable

insight for design decisions involving new or derivative engine concepts [13]. These advanced models are particularly effective when combined with empirical data or modular cost build-ups, allowing for both quick screening and detailed techno-economic analysis during preliminary engine design.



3. DESIGN REQUIREMENTS

Preliminary design of a turbofan engine generally starts with thrust and SFC requirements of an airframer. These thrust and SFC requirements are tied with weight restriction to achieve the required mission profile of the airframe with the designated payload. Also, engine bay dimensions put another constraint for the engine dimensions for both radial and axial length.

Role of the airframe shapes the thrust and SFC requirements of the engine besides the requirement of an afterburner and nozzle configuration. Considering the modern aviation needs a multi-role aircraft chosen for the studies example in this thesis study.

A typical initial thrust and SFC requirement set is given in the following table.

Table 3.1. Thrust and SFC Requirements.

	Altitude [feet]	Flight Mach Number	Thrust [lbf]	SFC [lb/lbf*h]
Part Power	30000	0.85	3500	0.855
Max Dry	20000	0.7	9000	0.860
Max Dry	35000	1.5	11000	1.050
Max Dry	0	0	19000	0.700
Max AB	0	0	32300	1.900
Max AB	0	0.9	38300	2.400
Max AB	0	1.2	39000	2.600
Max AB	30000	0.9	15000	2.150
Max AB	30000	1.3	17000	1.440
Max AB	40000	0.7	7700	2.450
Max AB	50000	0.9	5750	2.550

3.1. Evaluation of Requirements

As a first step, the requirement set of thrust can be evaluated with a non-dimensional cycle approach to determine the required thrust class of the engine that covers all the thrust requirements and evaluating either requirement point requires an over-designed engine respect to other requirement points.

Thrust requirements given in net thrust, therefore thrust formula in equation (3.1) is the starting point of the approach.

$$\dot{m}_e V_e - \dot{m}_0 V_0 + A_e(P_e - P_0) = FN \quad (3.1)$$

For a simpler approach exit mass flow can be set equal to the inlet mass flow meaning that any overboard bleed and fuel flow addition is disregarded.

$$\dot{m}_e(V_e - V_0) + A_e(P_e - P_0) = FN \quad (3.2)$$

Even for a simpler approach, pressure terms of the equation can be omitted.

$$\dot{m}_e(V_e - V_0) = FN \quad (3.3)$$

Actual mass flow in the equation is tied with both flight conditions and engine characteristics. For the design point selection each point assumed to have maximum corrected mass flow. Therefore, actual mass flow can be calculated with flight conditions and taken out of the engine characteristic. Given the inlet velocity is the flight conditions characteristic, only exit velocity term remains as the engine output and can be evaluated as the engine characteristic.

$$V_e = \frac{FN}{\dot{m}_e} - V_0 \quad (3.4)$$

Higher the exit velocity for a requirement points, more difficult for the engine to achieve. For a limited time, afterburner can be used to achieve higher exit velocities but for the engine characteristic this short time should be disregarded. Therefore, requirements for the afterburner operational thrust value are divided to a constant value of 1.5 for a general exit velocity increase of the afterburner for simplicity in the early stages of the evaluation.

A more comprehensive approach is that for each requirement point fundamentals design parameters are assumed such as: BPR, FPR, OPR and T4 limit. Based on the T4/T2 ratio of the requirement points and corrected turbine inlet mass flow, pressure and bypass ratio distribution are calculated which also gives a prior examination of the

on-design and off-design behavior of the engine based on the selection of the design point.

Equation (3.2) is modified with the following equations for more accurate representation.

Exit momentum can be divided into core momentum and bypass momentum as follows:

$$\dot{m}_c V_c + \dot{m}_b V_b = \dot{m}_e V_e \quad (3.5)$$

Where bypass ratio definition:

$$\frac{\dot{m}_b}{\dot{m}_c} = BPR \quad (3.6)$$

By dividing both sides of the equation (3.5) to core mass flow first term of the net thrust equation is modified.

$$V_c + BPR * V_b = \frac{\dot{m}_e}{\dot{m}_c} V_e \quad (3.7)$$

For the pressure term, nozzle pressure ratio is defined in equation (3.8).

$$\frac{P_e}{P_0} = NPR \quad (3.8)$$

$$P_e = NPR * P_0 \quad (3.9)$$

And exit area can be defined in terms of exit mass flow, exit velocity and exit density.

$$A_e = \frac{\dot{m}_e}{\rho_e V_e} \quad (3.10)$$

By modifying equation (3.29) with (3.7), (3.9) and (3.10) exit velocity term left alone. Steps are shown in equations from (3.11) to (3.14).

$$\begin{aligned} \dot{m}_c V_c - \dot{m}_c V_0 + \dot{m}_b V_b - \dot{m}_b V_0 + \frac{\dot{m}_c}{\rho_e V_e} P_0 (NPR - 1) \\ + \frac{\dot{m}_b}{\rho_e V_e} P_0 (NPR - 1) = FN \end{aligned} \quad (3.11)$$

$$V_c - V_0 + BPR * V_b - BPR * V_0 + \frac{P_0 (NPR - 1)}{\rho_e V_e} + \frac{BPR * P_0 (NPR - 1)}{\rho_e V_e} = \frac{FN}{\dot{m}_c} \quad (3.12)$$

$$\frac{\dot{m}_e}{\dot{m}_c} V_e + \frac{P_0 (NPR - 1) * (BPR + 1)}{\rho_e V_e} = \frac{FN}{\dot{m}_c} + V_0 (BPR + 1) \quad (3.13)$$

When final form of the equations is examined, with the set BPR, FPR, OPR and T4 limit exit velocity of the engine can be estimated and there is no need for afterburner correction compared to the first approach.

$$V_e = \frac{\left(\frac{FN}{\dot{m}_c} + V_0 (BPR + 1) \right) + \sqrt{\left(\frac{FN}{\dot{m}_c} + V_0 (BPR + 1) \right)^2 - 4 \frac{\dot{m}_e}{\dot{m}_c} \frac{P_0 (NPR - 1) * (BPR + 1)}{\rho_e}}}{2 \frac{\dot{m}_e}{\dot{m}_c}} \quad (3.14)$$

3.2. Determination of Critical Points

Taking each requirement point as a design point and designing than optimizing an engine with performance and layout and checking other requirements as an off-design point is too much work for early stages of the design where requirements can change drastically with airframers demand. Therefore, after the evaluation of the requirements, critical points for the engine design are decided based on the equation (3.4) or (3.14). Evaluation of the requirements of **Table 3.1** is shown below.

Table 3.2. Exit Velocities Comparison.

	Altitude [feet]	Flight Mach Number	Thrust [lbf]	Equation (3.4) (Normalized)	Equation (3.14) (Normalized)
Part Power	30000	0.85	3500	0.54	0.56
Max Dry	20000	0.5	9950	0.83	0.52
Max Dry	35000	1.5	11000	1.00	0.99
Max Dry	0	0	19000	0.73	0.26
Max AB	0	0	32300	0.73	0.45
Max AB	0	0.9	38300	0.74	0.89
Max AB	0	1.2	39000	0.67	0.99
Max AB	30000	0.9	15000	0.82	0.89
Max AB	30000	1.3	17000	0.73	1.00
Max AB	40000	0.7	7700	0.74	0.75
Max AB	50000	0.9	5750	0.79	0.86

Both approaches define the 35000 feet 1.5 Mach point as the most challenging point. The second approach also shows that 0 feet 1.2 Mach and 30000 feet 1.3 Mach points challenging points as well. Both points are afterburner operation which is simplified in the first approach. By selection of 35000 feet 1.5 Mach both approaches will be examined further with the creation of the engine model.

4. PERFORMANCE CYCLE METHODOLOGY

After determining the critical points for evaluating the possible engine design points, thermodynamic cycle of on-design and off-design solutions are required. At this chapter of the thesis, thermodynamic cycle methodology for the on-design and off-design is explained. Calculations based on the gas model from CEA [2].

4.1. Turbofan Engine Layout for Thermodynamic Calculations

The performance cycle is based on Brayton Cycle in **Figure 4.1** with turning into an open system and modifications for turbofan engines. These modifications are examined in the following subchapters block by block.

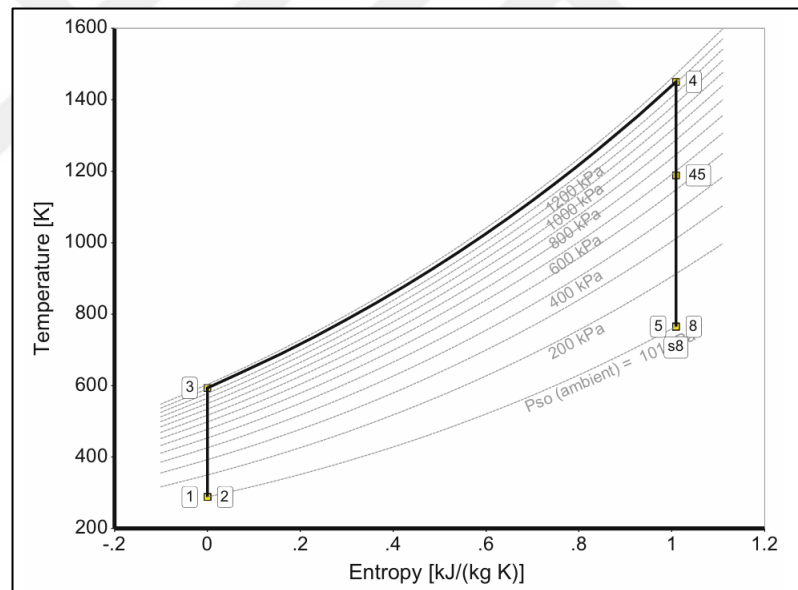


Figure 4.1. Ideal Brayton Cycle [2].

In the thermodynamic cycle each turbomachinery considered as a black box with inlet and exit conditions satisfying the requirements of the required performance. Turbofan engine is built around the turbomachinery parts with main combustor and afterburner, ducts, mixer, exhaust nozzle and supporting structures. Open Brayton cycle is completed with ambient on both sides of the nozzle and the fan where the connection is done with an intake. The overall turbofan block structure can be seen in the **Figure 4.2** with station numbering used in the system.

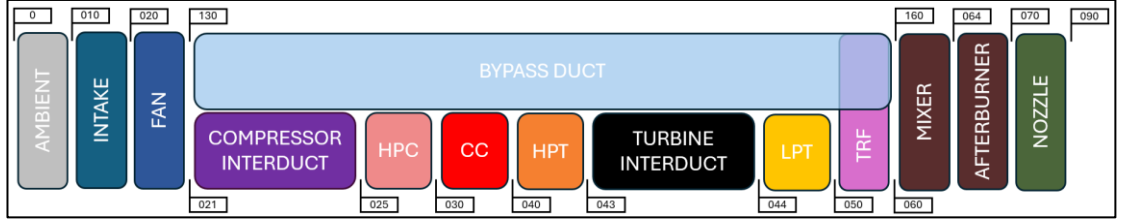


Figure 4.2. Turbofan Engine Block Layout.

4.2. Ambient Block

The International Standard Atmosphere (ISA) model is used in the ambient block with the following equations where temperature in Kelvin, pressure in kilopascal and altitude in meters.

For the temperature at ISA0:

If altitude < 11000 m then,

$$T(\text{altitude}) = 288.15 - 0.0065 * \text{altitude} \quad (4.1)$$

If 24995 m > altitude ≥ 11000 m then,

$$T(\text{altitude}) = 216.65 \quad (4.2)$$

If 30000 m > altitude ≥ 24995 m then,

$$T(\text{altitude}) = 216.65 + 0.0029892 * (\text{altitude} - 24994) \quad (4.3)$$

For the delta from ISA conditions:

$$T_{\text{ambient}} = T(\text{altitude}) + \Delta ISA \quad (4.4)$$

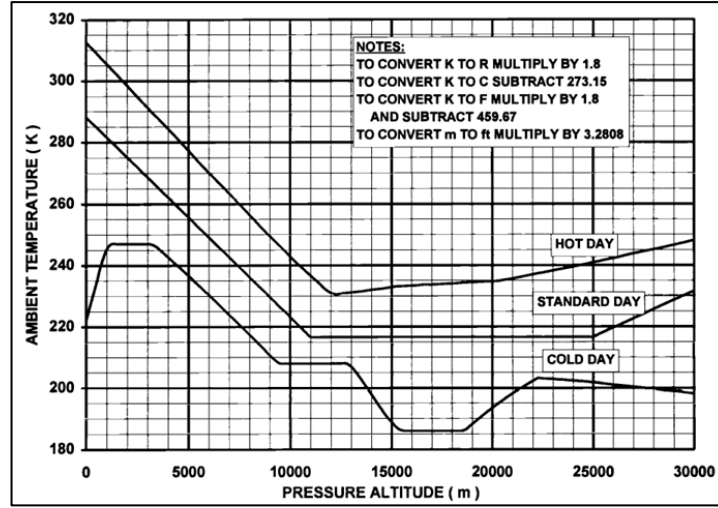


Figure 4.3. Ambient Temperature versus Pressure Altitude [1].

For the altitude pressure:

If altitude < 11000 m then,

$$P(\text{altitude}) = 101.325 * \left(\frac{288.15}{T(\text{altitude})} \right)^{-5.25588} \quad (4.5)$$

If 24995 m > altitude ≥ 11000 m then,

$$P(\text{altitude}) = 22.63253 / e^{(0.000157689 * (\text{altitude} - 10998.1))} \quad (4.6)$$

If 30000 m > altitude ≥ 24995 m then,

$$P(\text{altitude}) = 2.5237 * \left(\frac{216.65}{T(\text{altitude})} \right)^{11.8} \quad (4.7)$$

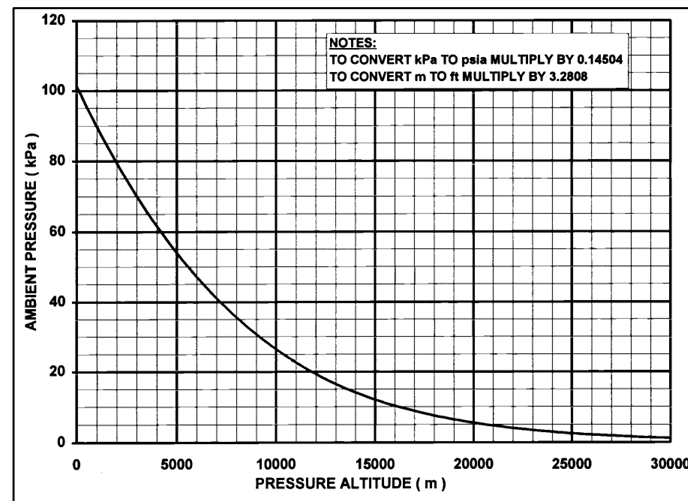


Figure 4.4. Ambient Pressure versus Pressure Altitude [1].

4.3. Intake Block

Intake is the part between the atmosphere and the fan inlet of the turbofan engine. For the integrated engine intake is the responsibility of the airframer where it suits the airframers flight envelope whether it is supersonic or subsonic (**Figure 4.5**) and observability requirements.

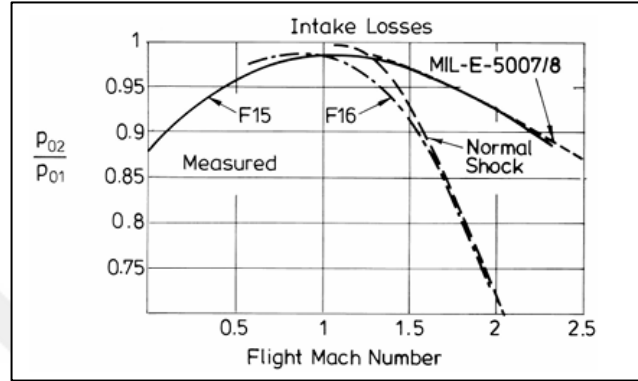


Figure 4.5. F15 and F16 Intake Pressure Losses [3].

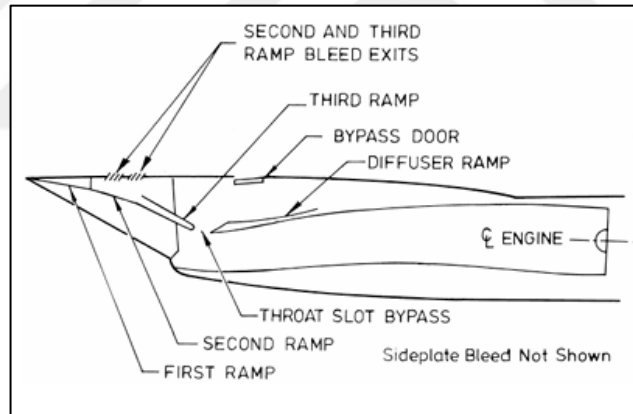


Figure 4.6. F15 Intake Representation [3].

For the uninstalled engine where the initial performance calculations and validations are performed generally a bell-mouth type intake is considered. For the uninstalled configuration MIL-E-5007/8 empirical expressions are used as follows [3].

If Flight Mach Number ≤ 1 then,

$$\frac{P_{020}}{P_{010}} = 1 \quad (4.8)$$

If Flight Mach Number >1 then,

$$P_{020}/P_{010} = 1 - 0.075(M - 1)^{1.35} \quad (4.9)$$

4.4. Fan Block

Fan is the first part of the compressions system that compresses the entire inlet mass flow with the support of low-pressure turbine (LPT). At the exit of the fan air splits into two streams bypass and core flows. In this method, fans are considered in two parts as inner and outer fans while compressing the air. The outer fan provides the air to the inlet of the bypass duct, and the inner fan provides the air to the compressor interduct inlet. Fan blades pressure ratio and efficiency are calculated spanwise of the blade and divided into the bypass ratio as mass averaged. Pressure ratio, efficiency and mass flow rate are inputs for the on-design calculations and for the off-design calculations these 3 inputs are provided in turbomachinery maps. Inlet conditions of the fan come from the intake exit conditions and fan exit conditions are calculated as follows:

Isentropic efficiency for the compression system:

$$\eta = \frac{h_{ideal} - h_{in}}{h_{out} - h_{in}} \quad (4.10)$$

Isentropic efficiency for the inner fan:

$$\eta_{020-021} = \frac{h_{021,ideal} - h_{020}}{h_{021} - h_{020}} \quad (4.11)$$

Isentropic efficiency for the outer fan:

$$\eta_{130-021} = \frac{h_{130,ideal} - h_{020}}{h_{130} - h_{020}} \quad (4.12)$$

While calculating the exit conditions of the fan, first isentropic conditions are considered and ideal outlet temperature calculated as follows.

$$s_{out} - s_{in} = C_p \ln \frac{T_{out}}{T_{in}} - R \ln \frac{P_{out}}{P_{in}} \quad (4.13)$$

Where entropy change is zero and the pressure ratio of the compressions system can be defined as:

$$\frac{C_p}{R} \ln \frac{T_{out}}{T_{in}} = \ln \frac{P_{out}}{P_{in}} \quad (4.14)$$

Gas constant (R) is a function of fuel-to-air ratio which is zero for the compression system and can be taken as constant and Cp is function of fuel-to-air ratio and temperature which is only correlated with temperature. Therefore, pressure increase of the system is the only entropy generation in the isentropic system and can be used to find the ideal outlet temperature with the use of gas tables.

$$s_{out,ideal} = s_{in} + \ln \frac{P_{out}}{P_{in}} \quad (4.15)$$

$$T_{out,ideal} = s_{out,ideal}(T, far) \quad (4.16)$$

After finding the ideal outlet temperature for the fan, difference in enthalpy which is a function of temperature and fuel-to-air ratio can be found for the isentropic process and by dividing into the isentropic efficiency real enthalpy difference is calculated.

$$h_{out} = h_{in} + \frac{h_{out,ideal} - h_{in}}{\eta} \quad (4.17)$$

$$T_{out} = h_{out}(T, far) \quad (4.18)$$

Exit conditions of temperature and pressure are calculated and mass flow rate does not change in the fan until the splitter. Therefore, the power demand of the fan calculated as follows.

$$Power_{FAN} = \dot{m}_{021} * (h_{021} - h_{020}) + \dot{m}_{130} * (h_{130} - h_{020}) \quad (4.19)$$

4.5. Compressor Interduct Block

Compressor interduct is the part between the splitter and the HPC inlet where FBS is also located. Core mass flow passes from this duct where because of the radius drop between the fan and the HPC it is shaped like a goose neck. There is a loss due to Fanno flow inside the duct that is an input in the on-design and in the off-design calculated based on the corrected mass flow rate in the inlet of the duct in this approach.

$$\Delta P_{loss} = \Delta P_{loss,design} * \left(\frac{\dot{m}_{corr}}{\dot{m}_{corr,design}} \right)^2 \quad (4.20)$$

4.6. HPC Block

HPC is the second part of the compressions system that compresses the core mass flow with the support of high-pressure turbine (HPT). At the exit of the HPC air diffuses through a diffuser and flows into the CC. Bleeds (if needed) such as customer and anti-ice extracted from interstage or exit of the HPC while starting bleed extracted from interstage and handling bleed from the exit of the HPC. Besides these bleeds, sump pressurization, LPT cooling and pressurization airs extracted from inlet or interstage of the HPC while HPT cooling and pressurization airs extracted from exit of the HPC. Pressure ratio, efficiency and mass flow rate are inputs for the on-design calculations and for the off-design calculations these 3 inputs are provided in turbomachinery maps. Inlet conditions of the HPC come from the inner fan exit conditions with pressure loss from the compressor interduct and HPC exit conditions are calculated with equations from (4.13) to (4.18). Power of the HPC calculated as follows where all interstage air extraction enthalpy is subtracted respect to their extraction location.

$$Power_{HPC} = \dot{m}_{025} * (h_{030} - h_{025}) - \sum \dot{m}_{SAS,interstage} * (h_{030} - h_{interstage}) \quad (4.21)$$

4.7. CC Block

Combustion chamber (CC) where the energy inputs into the cycle with the burning of fuel. Two types of pressure loss mechanism work in the CC, Rayleigh flow and Fanno flow. In this approach, diffuser loss is part of the CC loss there it is a part of the Fanno flow of the CC. For the off-design pressure loss correlation equation (4.20) is used with the inputs from the CC inlet conditions. For the on-design part, CC outlet temperature given as input. Therefore, required fuel flow calculated with the following equations.

$$\dot{m}_{f,CC} = \frac{(\dot{m}_{030} + \dot{m}_{f,CC})h_{040} - \dot{m}_{030}h_{030}}{\eta_{CC} * LHV} \quad (4.22)$$

4.8. HPT Block

High-Pressure Turbine (HPT) provides the power demand of the HPC and other power demands from the high-pressure shaft such as accessory powers of the engine. For the installed engine, customer power take-off is also provided by the HPT. It is the turbomachinery that faces the highest temperature thus, cooling requirements are substantial. In modern engines almost 20% of HPC inlet mass flow (W025) is used for cooling and pressurization of the HPT. HPT exit conditions are calculated with the following equations based on the power equality of the shaft at steady state with the account of mechanical efficiency of shaft which consists of bearing losses and windage losses in this approach.

$$Power_{HPT} = \frac{Power_{HPC} + PTO}{\eta_{HP}} \quad (4.23)$$

$$h_{042} = h_{041} - \frac{Power_{HPT}}{\dot{m}_{041}} \quad (4.24)$$

Where station enthalpies are calculated with the SAS:

$$h_{041} = \frac{\dot{m}_{040}h_{040} + \sum \dot{m}_{SAS}h_{SAS}}{\dot{m}_{041}} \quad (4.25)$$

With the known actual enthalpy at the exit of the rotor:

$$h_{042,ideal} = h_{041} - \frac{h_{041} - h_{042}}{\eta_{HPT}} \quad (4.26)$$

$$T_{042} = h_{042}(T, far) \quad (4.27)$$

$$T_{042,ideal} = h_{042,ideal}(T, far) \quad (4.28)$$

With the isentropic system approach in the equation (4.15) expansion ratio of the HPT is calculated as follows:

$$\frac{P_{041}}{P_{042}} = e^{s_{041} - s_{042,ideal}} \quad (4.29)$$

And the exit conditions of HPT calculated with the remaining SAS of rotor cooling and rotor exit pressurizing:

$$h_{043} = \frac{\dot{m}_{042}h_{042} + \sum \dot{m}_{SAS}h_{SAS}}{\dot{m}_{043}} \quad (4.30)$$

$$\dot{m}_{043} = \dot{m}_{042} + \sum \dot{m}_{SAS} \quad (4.31)$$

4.9. Turbine Interduct Block

Turbine interduct is the part between the HPT and the LPT inlet where swirl is high at the exit of HPT. Pressure loss value is given as input at on-design and at off-design it is calculated with equation (4.20).

4.10. LPT Block

Low-Pressure Turbine (LPT) provides the power demand of the fan. It is located after the HPT therefore, it has lower temperature and pressure inlet conditions while due to the bypass ratio, lower mass flow rate of the fan that provides power. Thus, the specific power of the LPT differs from the specific power of the fan significantly in the case of high BPR. This leads to a higher stage number of LPT than HPT in some cases. LPT exit conditions calculated with the same principle as HPT that a power equality of the LP shaft at steady state with the account of mechanical efficiency of shaft which consists of bearing losses and windage losses.

$$Power_{LPT} = \frac{Power_{FAN}}{\eta_{LP}} \quad (4.32)$$

$$h_{049} = h_{045} - \frac{Power_{LPT}}{\dot{m}_{045}} \quad (4.33)$$

Where station enthalpies are calculated with the SAS:

$$h_{045} = \frac{\dot{m}_{044}h_{044} + \sum \dot{m}_{SAS}h_{SAS}}{\dot{m}_{045}} \quad (4.34)$$

With the known actual enthalpy at the exit of the rotor:

$$h_{049,ideal} = h_{045} - \frac{h_{045} - h_{049}}{\eta_{LPT}} \quad (4.35)$$

$$T_{049} = h_{049}(T, far) \quad (4.36)$$

$$T_{049,ideal} = h_{049,ideal}(T, far) \quad (4.37)$$

With the isentropic system approach in the equation (4.15) expansion ratio of the LPT is calculated as follows:

$$\frac{P_{045}}{P_{049}} = e^{s_{045} - s_{049,ideal}} \quad (4.38)$$

And the exit conditions of LPT calculated with the remaining SAS of rotor cooling and rotor exit pressurizing:

$$h_{050} = \frac{\dot{m}_{049}h_{049} + \sum \dot{m}_{SAS}h_{SAS}}{\dot{m}_{050}} \quad (4.39)$$

$$\dot{m}_{050} = \dot{m}_{049} + \sum \dot{m}_{SAS} \quad (4.40)$$

4.11. TRF Block

Turbine rear frame (TRF) is an important support structure of the engine where thrust mount and oil pipes are located. For the thermodynamic cycle its important is the Fanno losses of core and bypass parts. The core part of the pressure loss is defined as turbine exit pressure loss in the thermodynamic cycle the bypass part of the TRF loss is defined inside the bypass pressure loss. It is defined same as the other duct Fanno losses with equation (4.20).

4.12. Bypass Duct Block

Bypass duct located around the core engine between the fan exit and mixer inlet. All the pipes from engine carcass into the core pass through the bypass duct, therefore, there are losses attached to these structures besides the Fanno flow across the bypass duct. As mentioned above, bypass part of the TRF structure's loss is defined inside the overall bypass loss which is taken as input in the on-design and correlated at the off-design with equation (4.20).

4.13. Mixer Block

Mixer block for the thermodynamic cycle is where bypass stream and core stream meet and mixes. It can be either a free mixing where there is no structure or can be forced mixing with a physical mixer configuration. For the physical mixer configuration mixing losses are minimized therefore, thrust loss due to mixing of two streams is minimized. For the forced mixer configuration, a cold stream (bypass stream) pressure

loss and a hot stream (core stream) pressure loss is defined in on-design and scaled to off-design with equation (4.20). Mixing efficiency is taken as 1 in this approach and its effect on thrust is defined in CFG value. For the mixing of two streams their static pressure should be equal. In this approach mixer sizing done with agreeing a certain Mach number at the exit of the bypass channel to keep bypass Mach number at reasonable levels at the off-design of the engine due to high pressure losses at high Mach numbers with increasing the BPR at the off-design and risk the operability of the fan.

With the known value of total pressure and temperature and Mach number input of bypass exit static pressure and temperature of the bypass exit calculated as follows with the constant gamma assumption for the total and static conditions due to low Mach number values.

$$P_s = P_t \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{-\frac{\gamma}{\gamma - 1}} \quad (4.41)$$

$$T_s = T_t \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{-1} \quad (4.42)$$

Mass flow function is rearranged for calculating the area of hot stream area as follows with initial Mach number guess:

$$\dot{m} = \frac{A * P_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{-\frac{\gamma + 1}{2(\gamma - 1)}} \quad (4.43)$$

$$A = \dot{m} \frac{\sqrt{T_t}}{P_t} \sqrt{\frac{R}{\gamma}} \frac{1}{M} \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \quad (4.44)$$

After finding the area, the initial Mach number guess is checked with the following procedure with the constraint of static pressure equal to bypass exit static pressure loss:

$$\rho = \frac{P_s}{T_s * R} \quad (4.45)$$

$$V = \frac{\dot{m}}{\rho A} \quad (4.46)$$

$$M = \frac{V}{\gamma R T_s} \quad (4.47)$$

Equations from (4.44) to (4.47) solved numerically until the error of the Mach number difference from the previous solution 0.1%.

Summation of the calculated areas of hot stream and cold stream should be equal to the exit of the mixer and mass flows:

$$A_{064} = A_{060} + A_{160} \quad (4.48)$$

$$\dot{m}_{064} = \dot{m}_{060} + \dot{m}_{160} \quad (4.49)$$

For the exit conditions of mixer equations from (4.44) to (4.47) solved numerically until the error of the Mach number difference from the previous solution 0.1%. Exit Mach number of the mixer is important for afterburner operation. Therefore, it is kept below 0.3 Mach for the design calculations.

4.14. Afterburner Block

Afterburner located between the mixer and the nozzle as an additional combustor for a short-term dramatic increase of the nozzle exit temperature for increase the gas velocity, therefore the thrust. Different from the main combustor of the engine, the inlet of the afterburner consists of burned fuel means far is not zero unlike main combustor. Efficiency of the afterburner is a function of burned fuel and air, meaning it can be tied with the far of the inlet of the afterburner. Therefore, thrust correction for non-dimensional approach cannot be truly used for the afterburning operation thrust for evaluating the airframers requirements as mentioned in **Chapter 3.1**.

For the on-design exit temperature of the afterburner taken as input and for the off-design calculations either thrust requirement or stoichiometric limit taken for the exit temperature of the afterburner. Fanna flow pressure loss of the afterburner duct and spray bars are account in the mixed stream pressure loss in the thermodynamic cycle. Rayleigh loss of the heat addition at the operational afterburner is calculated with the following process.

$$\dot{m}_{070} = \dot{m}_{064} + \dot{m}_{fr} \quad (4.50)$$

Impulse function constant for the Rayleigh flow:

$$MoM = \rho V^2 + P_s \quad (4.51)$$

$$\rho_{064} V_{064}^2 + P_{s,064} = \rho_{070} V_{070}^2 + P_{s,070} \quad (4.52)$$

Pressure loss of the stream is calculated with the equation (4.15) where pressure ratio is taken as inlet pressure over exit pressure. Equations from (4.41) to ((4.47) solved numerically until the error of the pressure loss difference from the previous solution 0.1% with the constraint of inlet and exit impulse is the same (Equation (4.52)).

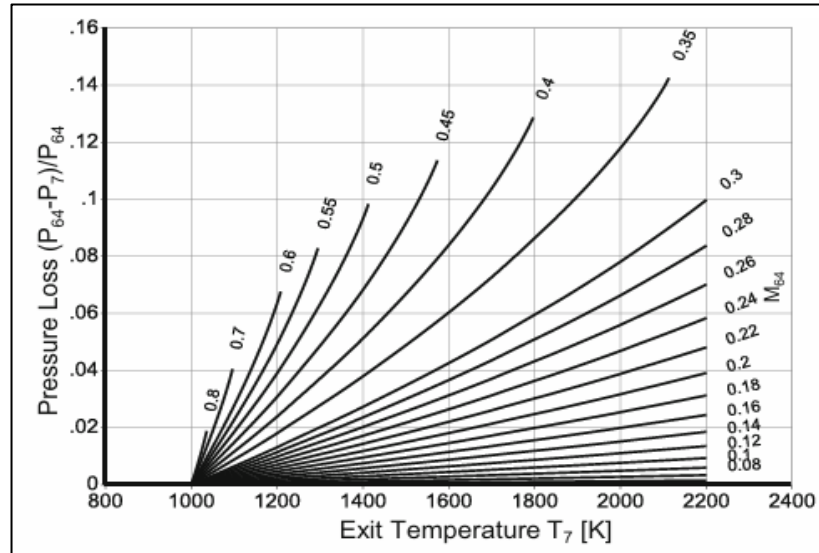


Figure 4.7. Rayleigh Flow Chart [2].

4.15. Nozzle Block

Nozzle is the exit of the engine to the ambient that can be either a convergent nozzle or convergent-divergent nozzle. Design practice for the selection of the convergent or con-di nozzle is a function of nozzle pressure ratio (NPR) that is stated as P/P_s in the **Figure 4.8**. Additional weight and complexity of the divergent part of the nozzle cannot be justified under the NPR value of 12. NPR is defined as:

$$NPR = \frac{P_{t,090}}{P_{ambient}} \quad (4.53)$$

When considering that fan pressure ratio minus the losses through the bypass duct and aft body it gives the nozzle exit pressure. Considering modern engines, up to 4.5 pressure ratio can be assumed at the exit of the nozzle which is not nearly enough to selecting the con-di nozzle. At this point, the ram effect of the Mach on the intake should be considered.

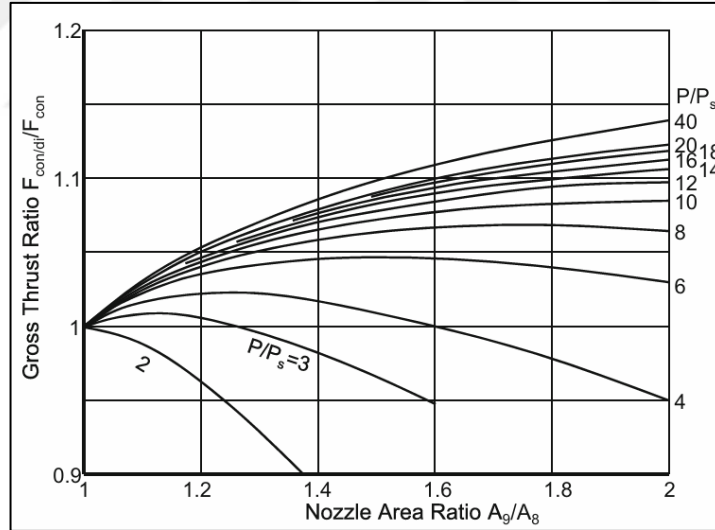


Figure 4.8. Thrust Ratio of the Con-Di to the Convergent Nozzle [2].

NPR can define as follows while assuming gamma equals to 1.4:

$$P_t = P_{amb} \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{\frac{\gamma}{\gamma - 1}} \quad (4.54)$$

$$\frac{P_{t,ram}}{P_{amb}} = \left(1 + \frac{1.4 - 1}{2} M^2\right)^{\frac{1.4}{(1.4-1)}} \quad (4.55)$$

Nearly 3 pressure ratio of the ram effect with the fan pressure ratio can justify the selection of the con-di nozzle. From equation (4.55) can be seen that flight Mach number from 1.4 Mach or higher is the key for selection of con-di nozzle.

Another aspect of the selection of nozzle configuration is the variable or constant convergent part of the nozzle. For the afterburning operation, sudden temperature increases, and the pressure drop changes the back pressure of the fan and shifts its operating line to the stall/surge line which is a major problem for the engine operability. For this reason, variable convergent part is key for the turbofan engines with afterburner.

For the nozzle calculation due to the high Mach numbers constant gamma assumption between the total and static conditions cannot be applied. On the other hand, for the con-di nozzle at the throat Mach taken as 1 from idle to max and nozzle exit static pressure assumed to be equal to the ambient (full expansion). There is no pressure loss and heat addition from inlet to exit of the nozzle therefore, flow is taken as isentropic and with known static pressure at the exit and entropy all other exit conditions calculated with the equations (4.45), (4.46) and (4.47) in order.

5. ENGINE SIZING

5.1. Fan Sizing

One of the critical performance parameters of the conceptual design is the corrected inlet mass flow rate. There are several constraints for the maximum mass flow rate such as: intake capability, maximum diameter of the engine bay, relative tip Mach number, mass flux of the inlet.

In this thesis, the engine bay limitation taken as constraint for the tip radius and hub to tip ratio taken as constraint for the first bearing placement under the fan. With the known ambient conditions and area of the inlet, Mach number at the fan inlet is calculated with equations (4.45) to (4.47). In this equation velocity term is the axial velocity that governs the mass flow rate. In the **Figure 5.1**, β_1 refers to the swirl angle of the flow which is the turning angle of the IGV or VIGV if there is one. For the same target corrected mass flow swirl increases the axial Mach number but decreases the relative tip Mach number as can be seen in equation (5.1).

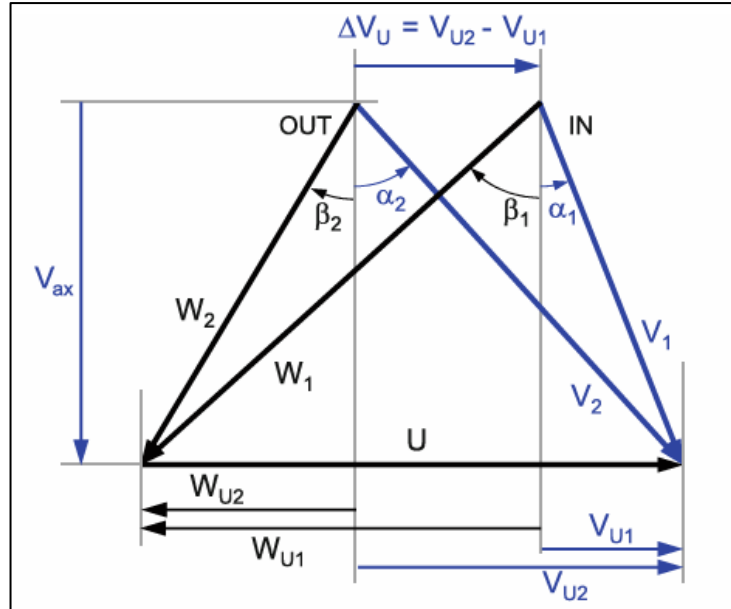


Figure 5.1. Compression System Velocity Diagram [2].

$$\text{Relative Tip Mach Number} = \frac{\sqrt{V_{020,axial}^2 + (U_{020} - W_{U020})^2}}{\sqrt{\gamma_{020} R_{020} T_{s,020}}} \quad (5.1)$$

For the turbomachinery sizing there are 3 basic configurations, constant hub, constant tip and constant mean-line approach. Engines in the literature are examples of these 3 configurations and combinations of them. For this thesis, mean-line turbomachinery calculations are used, therefore, constant mean-line configurations also enable easier calculations. But for the implementing the methodology in this thesis to existing engines, constant hub, constant tip and rising mean-line approaches are also implemented.

With the known mean radius, exit area can be set with the exit Mach number assumptions which can be taken 0.40-0.45 for highly loaded fan like in low bypass turbofan engines.

For the axial length of stators and rotors 2-D aspect ratios are used (projection of the chord length on z-axis). Aspect ratio definition given in equation (5.2).

$$AR = \frac{C_{axial}}{H} \quad (5.2)$$

For the axial length between the stators and rotors, the axial clearance ratio is used with the following equation where a_i can be taken bigger than a_{i+1} for rotor loadings.

$$CR_{axial} = \frac{H_{S_i} + H_{R_i} + H_{S_{i+1}}}{a_i + a_{i+1}} \quad (5.3)$$

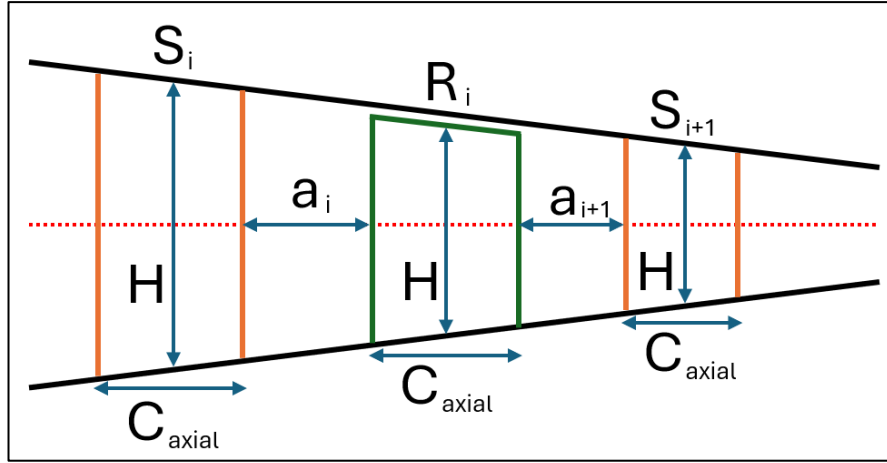


Figure 5.2. Compressor Radial and Axial Dimension Representation.

Stage loading coefficient, also known as work coefficient, is the change angular momentum normalized by the mean blade velocity. Work and flow coefficients are given with the following equations respectively:

$$\psi = \frac{\Delta H}{U_{mean}^2} \quad (5.4)$$

$$\phi = \frac{V_{axial}}{U_{mean}^2} \quad (5.5)$$

With the assumption of overall ΔH is divided equally to each stage. Stage numbers can be found with aligning the stage loading flow coefficient as close as possible along the diagonal line from starting from 0. Rotational speed of the shaft decided with the turbine calculations of the counterpart of the same shaft.

Besides the thermodynamic cycle input of ΔH , with mean-line analysis mean radius of the blades and rpm values is found and the Smith Chart (**Figure 5.3**) is utilized to ensure the isentropic efficiencies used in the thermodynamic cycle are feasible. It should be noted that Smith Charts normalized the blades around the tip clearance values and does not include secondary airs, therefore, efficiency values should be corrected accordingly while implementing into the cycle.

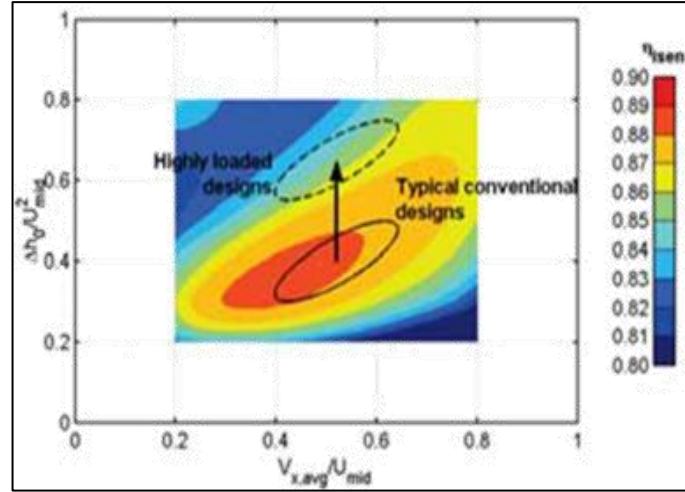


Figure 5.3. Compressor Smith Chart.

Reference values for given constraints for a low bypass turbofan given in the **Table 5.1**.

Table 5.1. Fan Design Constraints Values.

Inlet Mach Number	0.55-0.7
Exit Mach Number	0.40-0.45
Tip Relative Mach Number	<1.4
Rim Speed	300 – 450 m/s
Inlet Hub to Tip Ratio	0.30-0.45
Stator AR	2.5-3.5
Rotor AR	2.0-2.5
Axial Clearance Ratio	0.25-0.30

5.2. HPC Sizing

HPC inlet conditions are taken from inner fan exit conditions with pressure drop at the compressor interduct. With the given bypass ratio, the inlet mass flow of the HPC is calculated and with equation (4.43) annular area of the inlet and outlet can be calculated with axial Mach number assumptions. For the inlet of the HPC 0.45-0.50 Mach number can be taken for the conceptual design and for the exit 0.25-0.30 Mach number is suitable where the diffuser is considered in the CC calculations. Inlet hub to tip ratio taken as 0.7 for the inlet area. With the constant mean-line approach, exit radius and last rotor blade height are calculated which cannot be smaller than 18-20 mm for manufacturability.

In modern engines variable structures are key for the operability in the flight envelope (**Figure 5.4**). All state-of-the-art HPC uses variable inlet guide vane (VIGV) in the inlet and variable stator vane (VSV) in the front stages of the compressor with the ratio of 1/3 VSV over overall stator count. For the design speed one angle is set and in the off design of the turbomachinery angle of the stators are changed with an actuator depending on the corrected speed of the turbomachinery which can be calculated with the following equation.

$$N_c = \frac{N}{\sqrt{T_{inlet}/T_{ref}}} \quad (5.6)$$

For the design point calculations this value is set to an angle and calculation steps are same with fan sizing that explained from **Figure 5.1** to **Figure 5.3**.

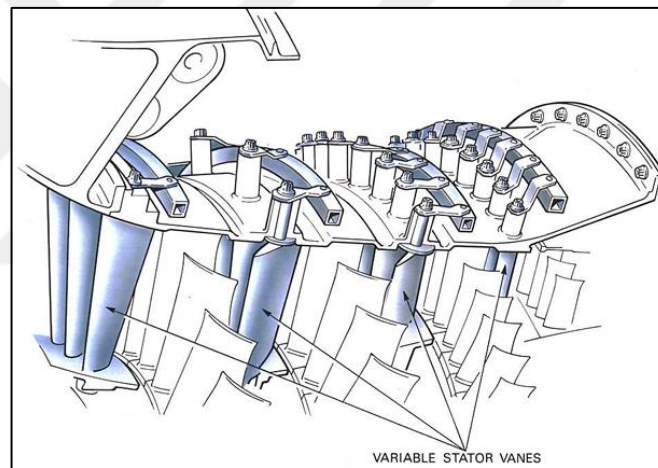


Figure 5.4. VSV Representation [14].

Reference values for given constraints for a low bypass turbofan given in the **Table 5.2**.

Table 5.2. HPC Design Constraints Values.

Inlet Mach Number	0.45-0.50
Exit Mach Number	0.25-0.30
Tip Relative Mach Number	<1.4
Rim Speed	350-500 m/s
Inlet Hub to Tip Ratio	0.65-0.75
Stator AR	2.00-2.50
Rotor AR	2.00-2.50
Axial Clearance Ratio	0.25-0.30
Last Rotor Height	≥ 18-20 mm

5.3. Compressor Interduct Sizing

Compressor interduct sets the axial length between the fan and the HPC with splitter for separating the bypass flow and the core flow. There are several rules of thumb that can be followed in the conceptual design phase that is shown in **Figure 5.5** and given in the following equations.

$$\frac{H_{splitter}}{L} = 0.3 \text{ to } 0.5 \quad (5.7)$$

$$\frac{H_{fan \text{ exit}}}{V_{splitter}} = 0.45 \text{ to } 0.50 \quad (5.8)$$

$$\frac{\Delta R_1}{L} = 0.30 \text{ to } 0.40 \quad (5.9)$$

Where:

$$\Delta R_1 = R_{mean, inner \text{ fan}} - R_{mean, HPC} \quad (5.10)$$

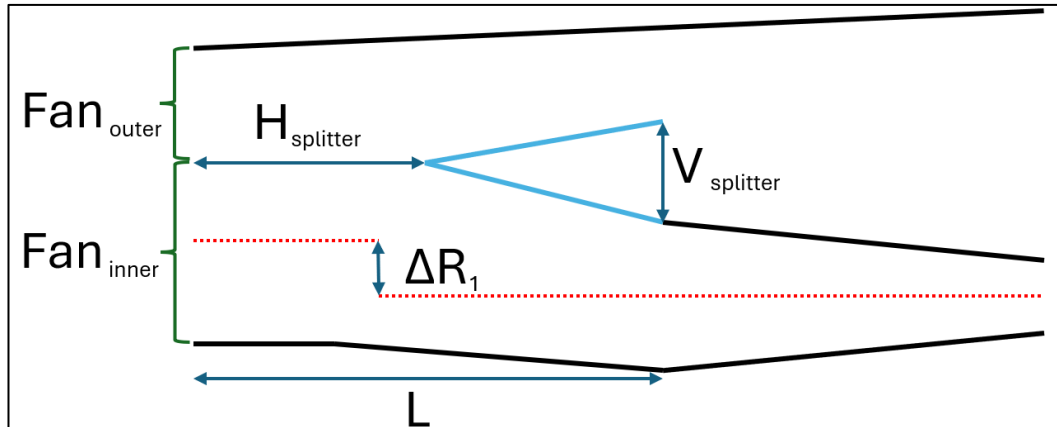


Figure 5.5. Splitter Sizing Dimensional Representation.

5.4. CC Sizing

Combustion chamber requires a low Mach number at swirler region for stable flame and sufficient speed along its length for stable flame propagation. Swirler region generally designed at 0.1 Mach thus, a diffuser is used between the compressor exit and combustion chamber inlet. In this approach, as mentioned before, diffuser design is considered part of the CC and pressure loss of the diffuser accounted in the CC pressure loss. For the basic sizing dome area calculated with the inlet conditions for the desired Mach number at the swirler. Exit of the CC is calculated based on the T4 inputs of the design and continuum of the mass flow and pressure at the maximum Mach number of 0.15 at the inlet of the HPT. Inlet area taken as same with swirler area and dome diameter is calculated. The following two rules of thumb are used in the initial sizing of the CC and shown in **Figure 5.6**.

$$\frac{L}{D_{dome}} = 2 \quad (5.11)$$

$$\frac{d}{D_{dome}} = 1 \quad (5.12)$$

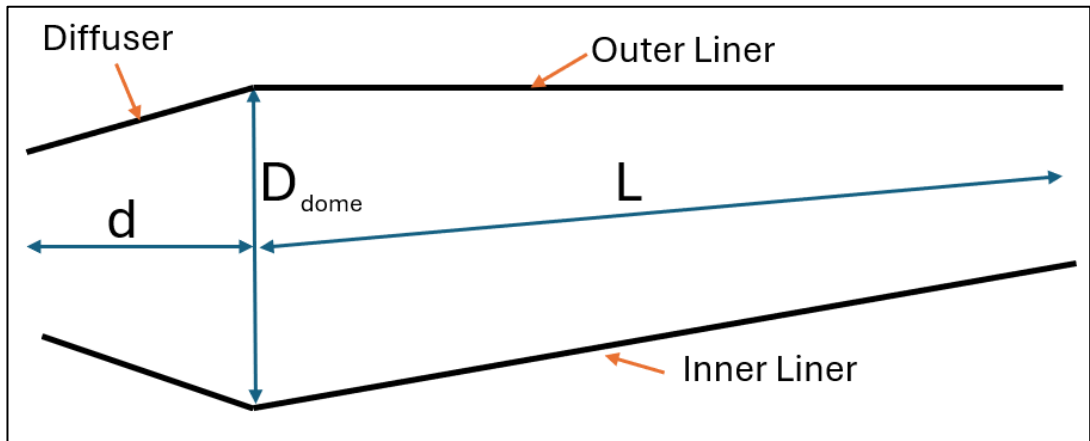


Figure 5.6. CC Sizing Dimensional Representation.

Volume of the CC calculated as two truncated cone from diffuser to dome and dome to exit of the CC.

Residence time is important for flow propagation, and it is the time of the flow along the length from dome to exit of the CC where 2-5 ms is preferred.

Loading is calculated with the following formula, and it is designed to be less than 10 kg/s*atm^{1.8}*m³ for the SLS condition.

$$Loading = \frac{W_{030}}{Volume * \left(\frac{P_{030}}{101.325}\right)^{1.8} * 10^{(0.00145*(T_{030}-400))}} \quad (5.13)$$

Intensity of the combustion that rate of heat release per volume and it should be less than 60 MW/m³ at SLS. Both loading and intensity are measures of combustion difficulty.

$$Intensity = \frac{W_f * \eta_{cc} * LHV}{P_{030} * Volume} \quad (5.14)$$

5.5. HPT Sizing

HPT inlet temperature and pressure are the highest for the turbomachinery's of the engine. Therefore, its flow path is nearly constant and with inlet hub radius input and inlet Mach number input its size can be calculated. Inlet hub radius can be estimated considering not exceeding through the bypass duct and keeping the AN² below its limits. AN² is the mechanical limit (blade stress) of the turbine blades representing the current technological level of the engine. With the following formula it can be represented in MPa.

$$AN^2 = A * N^2 * 10^{-9} * 1550 \quad (5.15)$$

Besides the area, AN² includes the rotational speed of the shaft which can be decided with two turbomachinery sharing the same shaft.

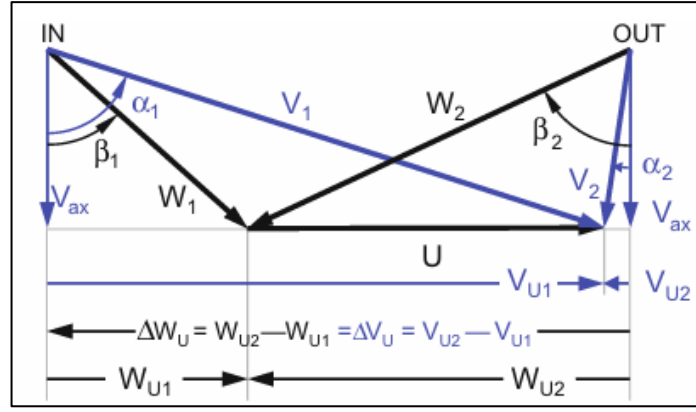


Figure 5.7. Expansion System Velocity Diagram [2].

Equations (5.2), (5.4) and (5.5) are same for the turbine calculations. Axial clearance ratio is simplified for the turbines with the following equations.

$$CR_{axial} = \frac{a}{H_{NGV}} \quad (5.16)$$

For the turbines Smith Chart in **Figure 5.8** is used and similar with compressor Smith Chart normalized the blades around the tip clearance values and does not include secondary airs, therefore, efficiency values should be corrected accordingly while implementing into the cycle.

With the assumption of overall ΔH is divided equally to each stage. Stage numbers can be found with aligning the stage loading flow coefficient as close as possible along the diagonal line from starting from 0.

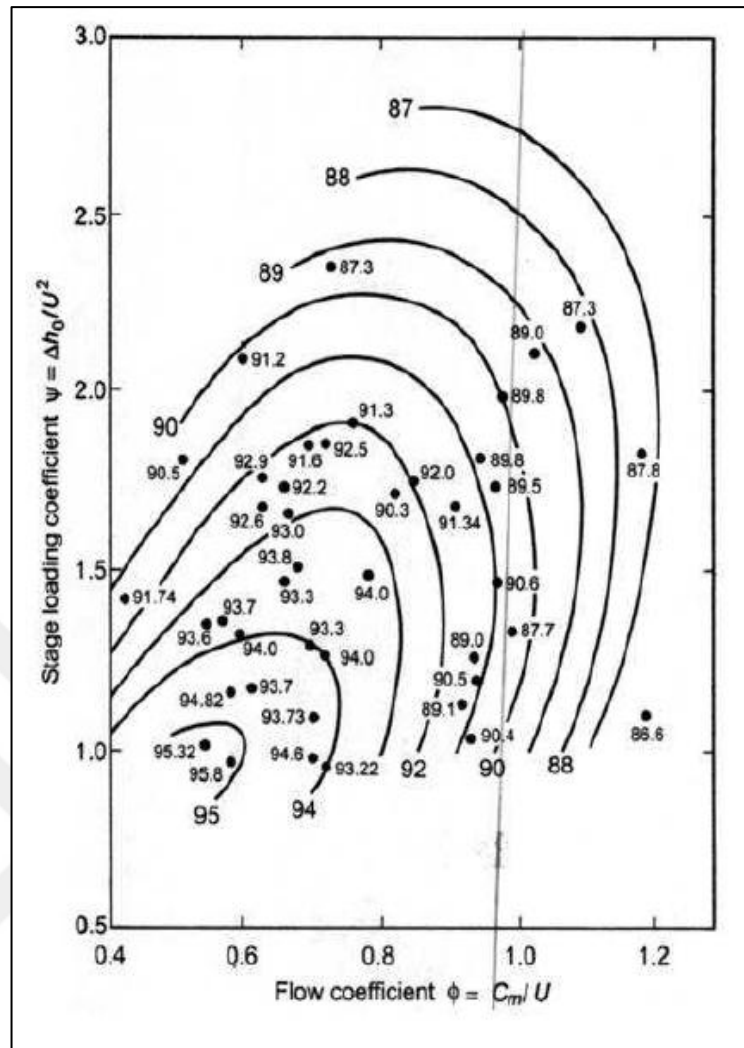


Figure 5.8. Turbine Smith Chart.

Reference values for given constraints for a low bypass turbofan given in the **Table 5.3**.

Table 5.3. HPT Design Constraints Values.

Inlet Mach Number	0.15-0.20
Exit Mach Number	0.45-0.50
Rim Speed	350-500 m/s
Inlet Hub to Tip Ratio	0.50-0.85
Stator AR	1.00-1.20
Rotor AR	1.00-1.20
Axial Clearance Ratio	0.50
AN^2	< 50 MPa
Hade Angle per Stage	< 15 °

5.6. Turbine Interduct Sizing

There are 2 main concepts in the turbofan design that support the rear structure; located between turbines called TMF or behind the last turbine stage called TRF. In this thesis, TRF type of configuration is examined, therefore, turbine interduct is a simple short duct. Thus, its sizing is the same as HPT NGV axial chord length as a rule of thumb.

5.7. LPT Sizing

LPT sizing steps are the same as the HPT with the following constraints.

Table 5.4. LPT Design Constraints Values.

Inlet Mach Number	0.45-0.50
Exit Mach Number	0.45-0.50
Rim Speed	200-350 m/s
Inlet Hub to Tip Ratio	0.50-0.85
Stator AR	1.50-2.00
Rotor AR	1.50-2.00
Axial Clearance Ratio	0.50
AN^2	< 50 MPa
Hade Angle per Stage	< 15 °

5.8. Mixer Sizing

With the increasing length to diameter ratio of a mixer, its thrust gain is increasing but due to the length restrictions of the engine dimensional envelope L/D is taken as 1.25 for the mixer as a common practice.

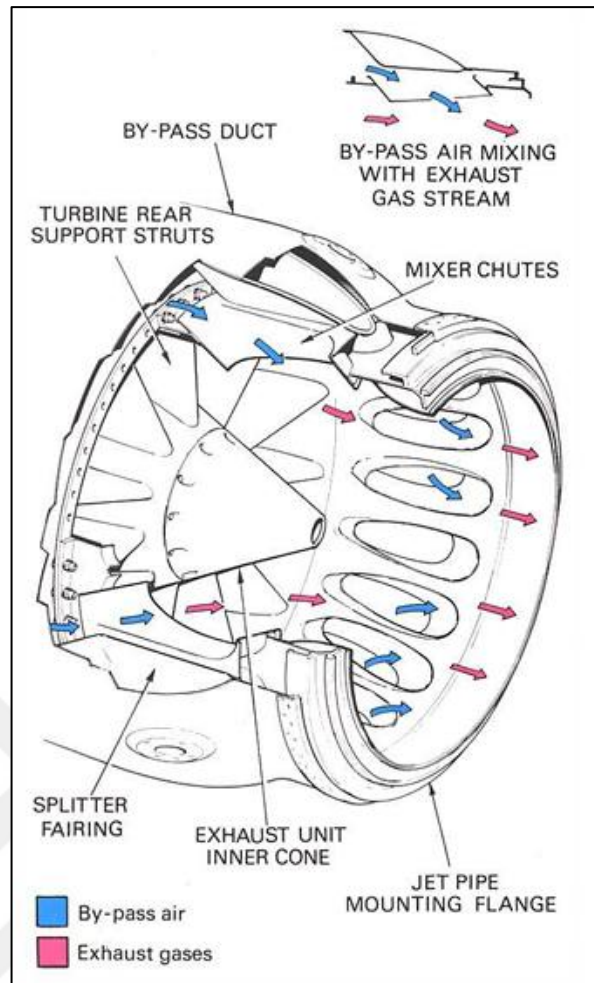


Figure 5.9. Mixer Representation [14].

5.9. Afterburner Sizing

Inlet Mach number should be between 0.20 – 0.30 Mach for combustion stability and light off capability. With inlet Mach number requirements fulfilled, length can be taken as 2.5 times of the diameter at conceptual design phase due to the engine dimensional envelope. Loading value calculated the same as the main combustor and it should be less than 100 for efficiency around 90%.

5.10. Nozzle Sizing

Convergent-divergent nozzle sizing depends on the petal angle of the desired design point. If at the design point is afterburner operational convergent part petal angle is close to its minimum to expand the nozzle to its maximum area for fan operability. For

the convergent petal angle $5\text{-}40^\circ$ limits taken as a rule of thumb for both axial length restrictions and actuator packaging. For the divergent part $1\text{-}15^\circ$ limits taken as a rule of thumb.

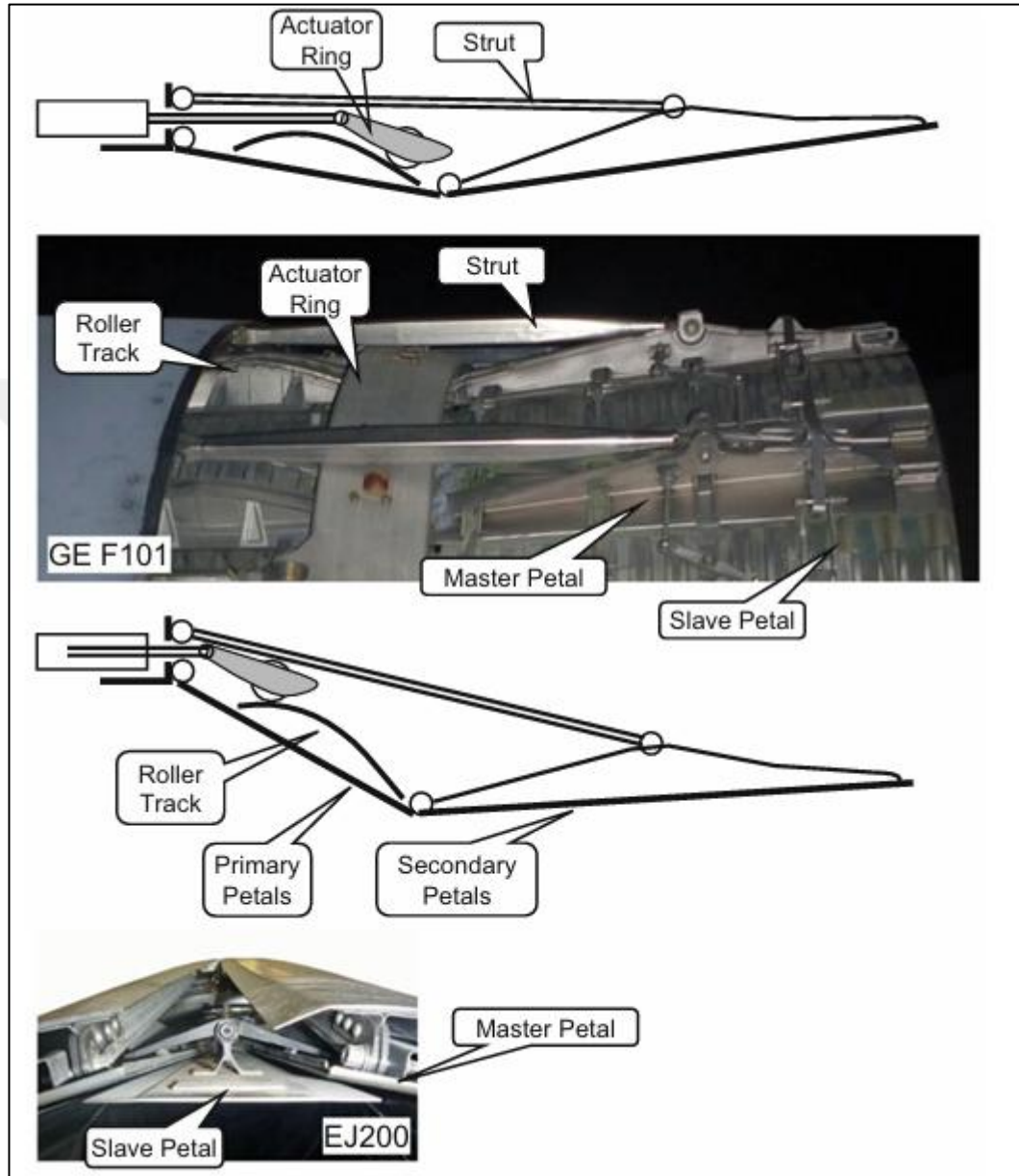


Figure 5.10. Convergent-Divergent Nozzle Representations [2].

6. DESIGN SPACE SOLUTION

Aero-engines are complex systems with many inputs and so many possibilities to achieve similar design goals. Therefore, a simplified approach is needed to narrow down the pool of possibilities. As in literature, OPR, BPR, TIT and MFR are the most influence in the design space of the engine. This chapter is focused on these 4 parameters to obtain a feasible engine for the requirements.

6.1. Determination of Independent & Dependent Performance Parameters

BPR, TIT and MFR are design inputs will be independent parameters already. On the other hand, OPR is the pressure ratio of compressor exit pressure divided by engine inlet pressure. Therefore, it is a combination of inner fan pressure ratio, compressor pressure ratio and compressor interduct pressure loss. For the design case, compressor interduct pressure loss can be taken as constant for all the design possibilities for the start. Thus, OPR reduces to a combination of HPC PR and FPR and in this approach HPC PR is taken as the independent performance parameter.

One more parameter is introduced where literature does not emphasis enough which is P_{160}/P_{060} which is the ratio of the total pressure at the exit of the bypass divided by the total pressure of the core exit. This parameter ensures that the mixing of the two streams creates minimum loss as possible and cooling requirements of the afterburner. P_{160}/P_{060} value being as close as to 1 minimizes the losses and improves the SFC at desired locations. On the other hand, P_{s160}/P_{s060} value being higher than 1 ensures the cooling flow from the afterburner liner into the afterburner combustion area for cooling the liner. Therefore, P_{160}/P_{060} value should be close to 1 and static difference is higher than 1 is an important design criterion.

For this thesis MFR value is taken out of the dependent parameters and taken as constant. For achieving a thrust requirement MFR is critical, but it has limitations from the airframers side. Therefore, maximum possible value is taken for the determination of the engine.

6.2. Determination of Design Point

With the given approach in the thesis, everything needed to create a design point of the engine and 1-D layout are ready. HPC PR, BPR, TIT and P_{160}/P_{060} are chosen as independent parameters so, for the other design parameters some assumptions are made which are given in the next subsection.

6.2.1. Design Point Assumptions

As mentioned in the 6.1 P_{160}/P_{060} is not an input of the engine but chosen as an independent parameter. It is constructed as follows:

$$P_{160} = \text{Outer FPR} * \text{Bypass Loss} \quad (6.1)$$

By taking bypass loss as a constant for the design point iteration, P_{160}/P_{060} is a function of outer fan pressure ratio. For the inner fan pressure ratio, a ratio between outer and inner fan is constructed as follows to keep them close to each other. In the end, outer and inner fan does not exist for the turbomachinery, and it is just a mass averaging along to the span of the fan blade based on the bypass ratio.

$$\frac{FPR_{outer}}{FPR_{inner}} = \text{Constant} \quad (6.2)$$

For simplicity, both P_{160}/P_{060} and FPR_{outer}/FPR_{inner} are taken as 1.

Isentropic efficiencies are used in the cycle and are assumed to have constant polytropic efficiencies by using the same turbomachinery characteristics. Isentropic and polytropic efficiency conversion formulas given below.

$$\eta_{isentropic} = \frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{PR^{\frac{\gamma-1}{\gamma} * \frac{1}{\eta_{polytropic}}} - 1} \quad (6.3)$$

$$\eta_{polytropic} = \frac{\ln PR}{\ln \frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{\eta_{isentropic}}} + 1 \quad (6.4)$$

All the constant values taken for the design iteration given in the **Table 6.1**.

Table 6.1. Constant Design Parameters.

Inlet Mass Flow Rate [kg/s]	122.4
Outer Fan Polytropic Efficiency	89.05%
Inner Fan Polytropic Efficiency	89.89%
HPC Polytropic Efficiency	89.05%
HPT Polytropic Efficiency	85.73%
LPT Polytropic Efficiency	87.98%
CC Pressure Loss	4.50%
CC efficiency	99.50%
LHV [MJ/kg]	43.10
Compressor Interduct Pressure Loss	1.50%
Turbine Interduct Pressure Loss	1.50%
Turbine Exit Pressure Loss	1.50%
Bypass Loss	3.00%
Cold Stream Mixer Pressure Loss	0.00%
Hot Stream Mixer Pressure Loss	0.00%
Mixed Stream Pressure Loss	1.50%
Power Take-Off [kW]	200.00
Overboard Bleed [kg/s]	0.00%
LP Spool Mechanical Efficiency	99.50%
HP Spool Mechanical Efficiency	98.50%
AB Efficiency	92.00%
CD080	0.966
CFG	0.970
HPT Stator SAS [W/W25]	10.00%
HPT Rotor SAS [W/W25]	7.00%
LPT Stator SAS [W/W25]	2.20%
LPT Rotor SAS [W/W25]	1.80%

6.3. Multi-Point Design Optimization

For the selected design point, there are several viable engine cycles that can provide the desired thrust and SFC values. To pick the most suitable engine option, other thrust and SFC requirement points of the engine should be checked as well. For this purpose, multi-point design optimization is used for both after the cycle selection and at the cycle selection phase. In order not to complicate the cycle selection process, generally only one off-design point is checked and optimized accordingly. At the beginning of requirement evaluation, the most difficult point for the thrust point of view was selected. Therefore, the most important point for the SFC being the second point of the design optimization is key to achieving overall best performance for the given requirement list.

6.3.1. Off-Design Assumptions

Like the on-design assumptions some values kept constant for the off-design cases, and they are given in the table below.

Table 6.2. Constant Off-Design Parameters.

Maximum Inlet Mass Flow Rate [kg/s]	122.4
Outer Fan Polytropic Efficiency	89.05%
Inner Fan Polytropic Efficiency	89.89%
HPC Polytropic Efficiency	89.05%
HPT Polytropic Efficiency	85.73%
LPT Polytropic Efficiency	87.98%
CC Pressure Loss	4.50%
CC efficiency	99.50%
LHV [MJ/kg]	43.10
LP Spool Mechanical Efficiency	99.50%
HP Spool Mechanical Efficiency	98.50%
AB Efficiency	92.00%
CD080	0.966
CFG	0.970
HPT Stator SAS [W/W25]	10.00%
HPT Rotor SAS [W/W25]	7.00%
LPT Stator SAS [W/W25]	2.20%
LPT Rotor SAS [W/W25]	1.80%

Normally turbomachinery maps are used in the off-design performance of the engine but, for the conceptual design constant polytropic efficiencies are used which are omitting the tip clearance and secondary loss mechanism over the off-design points for the turbomachinery. For the corrected mass flows, HPT is assumed to be NGV choked from idle to max conditions which is also a requirement for the turbomachinery from the performance perspective for mass flow control of the engine over the flight envelope. For FAN and HPC below curves are used for the corrected mass flow and pressure ratio characteristics of the off design.

Figure 6.1 and **Figure 6.2** used for HPC off-design characteristics for this thesis. 1.00 relative corrected spool speed is taken as the design point of the turbomachinery and for each different engine design this point is scaled with design point values. **Figure 6.3** and **Figure 6.4** is used for FAN off-design characteristics and same procedure is implemented as the HPC.

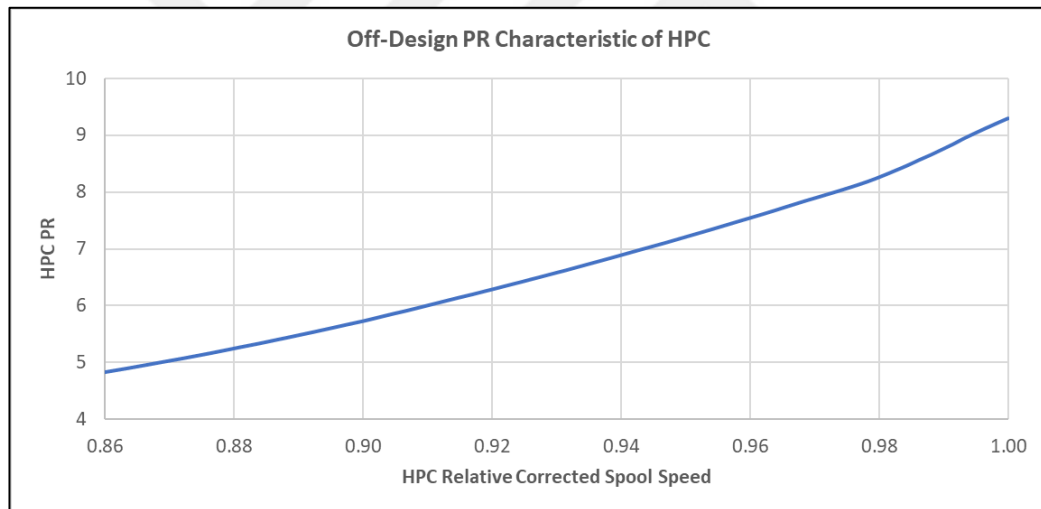


Figure 6.1. HPC PR vs HPC Relative Corrected Spool Speed.

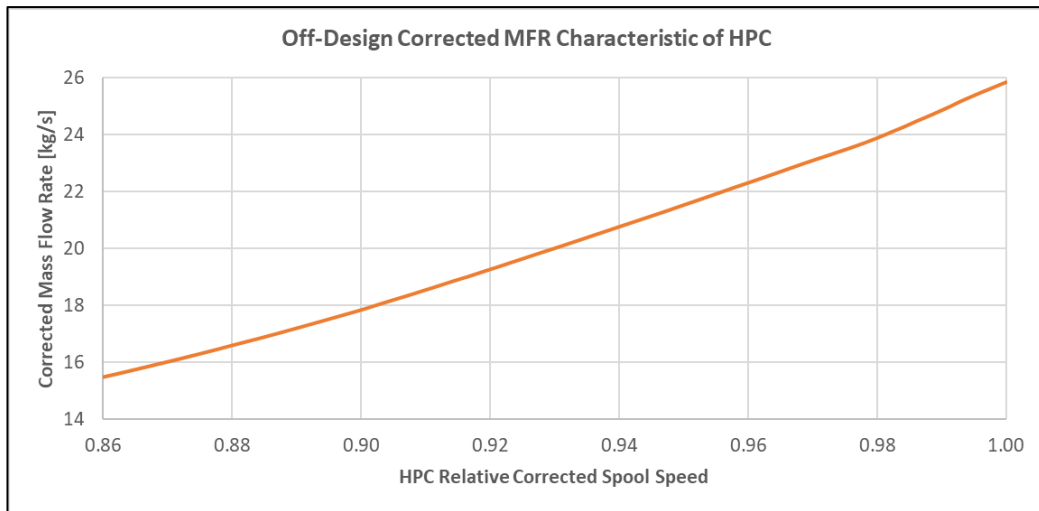


Figure 6.2. HPC Corrected MFR vs HPC Relative Corrected Spool Speed.

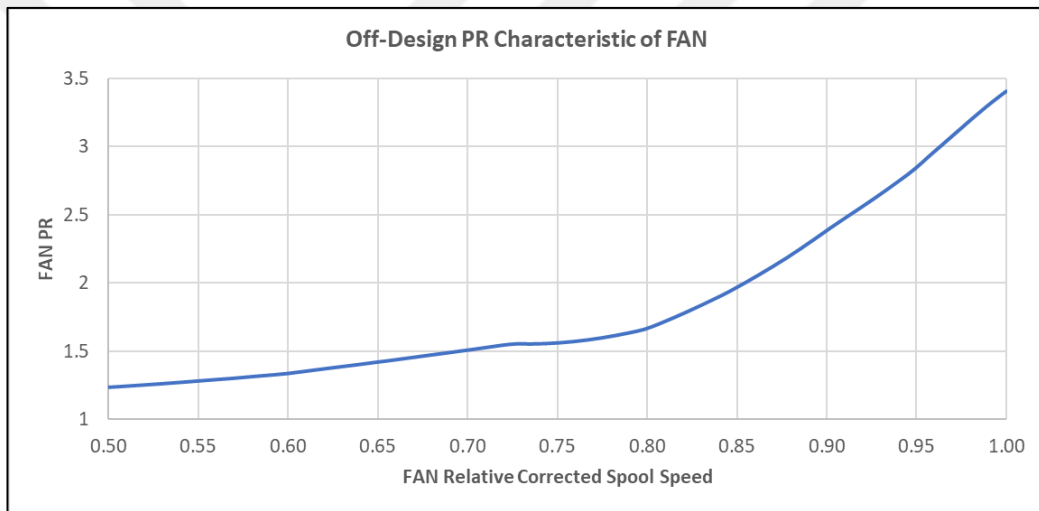


Figure 6.3. FAN PR vs FAN Relative Corrected Spool Speed.

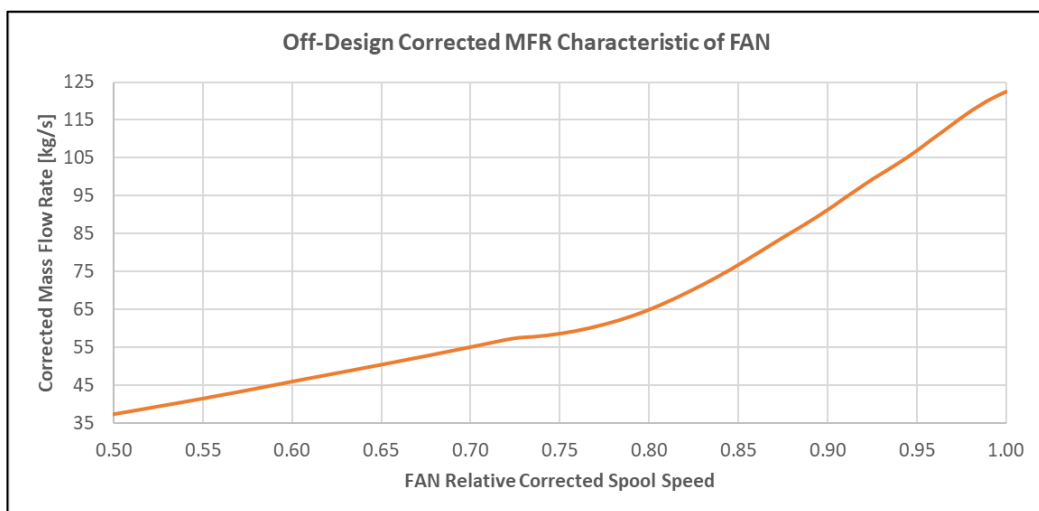


Figure 6.4. FAN Corrected MFR vs FAN Relative Corrected Spool Speed.

For BPR determination, mixer design parameters are used with conservation of momentum. For nozzle control, full expansion for each off-design point is aimed at the sizing constraints mentioned in **Chapter 4.15**. For afterburning operation, stoichiometric limit is taken as the reheat exit temperature limit and implemented as a reheat exit fuel-to-air ratio limit of 0.65 for easy use in the conceptual design phase.

6.3.2. Generating Carpet Plots

By implemented the assumptions, dependents and independents mentioned from **Chapter 6.1** to **Chapter 6.3.1**, design point thrust and cruise SFC curves are plotted for the solution space of the requirement list. Dependents range of iteration given below:

$$10.0 > HPC\ PR > 8.0 \quad (6.5)$$

$$0.8 > BPR > 0.6 \quad (6.6)$$

$$1900\ K > TIT > 1700\ K \quad (6.7)$$

A wider design solution range is studied at the beginning and range is shortened for better visualization of the solution space. Data filtered by HPT PR < 4 for the high-pressure turbine stage reaming as one stage and Thrust-SFC requirements shown in the **Figure 6.5**.

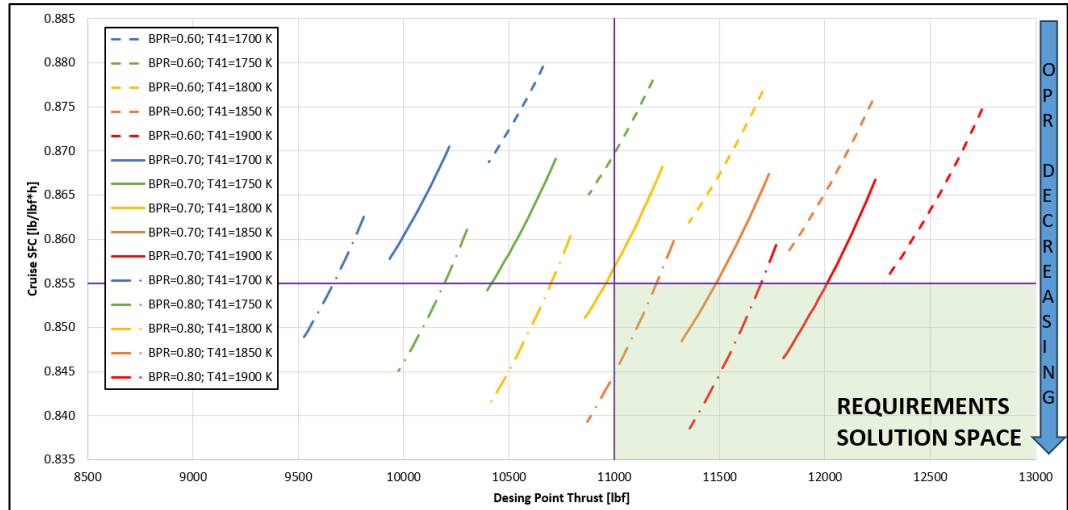


Figure 6.5. Solution Space.

Upper left corner of the solution space is the coldest engines that covers the requirements. Considering the combustor chamber exit temperature will be about 100 K higher than the TIT values, design margins and deterioration will also be considered overall value of 1800 K T41 can reach up to +2000 K gas temperature in the engine for mechanical and material design. Therefore, even if there are other solutions on the solution space, lower temperature options are selected as candidate cycles.

6.3.3. Inspection of Candidate Cycles

As the first inspection of design point, critical parameters are shown in table below.

Table 6.3. Design Point Parameters Comparison.

Parameter	Candidate 1	Candidate 2	Unit
Design Point Thrust	11000	11000	lbf
Design Point SFC	1.0331	1.0306	lb/lbf*h
Cruise SFC	0.8533	0.8503	lb/lbf*h
Outer FAN PR	3.41	3.43	-
Inner FAN PR	3.41	3.43	-
HPC PR	9.3	9.75	-
HPT PR	3.81	3.98	-
LPT PR	2.30	2.31	-
BPR	0.71	0.70	-
T3	916	931	K
T4	1907	1911	K
T41	1810	1815	K

For the cruise SFC, candidate 2 has significant improvement over candidate 1 with the increased OPR. To compensate for this OPR, HPT PR is reached almost 4 which has significant effect on the mechanical design of the HPT. This effect will show its importance in the weight requirement of the engine.

For the temperature side, T4 and T41 values increased about 5 K between candidate 1 and candidate 2 which seem not that important difference. However, T3 value increased about 15 K which can be significant over the flight envelope with increasing flight Mach numbers and ISA conditions this value will increase further, and it has two significant importance in the engine. One of them is the selection of HPC rear stages material which will add cost and weight with these increased T3 values. And the other is T3 and subsequently the HPC interstage bleeds are used for cooling of stator, rotors and discs of the hot section of the engine. With these increases these part's life expectancy could lower significantly.

6.4. 1-D Layout of Candidate Cycles

For the evaluation of both concepts, turbomachinery loadings are kept the same due to the same polytropic efficiency assumptions for design point. Layouts for both cycles are given below with compression and expansion system smith charts and sizing assumptions for each component.

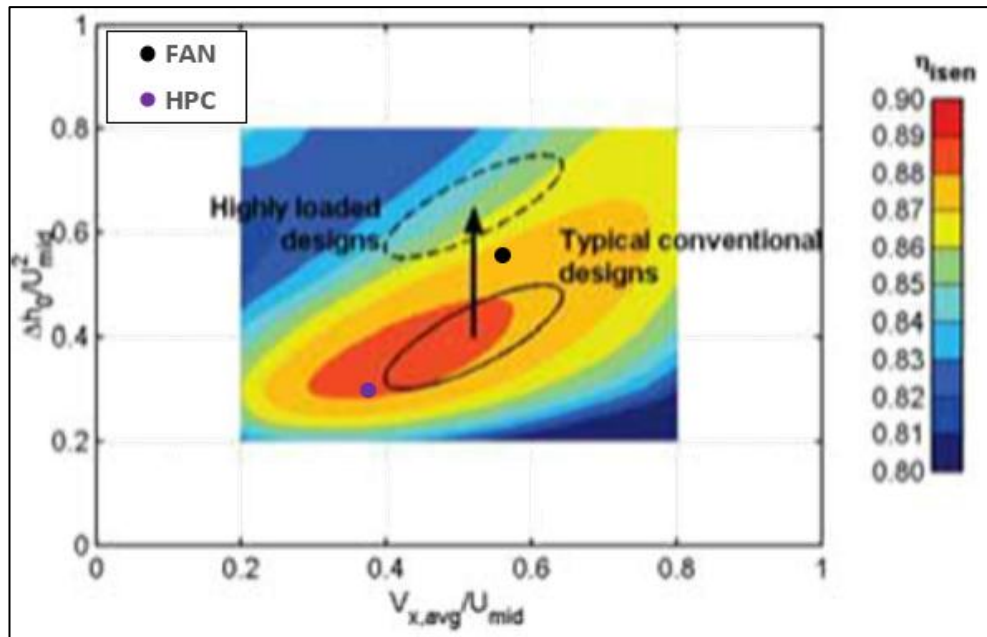


Figure 6.6. Compression System Smith Chart Candidate Cycle 1.

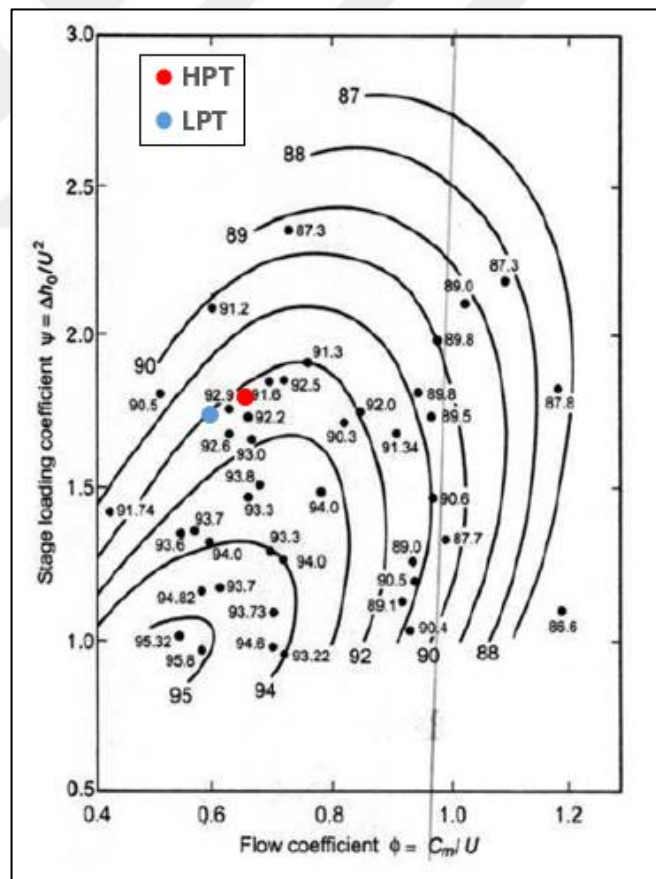


Figure 6.7. Expansion System Smith Chart Candidate Cycle 1.

INLET	M_2_axial	0.681	-
	M_2_abs	0.688	-
	rh_rt	0.450	-
	Tip Radius	0.475	[m]
	Hub Radius	0.21375	[m]
	Swirl	8	[°]
	Cd	1	-
	Axial Velocity	232.7	[m/s]
	Theta Velocity	32.7	[m/s]
	Absolute Velocity	234.9	[m/s]
	Tip Relative Mach	1.318	-
	Flow coeff.	0.768	-
	Mass Flux	216.53	[kg/s*m^2]
Inner FAN	M_St21	0.40	-
	LPC ID exit	0.633	[m]
	LPC OD exit	0.897	[m]
	h/t exit	0.705	-
Outer FAN	M_St13	0.45	-
	M_St16	0.300	-
	Stator AR	3.40	-
	Rotor AR	1.90	-
	Ax. Cl. Ratio	0.25	-
	Axial Length	0.84531	[m]
Interduct	V Splitter/Fan Exit R	0.30	-
	V Splitter	0.03973	[m]
	Radius Drop	0.12073	[m]
	Radius Drop/Length	0.30	-
	H Splitter/Length	0.20	-
	Length	0.40244	[m]
	H Splitter	0.08049	[m]
HPC	M_St25	0.45	-
	HPC ID inlet	0.430	[m]
	HPC OD inlet	0.617	[m]
	rh/rt inlet	0.697	-
	M_St3	0.27	-
	HPC ID exit	0.506	[m]
	HPC OD exit	0.541	[m]
	rh/rt exit	0.934	-
	Stator AR	2.35	-
	Rotor AR	2.30	-
	Ax. Cl. Ratio	0.25	-
	Axial Length	0.69428	[m]
CC	M_St31	0.2	-
	r_exit	0.161	[m]
	r_dome	0.096	[m]
	Volume	0.010	[m^3]
	Length	0.193	[m]
	Loading	2.21	[kg/s*atm^1.8*m^3]
	Intensity	228.01	[MW/atm*m^3]
	Time	1.260	[ms]
HPT	M_St4	0.15	-
	HPT ID inlet	0.620	[m]
	HPT OD inlet	0.698	[m]
	h/t HPT inlet	0.888	-
	M_St43	0.50	-
	HPT ID exit	0.61000	[m]
	HPT OD exit	0.70818	[m]
	h/t HPT exit	0.86136	-
	NGV AR	1.10	-
	Rotor AR	1.10	-
	Ax. Cl. Ratio	0.45	-
	Axial Length	0.09813	[m]
LPT	M_St44	0.50	-
	LPT ID inlet	0.610	[m]
	LPT OD inlet	0.717	[m]
	h/t LPT inlet	0.851	-
	M_St5	0.50	-
	LPT ID exit	0.490	[m]
	LPT OD exit	0.735	[m]
	h/t LPT exit	0.667	-
	NGV AR	1.70	-
	Rotor AR	1.70	-
	Ax. Cl. Ratio	0.50	-
	Axial Length	0.25896	[m]
AFTER BURNER	Loading	5.61	[kg/s*atm^1.8*m^3]
	Length	1.048	[m]
	Volume	0.57921	[m^3]
NOZZLE	NPR	11.48	-
	Conv. Petal Angle	15.00	°
	Conv. Length Ax.	0.47054	[m]
	A9/A8	2.14	-
	Div. Petal Angle	10.00	°
	Div. Length Ax.	0.76706	[m]

Figure 6.8. Sizin Assumptions of Candidate Cycle 1.

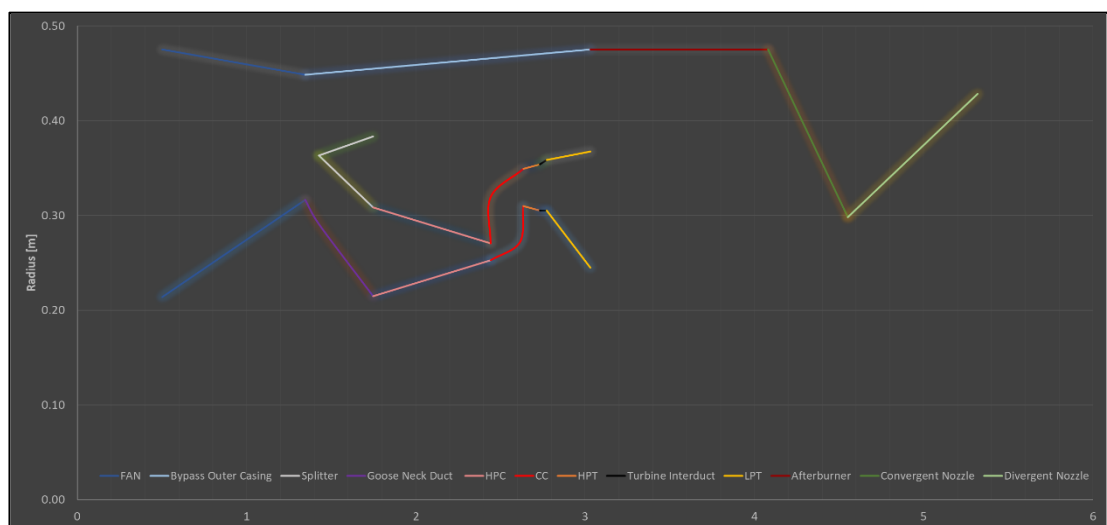


Figure 6.9. Layout of Candidate Cycle 1.

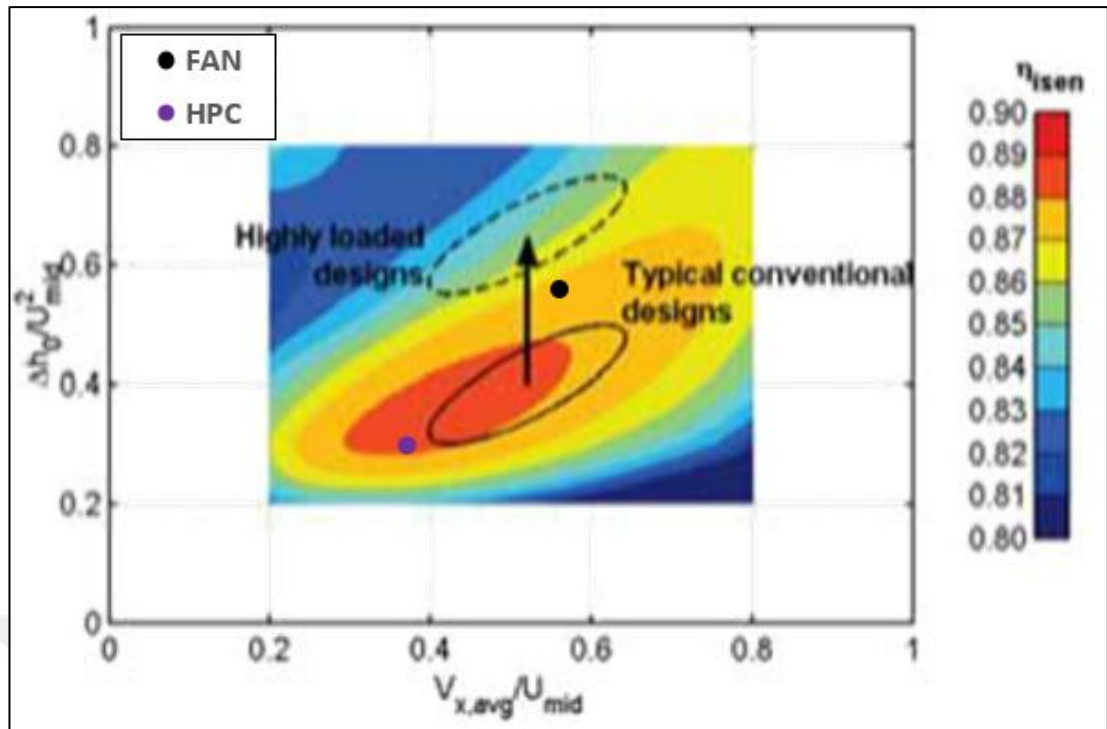


Figure 6.10. Compression System Smith Chart Candidate Cycle 2.

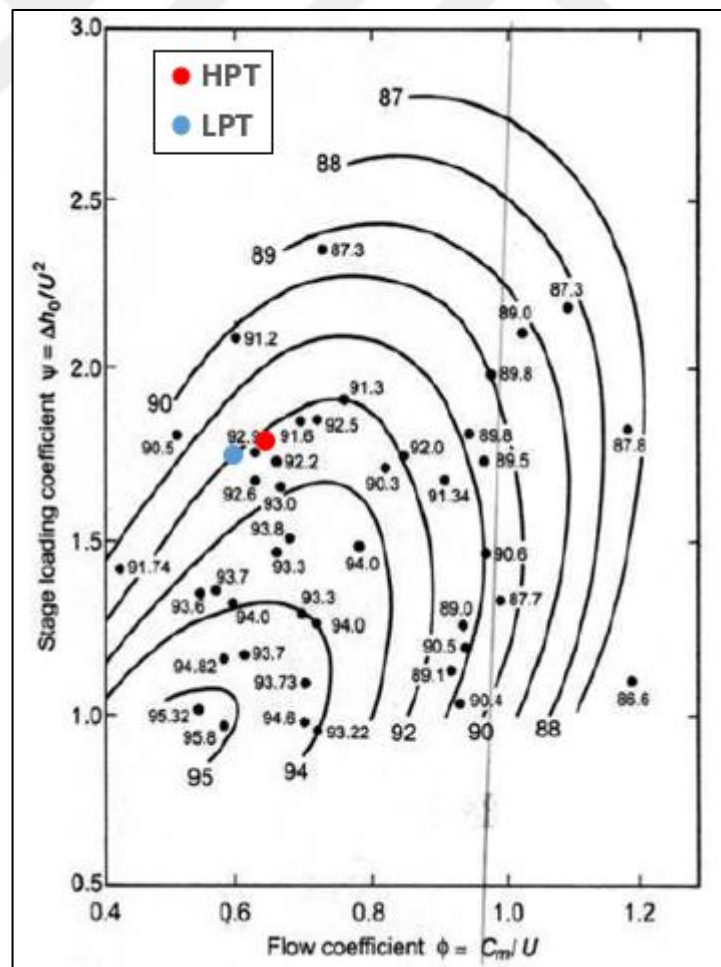


Figure 6.11. Expansion System Smith Chart Candidate Cycle 2.

INLET	M_2_axial	0.681	-
	M_2_abs	0.688	-
	rh_rt	0.450	-
	Tip Radius	0.475	[m]
	Hub Radius	0.21375	[m]
	Swirl	8	[°]
	Cd	1	-
	Axial Velocity	232.7	[m/s]
	Theta Velocity	32.7	[m/s]
	Absolute Velocity	234.9	[m/s]
	Tip Relative Mach	1.318	-
	Flow coeff.	0.768	-
	Mass Flux	216.53	[kg/s*m^2]
Inner FAN	M_St21	0.40	-
	LPC ID exit	0.633	[m]
	LPC OD exit	0.897	[m]
	h/t exit	0.705	-
Outer FAN	M_St13	0.45	-
	M_St16	0.300	-
	Stator AR	3.40	-
	Rotor AR	1.90	-
	Ax. Cl. Ratio	0.25	-
Interduct	Axial Length	0.84523	[m]
	V Splitter/Fan Exit R	0.30	-
	V Splitter	0.03970	[m]
	Radius Drop	0.12079	[m]
	Radius Drop/Length	0.30	-
HPC	H Splitter/Length	0.20	-
	Length	0.40263	[m]
	H Splitter	0.08053	[m]
	M_St25	0.45	-
	HPC ID inlet	0.430	[m]
HPT	HPC OD inlet	0.617	[m]
	rh/rt inlet	0.697	-
	M_St3	0.27	-
	HPC ID exit	0.506	[m]
	HPC OD exit	0.540	[m]
	rh/rt exit	0.937	-
	Stator AR	2.35	-
	Rotor AR	2.30	-
	Ax. Cl. Ratio	0.25	-
	Axial Length	0.69033	[m]
CC	M_St31	0.2	-
	r_exit	0.157	[m]
	r_dome	0.094	[m]
	Volume	0.010	[m^3]
	Length	0.189	[m]
	Loading	2.05	[kg/s*atm^1.8*m^3]
	Intensity	230.23	[MW/atm*m^3]
	Time	1.232	[ms]
LPT	M_St4	0.15	-
	HPT ID inlet	0.620	[m]
	HPT OD inlet	0.695	[m]
	h/t HPT inlet	0.893	-
	M_St43	0.50	-
	HPT ID exit	0.61000	[m]
	HPT OD exit	0.70675	[m]
	h/t HPT exit	0.86311	-
	NGV AR	1.10	-
	Rotor AR	1.10	-
AFTER BURNER	Ax. Cl. Ratio	0.45	-
	Axial Length	0.09537	[m]
	M_St44	0.50	-
	LPT ID inlet	0.610	[m]
	LPT OD inlet	0.716	[m]
	h/t LPT inlet	0.852	-
	M_St5	0.50	-
	LPT ID exit	0.490	[m]
	LPT OD exit	0.733	[m]
	h/t LPT exit	0.668	-
NOZZLE	NGV AR	1.70	-
	Rotor AR	1.70	-
	Ax. Cl. Ratio	0.50	-
	Axial Length	0.25628	[m]
NOZZLE	Loading	5.88	[kg/s*atm^1.8*m^3]
	Length	1.029	[m]
	Volume	0.54688	[m^3]
	NPR	11.59	-
NOZZLE	Conv. Petal Angle	15.00	°
	Conv. Length Ax.	0.44643	[m]
	A9/A8	2.15	-
	Div. Petal Angle	10.00	°
NOZZLE	Div. Length Ax.	0.76938	[m]

Figure 6.12. Sizin Assumptions of Candidate Cycle 2.

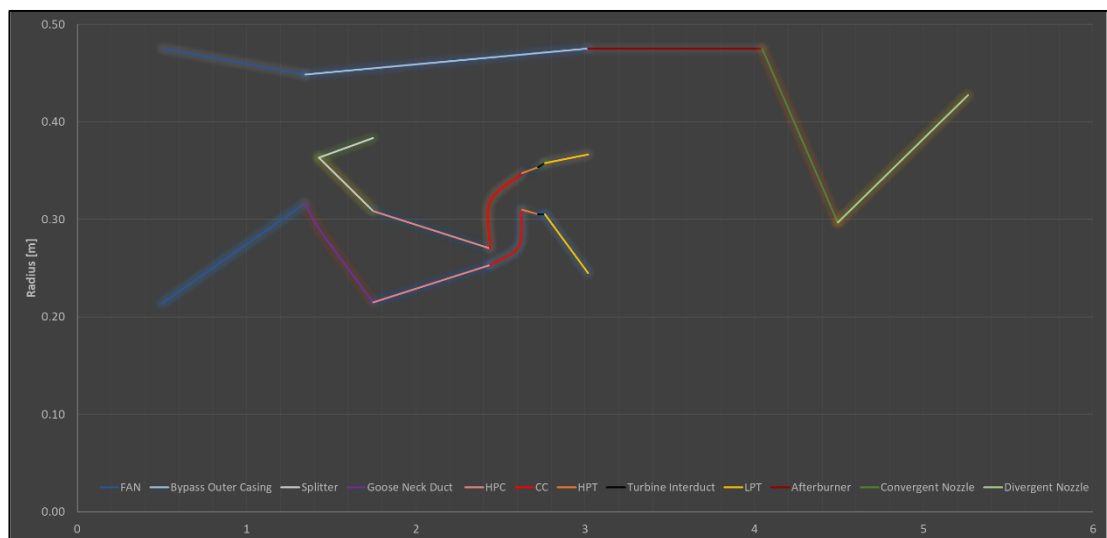


Figure 6.13. Layout of Candidate Cycle 2.

6.5. Iteration for Determining the Cycle

At this point, performance cycle with initial aerodynamic work is used to create the flow path. TEI weight estimation tool is used to estimate the weight of the engine module by module. In the **Figure 6.14** inputs of the weight estimation tool are given in block figures. When the inputs break down into engine parameters, at least 86 inputs are needed to estimate the weight within the error rate of 5%.

Mass flow, turbomachinery inlet and outlet temperature, pressures, corrected mass flows, pressure ratio, reference pressure and temperatures, isentropic efficiencies, modular powers and shaft powers provided by the performance cycle. Shaft speed and shaft overspeed are provided by the initial aerodynamic work in the methodology.

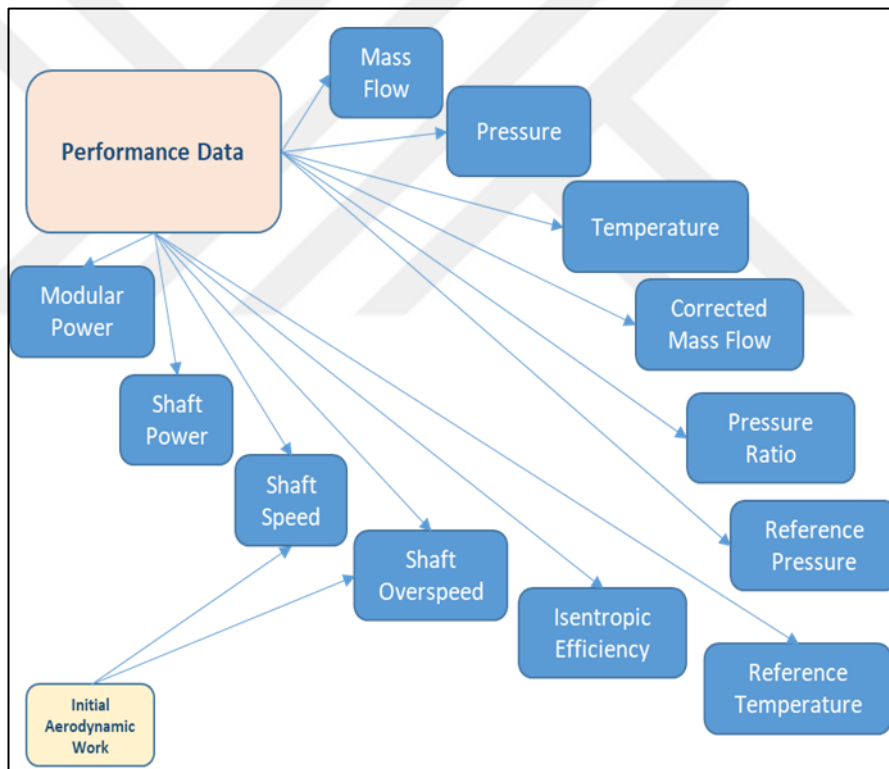


Figure 6.14. TEI Weight Estimation Tool Inputs.

4 versions of the estimation tool with their extend of inputs given in **Figure 6.15**. With the methodology given in **Chapter 4** and **Chapter 5** inputs for the 4. version of the tool can be provided.

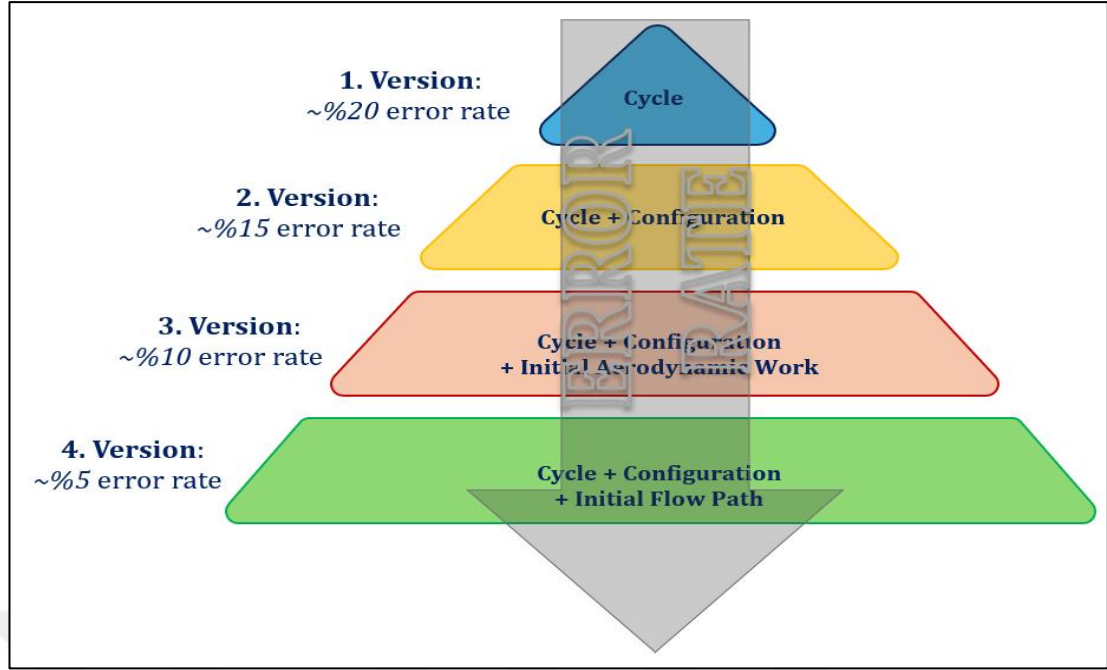


Figure 6.15. TEI Weight Estimation Tool Versions.

For the cost estimation following formulas are used for both development of derivative and new engine costs.

$$Cost_{derivate} = e^{-39.422+5.066*\ln(T_{41}-460)-1.299*\ln(SFC)+0.582*\ln(FSTH)} \quad (6.8)$$

$$Cost_{new} = e^{-24.429+4.027*\ln(T_{41}-460)} \quad (6.9)$$

For the production cost estimation following formulas are used where slope based on the U.S. Air Force database.

$$\ln(slope) = 0.168 - 0.116 * \ln(TWR) \quad (6.10)$$

$$Cost_1 = e^{-10.40-8.550*\ln(slope)+0.482*AB+1.162*\ln(T_{41}-460)+0.261*\ln(W)} \quad (6.11)$$

$$Cost_{400} = e^{-10.40+0.482*AB+1.162*\ln(T_{41}-460)+0.261*\ln(W)} \quad (6.12)$$

Equations from (6.8) to (6.12) based on the works of RAND Corporation [12].

7. RESULTS AND DISCUSSION

The first subsection of this chapter shows the performance, sizing and weight estimation accuracy of the thesis with existing engines where F110- GE-129 is taken as reference. Cost estimation part is taken from [12] and its already proven itself with comparison to existing turbofan engines.

In the rest of the results section, 2 candidate cycles are examined from the weight and cost perspective considering 2 requirement points. Additionally, the rest of the requirement set is examined for further evaluation.

7.1. Benchmark Comparison

Inlet mass flow and bypass ratio of the performance cycle selected according to **Figure 7.1**. Maximum diameter given in to **Figure 7.1** includes the accessory of the engines, therefore, inlet diameter of the engine calculated with assumed inlet conditions. 1-D layout generated according to thesis study compared with the given overall length of the engine.

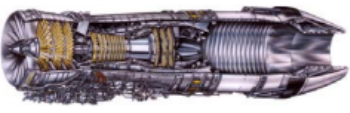
	Performance Specifications (Sea level/standard day)		
	F110-GE-129	English	SI
	Thrust class	29,000 lb	129 kN
	Length	181.9 in	4.6 m
	Airflow	270 lb/sec	122.4 kg/sec
	Maximum diameter	46.5 in	1.2 m
	Bypass ratio	0.76	0.76

Figure 7.1. F110-GE-129 Performance Specifications [15].

With the combination of 1-D layout aerodynamic effects on the efficiency assumptions thrust and SFC class of the engine is estimated and given in the **Figure 7.2** and detailed station outputs of the performance cycle given in **Figure 7.3**.

Net Thrust (SI)	129.99	[kN]
Net Thrust (IMP)	29224.02	[lbf]
SFC (SI)	53.91	[g/(kN*s)]
SFC (IMP)	1.90	[lb/(lbf*h)]
P16/P6	0.9627	-
Wf	1.58787	[kg/s]
WFR	5.42069	[kg/s]

Figure 7.2. Main Performance Outputs of the Benchmark Engine.

	W[kg/s]	T [K]	P [kPa]	Wc [kg/s]	FAR [-]	Ts [K]	Ps [kPa]	γ [-]	ρ [kg/m ³]	V [m/s]	M [-]	Area [m ²]
1	122.400	288.15	101.325	122.400	0.0000	288.15	101.325	1.400	1.225	0.00	0.000	-
2	122.400	288.15	101.325	122.400	0.0000	263.21	73.828	1.400	0.977	223.77	0.688	0.5653
13	52.855	422.06	334.373	19.384	0.0000	405.85	291.136	1.395	2.499	181.39	0.450	0.1166
21	69.545	424.44	344.505	24.825	0.0000	411.46	308.677	1.394	2.613	162.32	0.400	0.1639
25	69.545	424.44	339.337	25.203	0.0000	408.13	295.465	1.395	2.522	181.89	0.450	0.1493
3	54.941	835.79	3110.028	3.049	0.0000	825.22	2961.488	1.351	12.502	152.75	0.270	0.0288
4	56.529	1770.00	2970.076	4.780	0.0289	1764.36	2927.551	1.284	5.781	120.96	0.150	0.0803
41	63.483	1676.46	2970.076	5.224	0.0257	1460.71	1623.324	1.295	3.872	736.99	1.000	0.0221
42	63.483	1299.12	800.754	17.057	0.0257	1251.49	682.336	1.304	1.899	342.27	0.500	0.0976
43	68.352	1268.29	800.754	18.145	0.0238	1221.57	682.127	1.306	1.945	338.36	0.500	0.1038
45	69.882	1255.57	788.742	18.739	0.0227	1209.00	671.796	1.308	1.936	489.43	0.500	0.1071
49	69.882	1057.21	342.026	39.654	0.0233	1016.49	290.922	1.320	0.997	310.35	0.500	0.2258
5	71.133	1050.31	342.026	40.233	0.0228	1009.77	290.898	1.321	1.004	309.41	0.500	0.2291
6	71.133	1050.31	334.201	41.175	0.0228	1031.04	309.158	1.320	1.045	213.71	0.342	0.3182
16	52.855	422.06	321.747	20.145	0.0000	417.33	309.158	1.394	2.581	98.07	0.240	0.2088
64	123.988	797.82	344.346	60.708	0.0130	785.99	324.767	1.341	1.439	163.14	0.296	0.5270
7	129.409	2000.00	279.650	123.530	0.0573	2046.34	246.604	1.259	0.420	386.93	0.450	0.5270
8	129.409	2000.00	279.650	123.530	0.0573	1767.70	154.340	1.265	0.304	801.19	1.000	0.5310
9	129.409	2000.00	279.650	123.530	0.0573	1617.72	101.325	1.270	0.218	1025.03	1.335	0.5786

Figure 7.3. Station Details of the Benchmark Engine.

1-D layout that generated parallel to the performance cycle for the benchmark engine given in **Figure 7.5** satisfies the given length in the **Figure 7.1**. Also, module by module breakdown of the dimensional envelope can be compared with **Figure 7.4**. Module by module 1-D layout offers more accurate weight estimation than the existing weight estimation models in the literature.

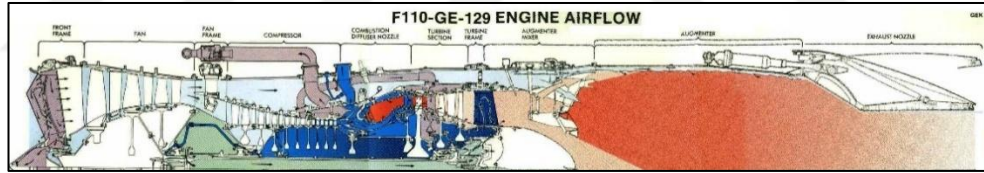


Figure 7.4. F110-GE-129 1-D Layout [16].

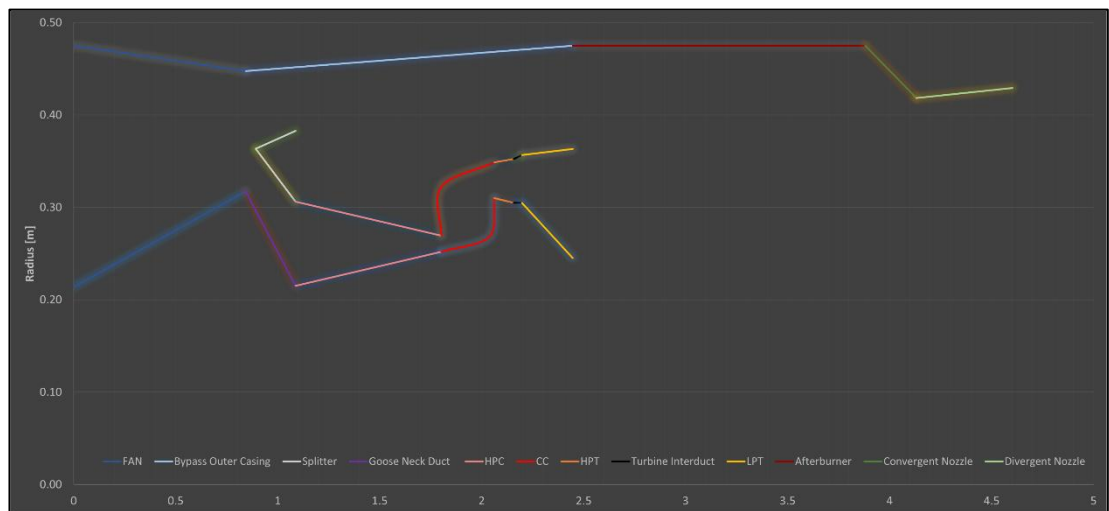


Figure 7.5. 1-D Layout of the Benchmark Engine.

In **Figure 7.6** dry weight of the benchmark engine given as 3980 lbs where version 4 of the weight estimation tool estimates the weight 3932.4 lbs yielding an error of 1.2%.

Technical Data	
Max. thrust at sea level	29,000 lbf
Pressure ratio (ratio at max power)	30.7:1
Bypass ratio	0.76:1
Length	15.1 ft
Dry weight	3,980 lbs

Figure 7.6. F110-GE-129 Technical Data [17].

Version 4	Weight [kg]	Weight [lbs]
Estimation	1783.7	3932.4
F110-129 Data	1805.3	3980.0
Error	1.2%	

Figure 7.7. Weight Estimation of the Benchmark Engine.

Cost estimation part taken directly from RAND Corporation report [12] which is based on the data provided by the United States Air Force. Due to the lack of information in the literature about the cost of the existing engines this report has been taken as the best option for the cost estimation. It is used to add value to the comparison of the candidate cycles.

7.2. Requirements Comparison

For both performance cycles, comparison with requirement given in the **Figure 7.8** and **Figure 7.9**.

Candidate 1	Altitude [feet]	Flight Mach Number	Requirement Thrust [lbf]	Requirement SFC [lb/lbf*h]	Cycle Thrust [lbf]	Cycle SFC [lb/lbf*h]	Thrust Difference [%]	SFC Difference [%]
Part Power	30000	0.85	3500	0.855	3500	0.8533	0.0%	-0.2%
Max Dry	20000	0.7	9000	0.860	9400	0.8626	4.4%	0.3%
Max Dry	35000	1.5	11000	1.050	11000	1.0331	0.0%	-1.6%
Max Dry	0	0	19000	0.700	19312	0.6917	1.6%	-1.2%
Max AB	0	0	32300	1.900	33762	1.8933	4.5%	-0.4%
Max AB	0	0.9	38300	2.400	38500	2.4273	0.5%	1.1%
Max AB	0	1.2	39000	2.600	38942	2.6143	-0.1%	0.6%
Max AB	30000	0.9	15000	2.150	15104	2.1229	0.7%	-1.3%
Max AB	30000	1.3	17000	1.440	16976	1.4250	-0.1%	-1.0%
Max AB	40000	0.7	7700	2.450	7812	2.4727	1.5%	0.9%
Max AB	50000	0.9	5750	2.550	5802	2.5425	0.9%	-0.3%

Figure 7.8. Candidate 1 Requirement Comparison.

Candidate 2	Altitude [feet]	Flight Mach Number	Requirement Thrust [lbf]	Requirement SFC [lb/lbf*h]	Cycle Thrust [lbf]	Cycle SFC [lb/lbf*h]	Thrust Difference [%]	SFC Difference [%]
Part Power	30000	0.85	3500	0.855	3500	0.8503	0.0%	-0.5%
Max Dry	20000	0.7	9000	0.860	9410	0.8595	4.6%	-0.1%
Max Dry	35000	1.5	11000	1.050	11000	1.0306	0.0%	-1.8%
Max Dry	0	0	18500	0.700	18860	0.6867	1.9%	-1.9%
Max AB	0	0	32300	1.900	33405	1.9777	3.4%	4.1%
Max AB	0	0.9	38300	2.400	38427	2.5097	0.3%	4.6%
Max AB	0	1.2	39000	2.600	38823	2.6970	-0.5%	3.7%
Max AB	30000	0.9	15000	2.150	15691	2.1144	4.6%	-1.7%
Max AB	30000	1.3	17000	1.440	17036	1.4211	0.2%	-1.3%
Max AB	40000	0.7	7700	2.450	7891	2.5332	2.5%	3.4%
Max AB	50000	0.9	5750	2.550	5874	2.6030	2.2%	2.1%

Figure 7.9. Candidate 2 Requirement Comparison.

Candidate 2, with higher OPR, has better SFC performance at dry conditions but for the wet conditions Candidate 1 has improved SFC performance. At dry conditions, high OPR and high BPR improve the SFC of the engine, whereas, for wet conditions high BPR means that less clean air (bypass air with no prior combustion gases) goes into the afterburner. Therefore, candidate 2 has better dry SFC performance while candidate 1 has better wet SFC performance.

Considering that dry SFC requirements are comply for both cycles, and wet SFC requirements challenging for both cycles candidate 1 has overall better coverage of the requirement set from the performance perspective.

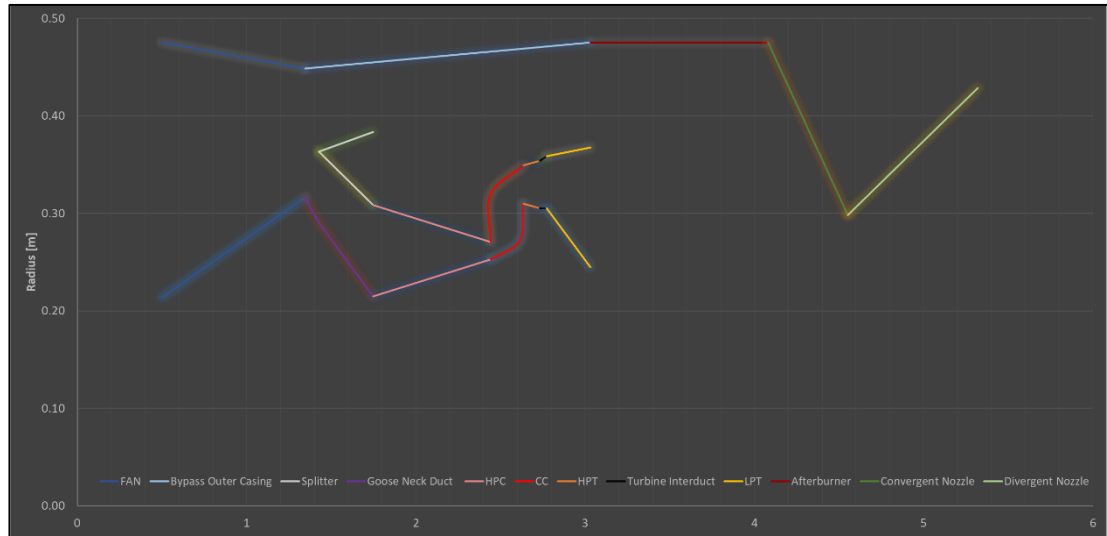


Figure 7.10. Candidate 1 Layout.

Module	Weight [kg]	Weight [lbs]
FAN & HPC	523.3	1153.7
CC	112.7	248.5
Turbines	370.6	817.0
Frames	380.3	838.4
AB + Exhaust	215.2	474.4
Accessories	259.1	571.2
TOTAL	1861.2	4103.2

Figure 7.11. Candidate 1 Weight Breakdown.

Development [\$millions]		Production [\$millions]	
New	Derivative	First	400th
3173.340	187.915	13.686	7.437

Figure 7.12. Candidate 1 Cost Estimation.

From **Figure 7.10** to **Figure 7.12** candidate 1 layout, weight breakdown and cost estimation according to these two inputs are given. Both newly developed engine and

derivative engine costs are calculated but for this study derivative engine of the F110 is considered.

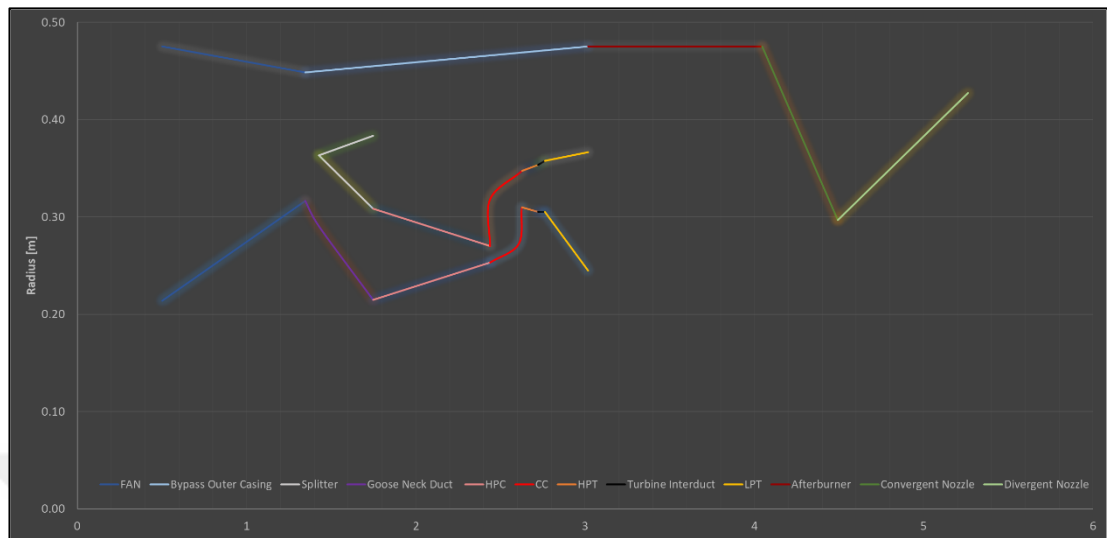


Figure 7.13. Candidate 2 Layout.

Module	Weight [kg]	Weight [lbs]
FAN & HPC	522.4	1151.7
CC	108.9	240.1
Turbines	354.8	782.2
Frames	380.3	838.4
AB + Exhaust	213.2	470.0
Accessories	258	568.8
TOTAL	1837.8	4051.7

Figure 7.14. Candidate 2 Weight Breakdown.

Development [\$millions]		Production [\$millions]	
New	Derivative	First	400th
3214.706	180.476	13.865	7.440

Figure 7.15. Candidate 2 Cost Estimation.

From **Figure 7.13** to **Figure 7.15** candidate 2 layout, weight breakdown and cost estimation according to these two inputs are given same as candidate 1. Considering both engines, candidate 1 has higher weight of about 25 kg which for a twin engine supersonic aircraft means 150-250 kg more weight in addition to the aircraft for keeping desired center of gravity in the same location. Also, cost of derivative engine development of candidate 2 is lower than candidate 1.

In the conceptual design phase with only performance cycle outputs respect to thrust and SFC requirement set, candidate 1 would be the chosen engine for the conceptual design phase. In between conceptual design and detailed design phase aerodynamic work will be more accurate with the initial dimensional envelope constraints, and it effects the performance cycle mainly for the efficiency targets. In addition, weight requirement of the engine is as important as the performance of the engine and in the detailed design phase weight reductions and cost reductions alter the initial performance cycle. With the generation of flow path and advanced aerodynamic works, candidate 2 is the possible outcome of the detailed design phase of the engine development.

By considering the flow path of the engine performance cycle would be less effected in the detailed design phase and with weight breakdown of the engine module by module shows that candidate 2 has best coverage of the entire requirement set of the engine.

From whole engine design perspective candidate 2 would be chosen in the conceptual design phase.

7.3. Final 1-D Layout

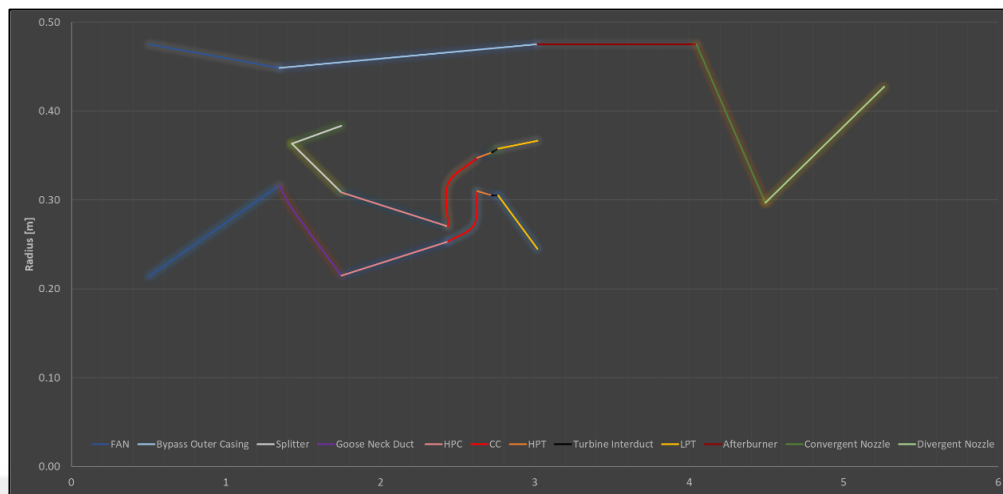


Figure 7.16. Final Layout of the Selected Cycle.

7.4. Key Performance Parameters

Final performance cycle stations outputs given in the Figure 7.17.

	W[kg/s]	T [K]	P [kPa]	Wc [kg/s]	FAR [-]	Ts [K]	Ps [kPa]	γ [-]	ρ [kg/m ³]	V [m/s]	M [-]	Area [m ²]
010	97.800	317.53	87.561	118.802	0.0000	218.81	23.842	1.401	0.380	444.97	1.500	-
020	97.800	317.53	84.985	122.403	0.0000	290.06	61.916	1.400	0.744	234.94	0.688	0.5653
130	40.271	469.95	291.499	17.876	0.0000	452.06	253.900	1.391	1.957	191.17	0.450	0.1077
021	57.529	468.26	291.499	25.492	0.0000	454.07	261.255	1.391	2.004	170.29	0.400	0.1685
025	57.529	468.26	287.127	25.880	0.0000	450.44	250.088	1.391	1.934	190.83	0.450	0.1535
030	45.448	930.73	2799.485	2.956	0.0000	919.25	2666.586	1.343	10.106	160.71	0.270	0.0280
040	46.868	1913.00	2673.508	4.577	0.0312	1907.01	2635.372	1.279	4.814	125.51	0.150	0.0769
041	52.621	1814.96	2673.508	5.005	0.0277	1585.27	1463.966	1.290	3.217	766.10	1.000	0.0212
042	52.621	1392.59	671.922	17.445	0.0277	1342.45	572.922	1.299	1.487	353.73	0.500	0.1001
043	56.648	1361.98	671.922	18.572	0.0257	1312.73	572.754	1.300	1.520	349.96	0.500	0.1064
045	57.914	1348.84	661.844	19.183	0.0245	1299.72	564.078	1.302	1.512	321.18	0.500	0.1099
049	57.914	1137.08	285.053	40.894	0.0251	1094.16	242.643	1.314	0.773	321.18	0.500	0.2334
050	58.949	1130.07	285.053	41.497	0.0247	1087.32	242.622	1.315	0.777	320.27	0.500	0.2368
060	58.949	1130.07	280.777	42.129	0.0247	1115.24	265.755	1.313	0.830	189.09	0.292	0.3757
160	40.271	469.95	282.754	18.429	0.0000	461.84	265.755	1.390	2.005	128.78	0.300	0.1560
064	99.220	878.51	280.429	62.597	0.0145	865.10	263.679	1.332	1.062	175.58	0.305	0.5317
070	99.220	878.51	280.427	62.598	0.0145	2046.34	246.631	1.287	0.420	386.93	0.450	0.5317
080	99.220	878.51	276.223	63.551	0.0145	749.34	148.067	1.350	0.688	538.89	1.000	0.2675
090	99.220	878.51	276.223	63.551	0.0145	459.15	23.842	1.383	0.181	955.54	2.239	0.5740

Figure 7.17. Stations Performance Outputs.

Candidate 2	Altitude [feet]	Flight Mach Number	Requirement Thrust [lbf]	Requirement SFC [lb/lbf*h]	Cycle Thrust [lbf]	Cycle SFC [lb/lbf*h]	Thrust Difference [%]	SFC Difference [%]
Part Power	30000	0.85	3500	0.855	3500	0.8503	0.0%	-0.5%
Max Dry	20000	0.7	9000	0.860	9410	0.8595	4.6%	-0.1%
Max Dry	35000	1.5	11000	1.050	11000	1.0306	0.0%	-1.8%
Max Dry	0	0	18500	0.700	18860	0.6867	1.9%	-1.9%
Max AB	0	0	32300	1.900	33405	1.9777	3.4%	4.1%
Max AB	0	0.9	38300	2.400	38427	2.5097	0.3%	4.6%
Max AB	0	1.2	39000	2.600	38823	2.6970	-0.5%	3.7%
Max AB	30000	0.9	15000	2.150	15691	2.1144	4.6%	-1.7%
Max AB	30000	1.3	17000	1.440	17036	1.4211	0.2%	-1.3%
Max AB	40000	0.7	7700	2.450	7891	2.5332	2.5%	3.4%
Max AB	50000	0.9	5750	2.550	5874	2.6030	2.2%	2.1%

Figure 7.18. Requirement Coverage of the Selected Cycle.

8. CONCLUSION

In this chapter, conclusions of the requirement evaluation, design and off-design performance calculation methodology, flow path generation, weight and cost estimation methodology are evaluated.

8.1. General Conclusions

Evaluation of the requirements is crucial for determining the design point of the engine and simplifying the approach by omitting the pressure term can be misleading. When **Table 3.2** is examined for a simpler approach other than 35000 feet 1.5 Mach is not a key point for the engine design. But, when the **Figure 7.8** and **Figure 7.9** are examined SL 0.9 Mach, SL 1.2 Mach and 30000 feet 1.3 Mach as much as difficult for the engine as 35000 feet 1.5 Mach. More complex evaluation that does not omit the pressure term does highlight these points as difficult points as much as 35000 feet 1.5 Mach. Therefore, the second approach is more suitable for something as critical as requirement evaluation.

For the design and off-design performance calculations fundamental equations are quite enough when implemented correctly. For conceptual design, constant polytropic efficiencies and compressions system curves are enough to compare different designs with the 1-D sizing backing the selected design assumptions, such as turbomachinery loadings, combustion efficiencies, dimensional limitations of the engine mass flow. Additionally, 1-D layout with extensive performance calculations offers valuable information for the more accurate weight and cost estimation which is critical to decide the conceptual engine design.

Weight estimation with modular breakdown is important to decide the detailed design engine configuration from the conceptual design phase of the engine. Considering the long development time of aero engines, methodology given in the thesis reduces the engineering and project time of the engine development phase.

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