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Turbofan design for the commercial aircraft

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Contents

Statement

This thesis was an examination .The remarks made during the defense of this project are not recorded in this document.

Abstract

In the first part of this work ,an overview is presented of the powerplant used to power the AIRBUS A-380 and the technologies used to construct these performant engines. A short comparison is made between the engines available when ordering those aircraft.

In the next part the process is studied to design a high bypass turbofan engine ,starting by defining the design parameters.

Step by step the design of the engine is explained starting from the ideal model to the real model in on design conditions. This will be done by using the PARA computer software .

Finally ,real engine performance in off design conditions will be analyzed by using the PERF computer software

Introduction

In this work we will study the basic parameters and the methods used for the conception and design of high performance power plants, in particular for high bypass ratio turbofan engines.

The production process of a new engine ,from conception to finished operational reliable product , is a process which involves a lot of research and development and a small army of engineers. To increase engine performance the industry is very dependent of innovative technologies , for example new alloys which are more heat resistant or cooling so that the turbine inlet temperature can be increased. This work only discusses a part of this process, as the complete process is a big team effort.

In the first chapter of this thesis we take a look at the power plants that are used to power one of the most modern aircraft, the AIRBUS A-380. Attention will be paid to the construction of the two different engines which are available on the aircraft. Some of the new technologies used in these engines also will be discussed.

The second chapter will regard aircraft performance and determines some of the basis requirements that the engine must be able fulfill. The specifications of the aircraft will be determined here, like aircraft dimension and weight ,and calculation of minimum take off and landing speed. The aerodynamic properties of the aircraft are also calculated in this part

Chapter three specifies the station numbering of the turbofan engine that will be adopted throughout this work in the formulas.

In chapter four , parametric cycle analysis is done for the ideal turbofan engine using the PARA program to analyze the data. This cycle analysis considers on design conditions.

Chapter five regards the individual components of the engine. As the components of the engine determine the overall efficiency of the engine, attention must be paid to the design of each component of the engine. an overview is given of figure of merits per component in function of the technological evolution.

In the sixth chapter , parametric cycle analysis is done for the real turbofan engine using the PARA program to analyze the data. The results obtained from this analysis will be compared with those obtained from the ideal engine analysis.

In the last chapter, engine performance in off design conditions is studied. This data analysis will be done using the PERFO software

Software used for calculations

- PARA program. This software allows us to make parametric cycle analysis of the real and ideal engine in design conditions .The variation of parameters like Mach numbers and compression ratio can be calculated and the change in engine performance evaluated.
- PERF program. This software allows us to make calculations of real engine performance in off design conditions .In this way the influence of different flight conditions can be studied and evaluated.

List of symbols

A - area; constant

a - speed of sound; constant

b - burner

C_D - coefficient of drag

C_{D0} - zero lift drag coefficient

C_L - coefficient of lift

C_L* - coefficient of lift corresponding to max (C_L / C_D) ratio

c_p - specific heat at constant pressure

c - compressor

D - drag

d - diffuser

e - energy per unit mass; polytrophic efficiency

F - uninstalled thrust

f - fuel/air ratio; fan

g_c - Newton's constant

g_0 - acceleration of gravity at sea level

h - enthalpy per unit mass; height

h_{PR} - low heating value of fuel

HPC - high pressure compressor

HPT - high pressure turbine

IPC - intermediate pressure compressor

IPT - intermediate pressure turbine

K_1 - constant in lift-drag polar equation

K_2 - constant in lift-drag polar equation

L - lift

LPC - low pressure compressor

LPT - low pressure turbine

M - Mach number

$\dot{m}$ - mass flow rate

n - nozzle, load factor

P - pressure

Po - ambient pressure

P_r - reduced pressure

P_t - total pressure

q - dynamic pressure

R - gas constant; additional drag

S - uninstalled thrust specific fuel consumption

S_w - wing platform area

s - spatium

T - installed thrust; temperature

Tt - total temperature

TSCF - installed thrust specific fuel consumption

t - time, turbine

V - velocity

W - weight

Greek

α - angle of incidence ;by pass ratio

γ - ratio of specific heats

δ - dimensionless static pressure (=P/P_{ref})

δ_0 - dimensionless toal pressure

η_i - isentropic efficiency ,isentropic evolution indicated by „i”

η_0 - overall efficiency

η_p - propulsive efficiency

η_t - thermal efficiency

ρ - density

σ - dimensionless static density (=p/p_{ref})

τ - total temperature ratio

τ_i - total temperature ratio, isentropic evolution

θ - dimensionless temperature ratio (=T/T_{ref})

ϕ_{inlet} - dimensionless inlet loss coefficient

ϕ_{nozzle} - nozzle loss coeficient

Review of the turbofan engines used to power the A380 airplane

In this part we will take a closer look at the powerplants used to power the A380,the newest airplane constructed by AIRBUS using the latest state of the art technology.

Figure 1.1. Source. www.airbus.com

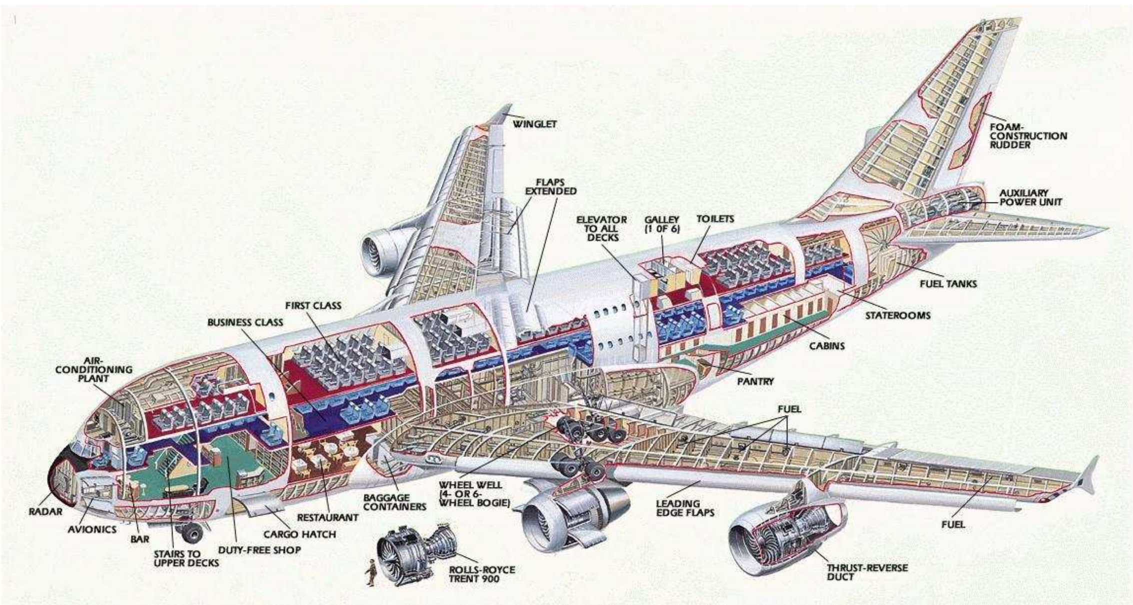


Figure 1.2. Source: www.airbus.com

The answer to this wide body jet was the conception and development of the B787 Dreamliner by Boeing

Airbus developed the A380 as the most spacious and efficient airliner ever conceived. This 525-seat aircraft will deliver an unparalleled level of comfort while retaining all the benefits of commonality with Airbus' other fly-by-wire aircraft Families.

By incorporating the latest advances in structures and materials, the A380 offers a direct operating cost per passenger that is 15 per cent lower than the competing large airliner. Reliability and maintainability will be further increased through the use of new technologies such as an enhanced onboard central maintenance system and variable frequency generators (which simplify the large aircraft's electrical generation network).

New-generation engines, combined with an advanced wing and landing gear design, will make the A380 significantly quieter than today's largest airliner – enabling the very large aircraft to meet strict local regulations at airports around the world.

A goal of the A380 program from the start has been to offer double-digit improvements in fuel burn and operating costs when compared with today's largest commercial aircraft.

The A380 has the potential to increase an operator's return by as much as 35%. Its increased capacity and longer range provide airlines with significantly more seat-miles on every flight. And because Airbus has maximized the boarding and deplaning process by cutting out choke points, the significantly shorter turn-round time for airport processing of an A380 allows schedules to be kept tight and extra flights flown.

Thanks to the incorporation of the latest advances in structures, materials, aerodynamics, systems and engine design, the A380 will provide a direct operating cost per seat which is 15-20 per cent lower than the 747-400.

Despite its ability to carry 35 per cent more passengers than its competitor, the A380 burns 12 per cent less fuel per seat – reducing operating costs and minimizing its effects on the environment at the same time through fewer emissions. The A380 burns fuel per passenger at a rate comparable to that of an economical family car.

AIRCRAFT DIMENSIONS

	metric	imperial
Overall length	73 m.	239 ft. 3 in.
Height	24.1 m.	79 ft. 7 in.
Fuselage diameter	7.14 m.	23 ft. 5 in.
Maximum cabin width	Main deck: 6.58 m. / Upper deck: 5.92 m.	Main deck: 21 ft. 7 in. / Upper deck: 19 ft. 5 in.
Cabin length	49.90 m.	163 ft. 8 in.
Wingspan (geometric)	79.8 m.	261 ft. 8 in.
Wing area (reference)	845 m2	9,100 ft2
Wing sweep (25% chord)	33.5 degrees	33.5 degrees
Wheelbase	30.4 m.	99 ft. 8 in.
Wheel track	14.3 m.	46 ft. 11 in.

BASIC OPERATING DATA

	metric	imperial
Engines	Trent 900 or GP 7000	Trent 900 or GP 7000
Engine thrust range	311 kN	70,000 lb. slst
Typical passenger seating	525	525
Range (w/max. passengers)	15,200 km.	8,200 nm.
Max. operating Mach number (Mmo)	0.89 Mo.	0.89 Mo.
Bulk hold volume - Standard/option	18.4 m3	650 ft3

DESIGN WEIGHTS

	metric	imperial
Maximum ramp weight	562 tonnes	1,239 lbs. x 1000
Maximum takeoff weight	560 tonnes	1,235 lbs. x 1000
Maximum landing weight	386 tonnes	851 lbs. x 1000
Maximum zero fuel weight	361 tonnes	796 lbs. x 1000
Maximum fuel capacity	310,000 Litres	81,890 US gal.
Typical operating weight empty	276.8 tonnes	608.4 lbs. x 1000
Typical volumetric payload	66.4 tonnes	145.5 lbs. x 1000

Table 1.1 Basic A380 data. Source: www.airbus.com

Airbus' 21st century flagship introduces a new era of airline transportation, carrying 525 passengers aboard the most advanced, spacious and efficient aircraft ever conceived. The world's first true double-deck jetliner, the A380 will have a maximum takeoff weight of 590 metric tons (1,300,000 lb.) and a range of 15,200 km/8,200 nm.

The A380 also benefits from the latest innovations in aerodynamics, reducing drag to the minimum and improving fuel efficiency further. Moreover, the A380 is fitted with new state of the art high by-pass engines that contribute to the overall reduction in fuel burn. The combination of 21st Century aerodynamic standards and advanced digital design tools enables the A380 to offer unsurpassable performance standards. A highly efficient wing design allows the A380 to take-off and land in less distance than today's largest aircraft. As a result the A380 uses existing runways while carrying 40 per cent more passengers per flight.

Pratt & Whitney GP 7000

Two of the most respected engine manufacturers in the industry, Pratt & Whitney and General Electric, bring the new GP7000 engine family to A380 customers. GP7000 is derived from some of the most successful wide body engine programs in aviation history - the GE90 and PW4000 families. These engines demonstrated industry leading ETOPS reliability from service entry and forged a record of over 250 million hours of superior performance. Building on the GE90 core and the PW4000 low spool heritage, the GP7000 is a refined derivative with a responsible infusion of new technologies. On the Airbus 380, the GP7000 will exceed the high standards of in-service reliability and performance expected by tomorrow's operators.

- Best of GE Aircraft Engines and Pratt & Whitney technologies.
- Two-spool simplicity for reliability and maintainability.
- Best payload capability, performance and performance retention.
- Quietest.
- Lowest emissions.
- Twice the capabilities of any one engine manufacturer ensure technical commitments will be met on schedule.

MTU Aero Engines is Germany’s largest aero engine manufacturer. MTU is a partner of General Electric and Pratt & Whitney, who in 1996 established the Engine Alliance for this GP7000 engine program. MTU is responsible for the development and manufacture of the turbine center frame and the low-pressure turbine as well as for the production of high-pressure turbine parts.

Engine models:

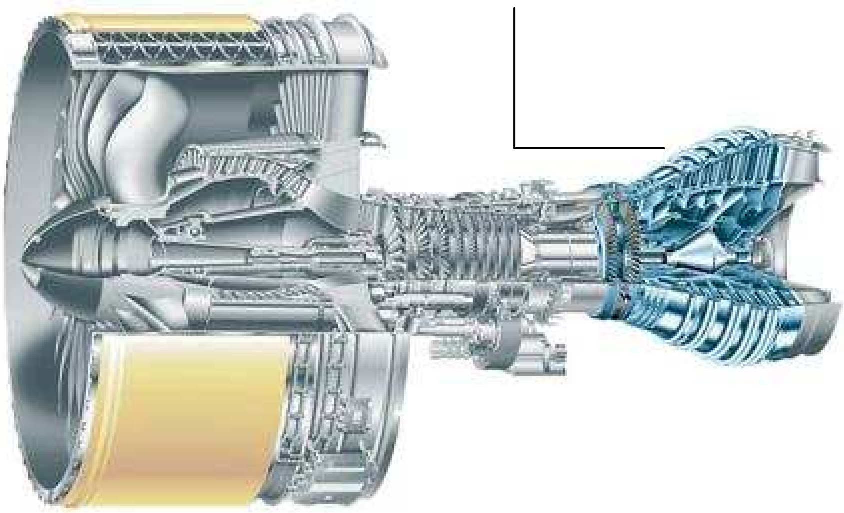
- GP7270
- GP7277

Planes powered by GP7200:

- Airbus A380-800
- Airbus A380-800F

Technical description of the GP 7000

MTU share (22.5 %):
5.0 % HP Turbine
5.0 % Turbine Center Frame
12.5 % LP Turbine



Program shares:
30.0 % P&W
30.0 % GE
22.5 % MTU
10.0 % Snecma
7.5 % Techspace Aero

Figure 1.3. Source: www.enginealliance.com

GP7200 Engine Features

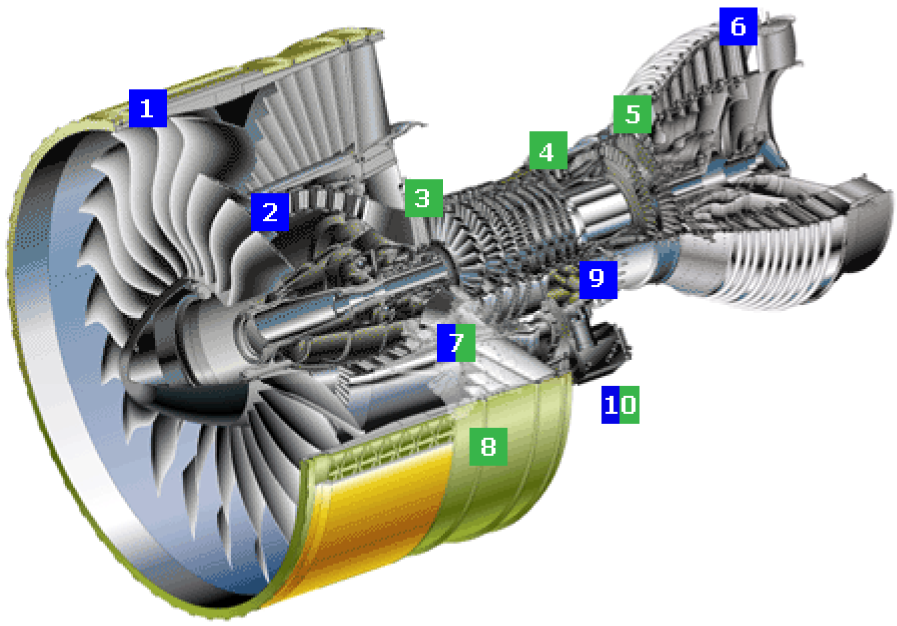


Figure 1.4. Source: www.enginealliance.com

Based on GE Engine Platform

Based on Pratt & Whitney Engine Platform

1. Fan Module/Blade Containment
   Built on PW4084 foundation

- FOD and erosion resistant, swept wide chord hollow titanium fan blade with 3D aerodynamic design
- Lightweight and durable Kevlar-aluminum containment system
- Improved fan/exit guide vane spacing for optimum noise
- Indwidual fan blade reparability and on- wing replacement

2. Low Pressure Compressor
   Built on PW4084 foundation

- 3D aerodynamic airfoil design for reduced fuel bum and increased EGT Margin and life
- Low dirt and debris particle ingestion design for improved durability and long on-wing life

3. High Pressure Compressor
   Built on GE90 foundation

- 3D aerodynamic airfoil design for improved efficiency, FOD resistance and superior stall-free operation
- Thermally matched casing and rotors for improved performance retention through blade rub reduction
- Wide chord, forward swept stage 1 integrally bladed disk for reduced maintenance cost and on-wing repair

4. Low Emission Combuster
   Derived from GE CF6 and CFM Technologies

- Low emission technology with singular combustor (SAC) design simplicity
- Meets all current and future CAEP4 emission requirements with significant margin

- Lowest Nox emissions forA380 aircraft

5. High Pressure Turbine
   Built on GE90 foundation

- 3D aerodynamic design, split blade cooling for optimum performance and durability
- Thermally matched casing and rotors for performance retention through tighter blade-tip clearances
- Thermal barrier coated Rene N5 single crystal airfoils for improved durability and performance
- Boltless rotor architecture reduces part count and increases disk life, reducing overall maintenance costs

6. Low Pressure Turbine
   Built on PW4000 foundation

- 3D aerodynamic airfoil design for reduced fuel bum and increased EG T Margin and life
- Axial gap optimization / low stage acoustic cut-off design for noise reduction
- Hollow airflows for increased turbine efficiency and reduced weight

7. Bearing & Lube System
   Built on GE90 and PW4000 technologies

- Simple two spool engine architecture results in better overall reliability and reduced maintenance costs
- Short$_y$ one piece tower shaft for improved accessory drive train reliability
- Carbon seals reduce oil consumption and fuel bum
- Low pressure, unregulated lube system allows easier system servicing and reduced oil volume

8. Digital Engine Control
   Based on GE90 and CFM technologies

- FADEC III Engine Control System builds upon experience of previous Iwo generations
- Advanced condition motoring system integration with A380 aircraft sets new standards in performance trending and remote diagnostic compatibility
- Advanced processing capability and redundant critical monitoring sensors results
  in significant reduction potential in aircraft delays and cancellations

9. Accessory Gearbox
   Based on PW4000 proven design

- Core mounted architecture maintains shorter, less complex tower shaft configuration for improved maintainability
- Internally cored lubrication passage minimizes external plumbing and oil leakage potential
- Proven gear, bearing and steel design for long life and durability

10. Transportability / Propulsor Option
    Based on GE90 and PW4000 designs and maintainability

- Engine split-ship capability to facilitate module transport by most wide-body combi and freighter aircraft
- Propulsor configuration option available to support out-station engine change-out and minimize spare engine investments

Emissions

The environmentally friendly GP7200 meets all ICAO gaseous emissions standards with substantial margin.

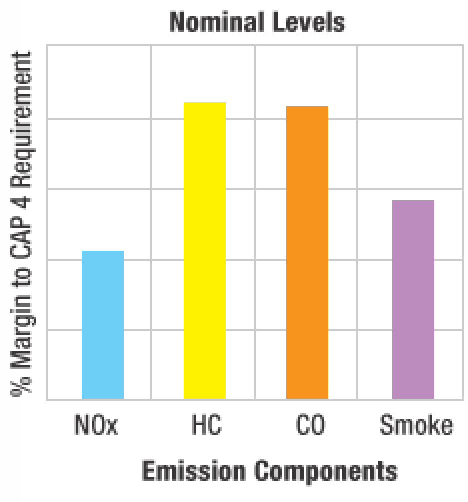


Figure 1.5. Source: www.enginealliance.com

Noise

The GP7200 is a good neighbor, allowing the A380 to meet current Stage 3 and proposed Stage 4 noise level standards with margin.

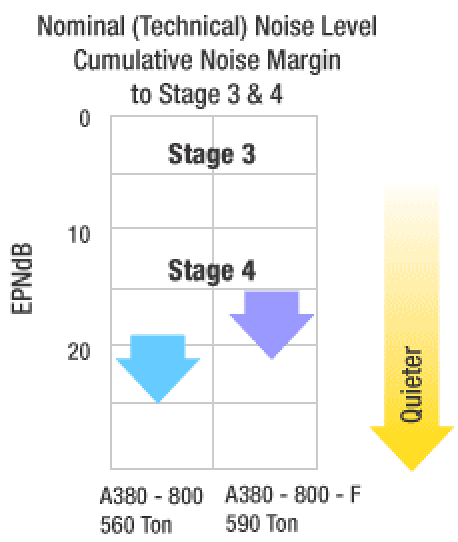


Figure 1.6. Source: www.enginealliance.com

Fan/LPC

- Swept fan blades
- Frangible #1B Support
- New booster aero

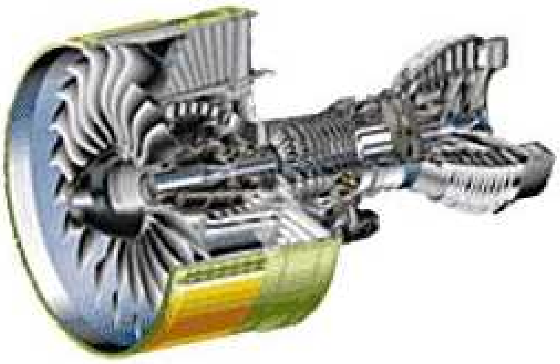


Figure 1.7. Source: www.enginealliance.com

Core

- 72% flow scale of GE90-115B
- Low Emissions Combustor
- FADEC III Control

Figure 1.8. Source: www.enginealliance.com

LPT

- High lift airfoils
- Airfoil clocking

Figure 1.9. Source: www.enginealliance.com

Rolls-Royce Trent 900

Rolls-Royce Trent are a family of high bypass turbofan engines manufactured by Rolls-Royce. All are developments of the RB211 with thrust ratings of between 53,000 and 95,000 lb_f (236 to 423 kN). Versions of the Trent are in service on the Airbus A330, A340, A380 and Boeing 777, and variants are in development for the forthcoming 787 and A350 XWB. The Trent has also been adapted for marine and industrial applications.

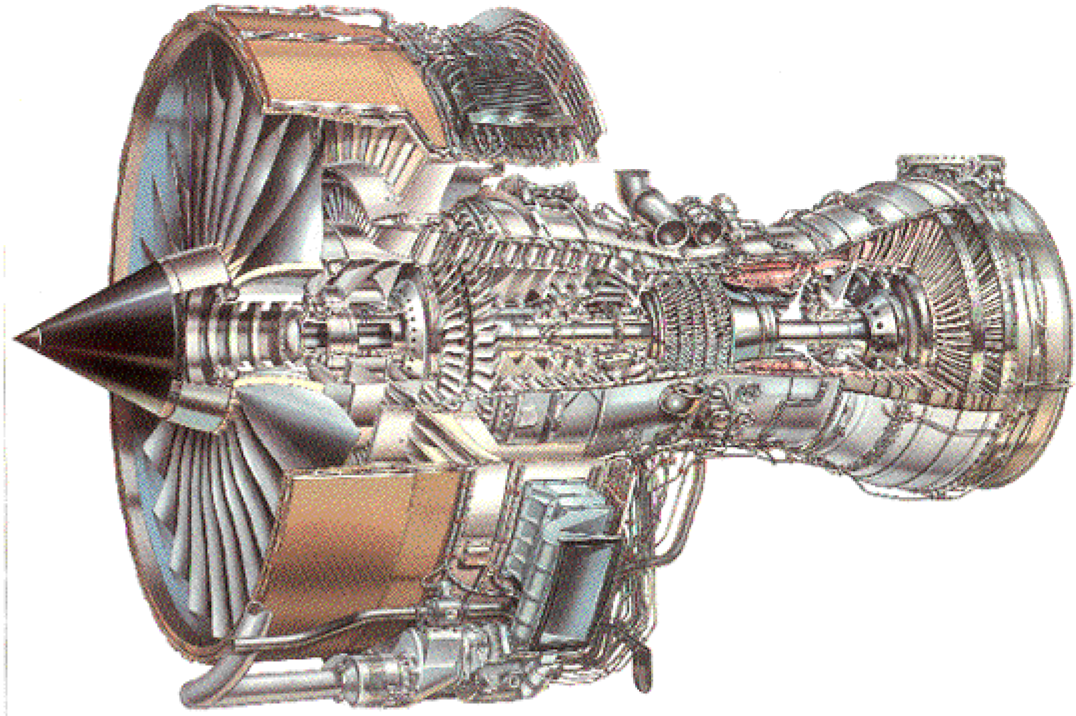


Figure 1.10. Source : /www.aircraftenginedesign.com

The Trent has achieved significant commercial success, being the launch engine for all three 787 variants, the A380 and A350. Its overall share of the markets in which it competes is around 40%. Sales of the Trent family of engines have made Rolls-Royce the second biggest supplier of large civil turbofans after General Electric

Rolls-Royce Trent 900 engines powered the first scheduled service of a Singapore Airlines' Airbus A380 from Changi Airport, Singapore to London Heathrow on 18 March 2008.

Although the Trent 900 is the largest and one of the most powerful engines produced by Rolls-Royce, it is the quietest and cleanest engine for the A380. Leading engine development programme on the A380 and launch engine for the aircraft (entry into service with Singapore Airlines, Spring 2006)

First run, on schedule, March 2003

Engine certification, on schedule, October 2004

Successfully completed 60-hour flight test programme on A340-300 flying test bed, August 2004, on schedule

Will power maiden flight of A380, Spring 2005

Cleared for 80,000lb thrust at certification, allowing margin for growth. Thrust required at entry into service is 70,000lb

Lightest, cleanest and quietest engine on A380. Lowest emissions of any large turbofan engine, measured per pound of thrust

New technology features include "swept" titanium fan blades (lower noise, improved aerodynamics) and counter-rotating HP system ("straightens" air flow, improves efficiency)

Its 116 inch diameter fan makes the Trent 900 physically the largest engine ever built by Rolls-Royce

Trent 900 risk and revenue sharing partners:

Industria de Turbo Propulsores (ITP), Hamilton Sundstrand, Avio, Marubeni, Volvo Aero, Goodrich and Honeywell.

Samsung Techwin, Kawasaki Heavy Industries (KHI) and Ishikawajima-Harima Heavy Industries (IHI) are programme associates.

Family background:

Trent 900 incorporates the successful and unique three-shaft design established by the RB211 series and continued by earlier Trent variants

Fourth version of the Trent series to enter service - all market leaders

Trent 700 (39%) Trent 800 (43%) Trent 500 (100%)

Firm and option orders placed for over 1,900 Trents by nearly 50 customers

Trent 900 will draw on the experience of 20 million total Trent flying hours by service entry

And, not a lot of people know that....

At take-off, the A380's four Trent 900s will deliver thrust equivalent to the power of more than 3,500 family cars

The engine's hollow, titanium fan blades are almost 10 feet across and suck in over 1.25 tons of air every second. By the time the air leaves the nozzle at the back of the engine it has been accelerated to a speed of nearly 1,000 miles per hour (1,600 kph)

Temperatures in the engine core are half those on the surface of the sun

The blades in the engine's high pressure system rotate at 12,500 rpm, with tip speeds reaching 1,200 miles per hour (2,000 kph)

At take-off, each of the 70 high pressure turbine blades in a Trent 900 produces over 800hp....more power than a Formula One racing car

The 116 inch fan operates at nearly 3,000rpm with tip speeds 1.5 times the speed of sound

A Trent 900 has around 20,000 individual components

The Trent 900 is the fourth member of the Trent family and includes the latest proven technology. For example:

Figure 1.11. Source: www.rolls-royce.com

- The engine's 24 fan blades are to a new swept design that reduces the effect of shock waves, as the tip of the fan rotates supersonically, making it lighter, quieter and more efficient.

- The fan containment system is also the first to be manufactured from Titanium and does not need the additional Kevlar wrap, making it a lighter and smaller system
- At the core of the engine, the high-pressure shaft rotates in the opposite direction to the other two shafts, meaning the engine can be made lighter and more fuel-efficient.

Designed for Service: Low cost of ownership

Engine Health Monitoring for no operational disruptions

Only engine on the A380 that can be transported whole in a Boeing 747F

Designed for ease of maintainability with Fancase mounted units

Figure 1.12. Source: www.rolls-royce.com

Excellent reliability from day one with enhanced service readiness

- Core scaled from the Trent 500
- Benefits of Trent family experience

Excellent Fuel efficiency

- High bypass ratio

Best for the environment

- Quietest engine on the A380
- Lowest emissions on the A380 - world's lowest turbofan emission

Technical description of the TRENT 900

Fan system

Figure 1.13. Source: www.rolls-royce.com

Lightest fan system in the industry

Proven swept fan design (116" diameter) - low noise, low weight and high performance

Proven light weight titanium fan case - robust and transportable whole in Boeing 747F

Compressor

Figure 1.14. Source: www.rolls-royce.com

Intermediate and high pressure compressors

- Proven Trent technology world-class performance
- Lowest risk
- Bladed discs throughout
- lowest maintenance cost

- Next generation 3D aerodynamics
- More efficient airflow throughout engine
- Better fuel consumption
- Less parts and lower weight
- Rolls-Royce unique 3-shaft design enables compressors to operate at optimum speeds

Combustor

Figure 1.15. Source: www.rolls-royce.com

Combustor

- Proven Trent technology
- Tiled combustor for lowest maintenance costs
- Engine tests have confirmed the Trent 900 as having the world's lowest emissions
- Meets all proposed legislations with margin Assured asset value

Turbines

Figure 1.16. Source: www.rolls-royce.com

- Proven Trent technology
- Low risk
- Next generation 3D aerodynamics with end wall profiling
- optimum performance and lowest fuel burn
- Proven shrouded turbine blades
- Best performance retention

Contra-rotation

Figure 1.17. Source: www.rolls-royce.com

High pressure contra rotation

- Contra-rotation of the high pressure system significantly improves the performance and efficiency of the intermediate pressure turbine
- Improved efficiency
- Fewer parts
- Lighter parts

Health monitoring

Figure 1.18. Source: www.rolls-royce.com

- QUICK' the predictive maintenance technology that enables real-time monitoring using today's sensors (oil, vibration, pressures and temperature).
- Operational flexibility and lower the cost of ownership * Technology already demonstrated successfully on the Trent 500 A340-500/600 flight test and a test bed uersion in operation at major overhaul bases.
- The on board intelligent software analyses the data in flight and combined with Data Systems and Solutions provides advanced Engine Health Monitoring capability.

Predictability

- Each new generation Trent has demonstrated the merits of our derivative approach, setting standards of world-class reliability with each new engine mark
- Over 250 million hours of Trent operational experience are incorporated into the Trent 900 design
- Engine certification on time in October 2004 in-line with the Rolls-Royce proven track record of delivering Trent engine programmes on time and to specification
- The Trent 900 engine development programme has delivered excellent results across the range, testament to the evolution of the proven Trent design
- World-leading Rolls-Royce operations technology ensures that each Trent 900 engine is continually monitored in service, and supported 24/7

… ensuring that our airline customers and their passengers receive the highest levels of operational excellence.

Comparison of engine characteristics

Table 1.2. Comparison of engine characteristics

	TRENT 900	**GP7000**
Programme partners	FiatAvio, Goodrich, Hamilton Sundstrand, Honeywell, Volvo Aero, Marubeni,	General Electric, Pratt & Whitney
Fan diameter (in)	116	116
Bypass ratio	8.15 – 8.02	8.7
OPR	38.5 – 41.1	43.9
Mass flow	2.603-2.700	3,000
Cruise SFC	0.518	0.518
Basic engine weight	14,155	_
Lenght	179	187.1
Stages	Fan, 8IPC, 6 HPC, Annular 20 burners, 1HPT, 1IPT, 5LPT	Fan, 5LPC, 9HPC, 2HPT, 6LPT

Source: Author

Aircaft performance

Before we can start with the analysis of the turbofan engine requirements, we must have an idea of the design parameters for the engine.

In order to establish these parameters ,attention must be paid to aircraft performance , because the aircraft dimensions, aerodynamic shape , configuration and maximum weight will determine the basic requirements in designing or choosing the turbofan engine for this aircraft.

Once these aircraft properties and required performance for this aircraft are chosen and set, proper turbofan engine design can begin

Characteristics of the commercial aircraft equipped with turbofan engines.

Maximum gross take off weight	W_{T0}=1,645,760 N
Empty weight	822,880 N
Maximum landing weight	1,356,640 N
Maximum payload (253 passengers plus 190,000N cargo)	420,780 N

Maximum fuel capacity	716706 N
Wing area	S_w=282,5 m^2
Maximum lift coefficient	C_{Lmax}=2,0

This aircraft will be propelled by two high-bypass-ratio turbofan engines.

The engines must be able to perform as follows:

- In the event of an engine failure , the aircraft must still be able to maintain a 2,4% single engine climb gradient. (P_S=1,5 m/s)
- The single engine aircraft flies at 5000 m altitude at a speed of 0,45 Mach. (P_S=1,5 m/s)

<u>Relationships for performance of an aircraft</u>

Here a closer look will be taken at the forces that affect and define aircraft performance. These forces and their magnitude and equilibrium will determine the power required by the turbofan engines.

<u>Lift and drag</u>

The classical expression for lift will be used:

$$L = nW = C_L q S_w \tag{2.1}$$

L Lift

n load factor (n=1 for a horizontal level steady flight)

W aircraft weight

C_L lift coefficient

S_w wing area

q dynamic pressure

The classical expression for drag is the following:

$$D = C_D q S_w \quad (2.2)$$

C_D drag coefficient

S_w wing area

q dynamic pressure

The drag coefficient curve can be a approximated by a second order equation from C

$$C_D = K_1 C_L^2 + K_2 C_L + C_{D0} \quad (2.3)$$

C_{D0} zero lift drag coeficient

The coefficients K_1,K_2 and C_{D0} are function of flight mach number and wing configuration.

The K_1 and K_2 term s account for the drag due to lift.

The drag coefficients for the commercial aircraft are given in table 2.1

Table 2.1 drag coefficients K_1,K_2 and C_{D0}

TABLE 2.1 DRAG COEFFICIENTS K_1,K_2 AND C_{D0}

Mo	Ki	K_2	CDO
0	0,056	-0,004	0,014
0,4	0,056	-0,004	0,014
0,75	0,056	-0,008	0,014
0,83	0,056	-0,008	0,015

Source : Elements of gas turbine propulsion ,Jack D. Mattingly, McGraw-Hill 1996

The data of table 2.1 which shows the variation of C_{D0} and K_2 with mach number is plotted in figure 2.1

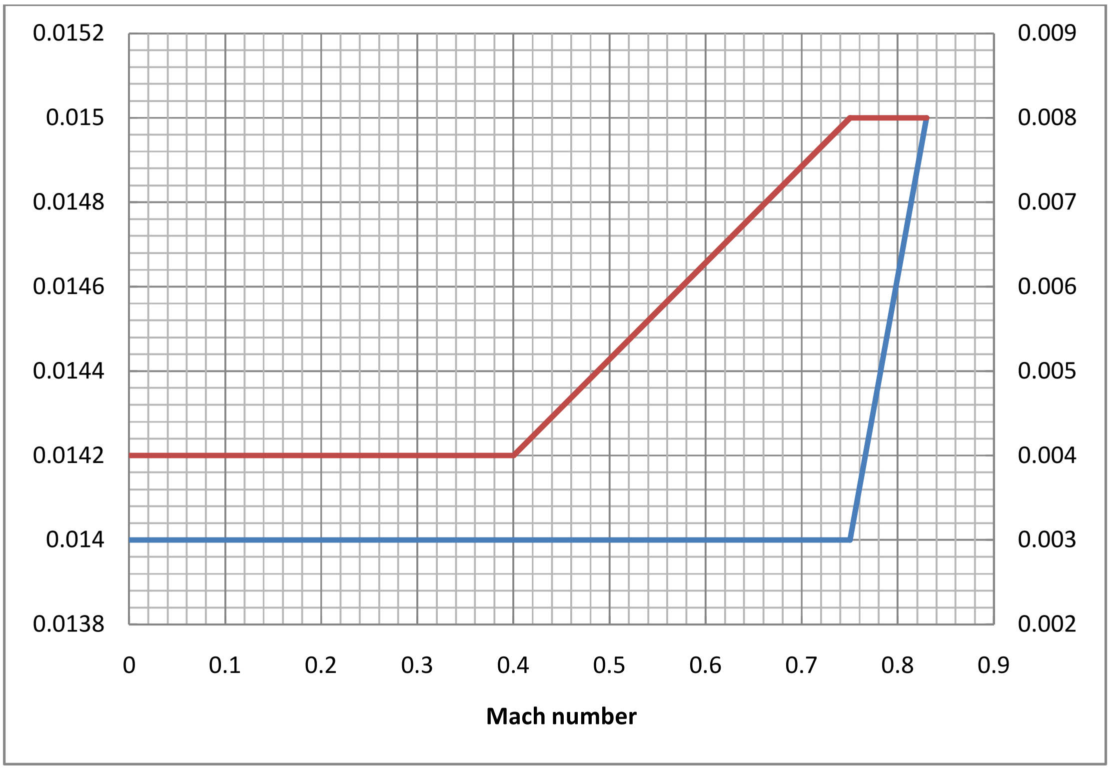


Figure 2.1. K_1, K_2 and C_{D0} in function of mach, Source: Author.

By using the information of table 2.1, in combination with equations (2.1), (2.2), (2.3), we are now able to express the drag in function of mach number.

The drag versus mach number is plotted in figure 2.2

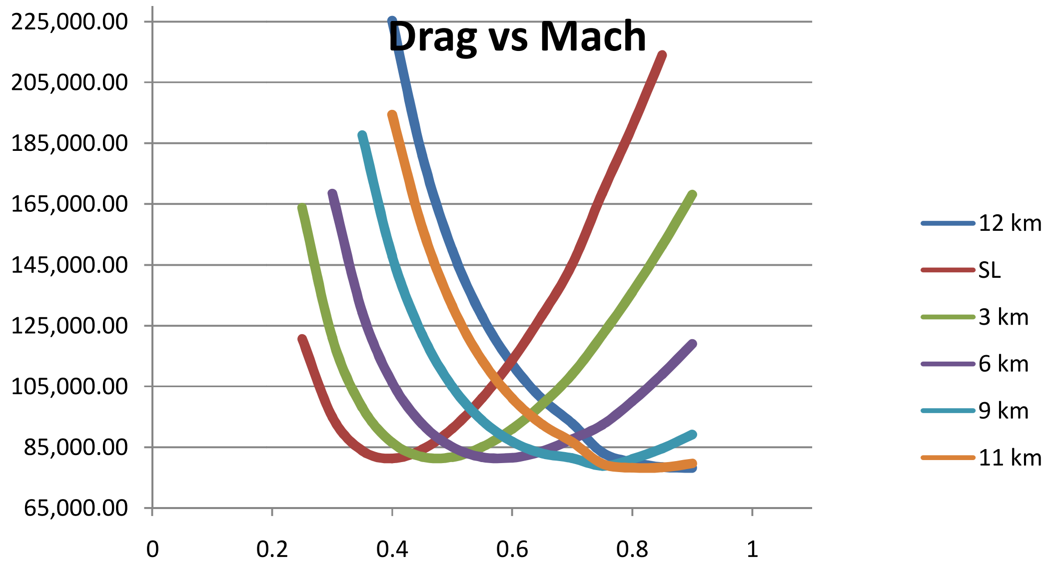


Figure 2.2. Drag versus mach number. Source: Author

Lift-drag polar

This polar consists of a plot of the lift coefficient C_L versus the drag coefficient $C_{D.}$

The polar will be plotted for the commercial passenger plane at 95% of the maximum gross takeoff weight .

Using these drag data and previous equations will determine the variation in drag with subsonic mach number and altitude for level flight (n=1).

W= 1563472 N

n = 1

dynamic pressure

$$q = \frac{\gamma}{2}\delta P_{ref} M_0^2 \tag{2.4}$$

γ =1,4 gas constant for air (ratio of specific heats C_p /C_v)

δ dimensionless pressure (=$P_{alttude}$/$P_{referece}$)

P_{ref} pressure at sea level in standard atmosphere (P_{ref} =101325 P_a)

In figure 2.3 the lift-drag polar is plotted for M= 0.75 and M=0.83

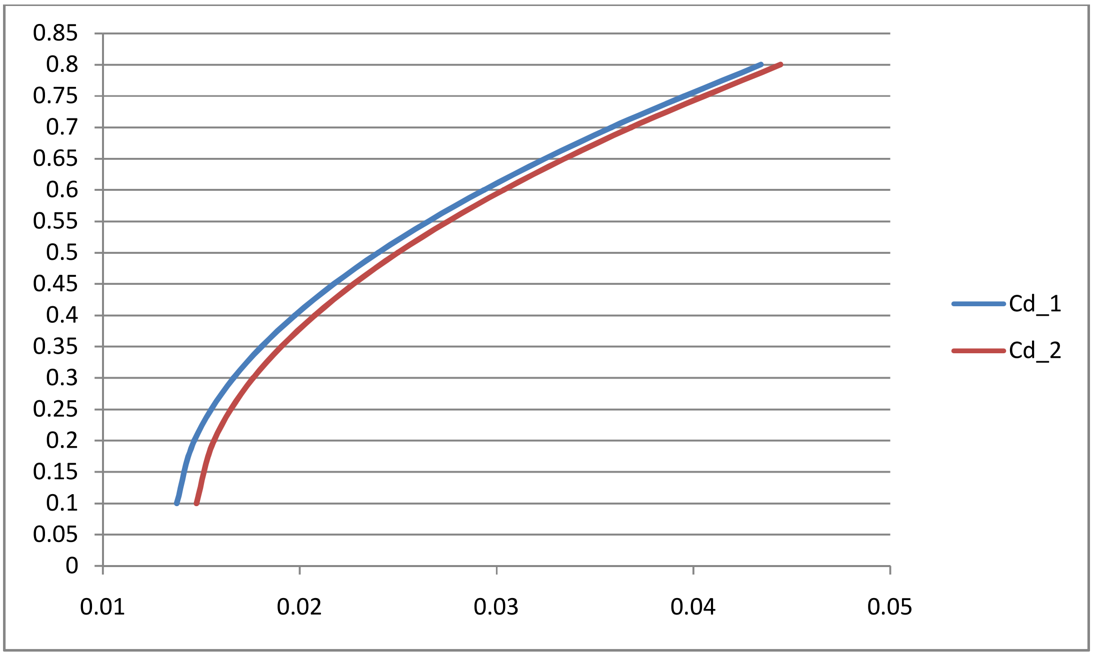


Figure 2.3. C_L, C_d polar. Source: Author.

The obtained lift-drag polar is important for our commercial passenger aircraft, because these speeds represent the cruising range of our aircraft .

Determination of Stall, Take-off speed and Landing speed

Stall is a sudden reduction in the lift forces generated by an airfoil. This occurs when the critical angle of attack of the airfoil is exceeded.

This means stall will occur during level flight (lift = drag)occurs when one tries to obtain a lift coefficient greater than the wing's C_{Lmax}.

The stall speed is defined as the level flight speed that corresponds to the wing's maximum lift coefficient.

The Stall speed is defined by following equation :

$$V_{stall} = \sqrt{\frac{2g_c}{\rho C_{Lmax}}\frac{W}{S_W}} \tag{2.5}$$

The stall condition means that control of the vehicle is lost for a certain amount if time.

As this condition is dangerous for certain flight conditions, but especially during takeoff and landing where there is no time for the aircraft to recover from this condition, safety margins will be set to take-off and landing speeds to avoid this situation. this margins only apply for a calm atmosphere, for a turbulent atmosphere it is advised increase these speeds for example by 10%.

For safety reasons the take off speed V_{T0} will be typically 20% higher than the stall speed, and the landing speed at touchdown V_{TD} will 15% greater.

$$V_{TO} = 1{,}2V_{stall} \tag{2.6}$$

$$V_{TD} = 1{,}15V_{stall} \tag{2.7}$$

Calculation of the stall speed and touchdown speed for the commercial passenger aircraft.

Stall speed at maximum gross take-off weight

$$V_{stall} = \sqrt{\frac{2\times 1}{1{,}225\times 2{,}0}\frac{1{,}645{,}760}{282{,}5}} = 69{,}0\,\frac{m}{s}$$

Take-off speed:

$$V_{TO} = 1{,}20V_{stall} = 82{,}8\frac{m}{s} = 298{,}1km/h$$

Touchdown speed:

Stall speed at maximum landing weight:

$$V_{stall} = \sqrt{\frac{2 \times 1}{1{,}225 \times 2{,}0} \frac{1{,}356{,}640}{282{,}5}} = 62{,}2\frac{m}{s} = 223{,}9km/h$$

$$V_{TD} = 1{,}15V_{stall} = 72m/s$$

<u>Fuel Consumption:</u>

The rate of change of the aircraft weight dW/dt is due do the fuel consumed by the engines during flight.

The mass rate of fuel consumed is equal to the product of the installed thrust T and the installed thrust specific fuel consumption.

We can write:

$$\frac{dW}{W} = - \ \dot{w}_f = - \ \dot{m}_f \frac{g_0}{g_c} = -T(TSFC)(\frac{g_0}{g_c}) \qquad (2.8)$$

where:

g_0	constant acceleration of gravity
g_c	Newtonian constant (in SI units g_c equals 1)
T	installed thrust

TSFC	installed thrust specific fuel consumption
$\dot{w}_f$	weight flow rate of fuel
$\dot{m}_f$	mass flow rate of fuel

The equation can be rewritten in dimensionless form as:

$$\frac{dW}{W} = -\frac{T}{W}(TSFC)\left(\frac{g_0}{g_c}\right)dt \qquad (2.9)$$

Estimation of the TSFC

In order to calculate the change in aircraft weight using equation (2.9), we need to estimate the installed engine thrust T and installed TSFC.

In many flight conditions ,the installed engine thrust T will be equal to the aircraft drag D.

The TSFC is a complex function of engine regime, temperature, pressure altitude, and speed.

The TSFC is also important to determine the endurance and range of the aircraft.

For preliminary analysis following equation can be used:

$$TSFC = (0{,}4 + 0{,}45M_0)\sqrt{\theta} \qquad \left[\frac{lb/hr}{lbf}\right] \qquad (2.10)$$

Where:

θ dimensionless temperature ratio $(=T/T_{ref})$

For un accelerated level flight thrust equals drag (T=D) and lift equals weight, applying these assumptions on equation (2.10) gives:

$$\frac{dW}{W} = -\frac{C_D}{C_L}(TSFC)\left(\frac{g_0}{g_c}\right)dt \tag{2.11}$$

Endurance

The endurance factor of an aircraft is defined by the following relation:

$$EF \equiv \frac{C_L}{C_D(TSFC)}\left(\frac{g_0}{g_c}\right) \tag{2.12}$$

By applying this relation equation (2.11) becomes:

$$\frac{dW}{W} = -\frac{dt}{EF} \tag{2.13}$$

The minimum fuel consumption for a time t will occur at the flight condition where the endurance factor is maximum.

When the endurance factor is constant or nearly constant ,we can integrate equation (2.13) from initial to final conditions we obtain following expression for the aircraft weight fraction:

$$\frac{W_f}{W_i} = exp\,(-\frac{t}{EF}) \tag{2.14}$$

Or

$$\frac{W_f}{W_i} = exp\left[-\frac{C_D}{C_L}(TSFC)t\frac{g_0}{g_c}\right] \tag{2.15}$$

Range

In flight phases where distance is important, the differential time dt is related to the differential distance ds by following law:

$$ds = Vdt \tag{2.16}$$

Where:

V aircraft velocity

Substitution of (2.16) in (2.15) gives:

$$\frac{dW}{W} = -\frac{C_D}{C_L}\frac{TSFC}{V}\frac{g_0}{g_c}ds \tag{2.17}$$

The range factor of an aircraft is defined by the following relation:

$$RF \equiv \frac{C_L}{C_D}\frac{V}{TSFC}\frac{g_c}{g_0} \tag{2.18}$$

Therefore equation (2.17) becomes:

$$\frac{dW}{W} = -\frac{ds}{RF} \tag{2.19}$$

The minimum fuel consumption for a distance s will occur at the flight condition where the range factor is maximum.

When the range factor is constant or nearly constant ,we can integrate equation (2.19) from initial to final conditions we obtain following expression for the aircraft weight fraction:

$$\frac{W_f}{W_i} = exp\,(-\frac{s}{RF}) \tag{2.20}$$

Or

$$\frac{W_f}{W_i} = \exp\left(-\frac{C_D}{C_L}\frac{(TSFC)\times s}{V}\frac{g_0}{g_c}\right) \tag{2.21}$$

This is called the Breguet range equation. For the range factor to remain constant, C_L/C_D and V/TSFC must be constant.

Above 11,000 m of altitude ambient temperature is constant and a constant velocity V wil correspond to constant Mach and constant TSFC for a fixed throttle setting .

If C_L is constant , C_L/C_D will be constant

Aircraft weight decreases during flight, thus the altitude must increase to reduce the density of the ambient air and produce the required lift (L=W) while C_L and velocity remain constant

We call this flight profile a cruise climb.

In the next figures (2.4), (2.5), (2.6) the TSFC, EF, and RF are plotted as a function of Mach number for the commercial passenger aircraft at 95% of the maximum gross take-off weight.

The best endurance Mach number (= minimum fuel consumption) occurs increases with altitude and best fuel consumption is at sea level.

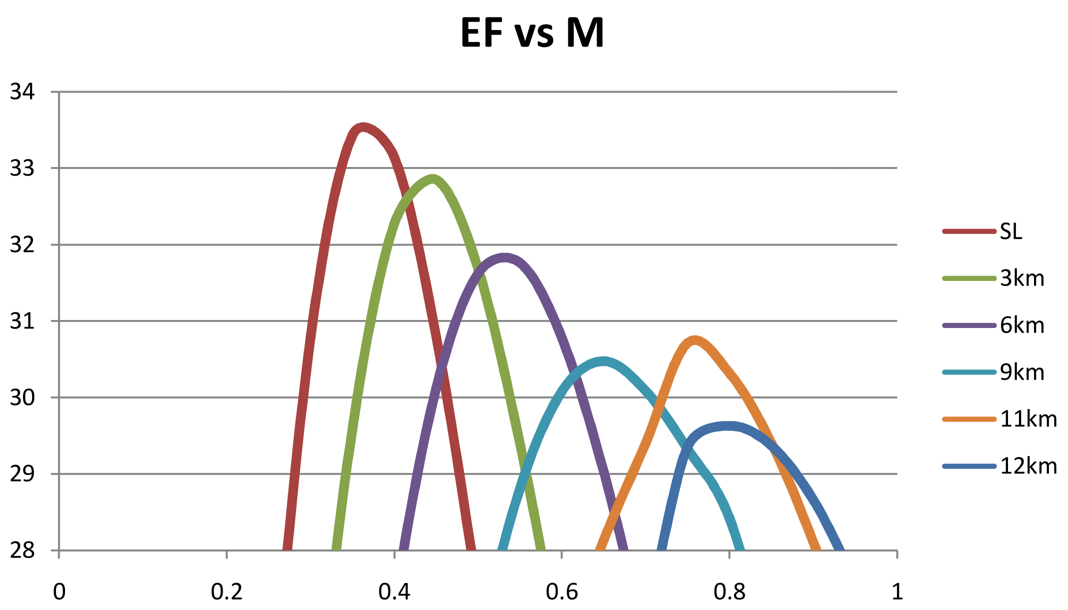


Figure 2.4. Endurance factor in hours in relation to mach number, Source: Author

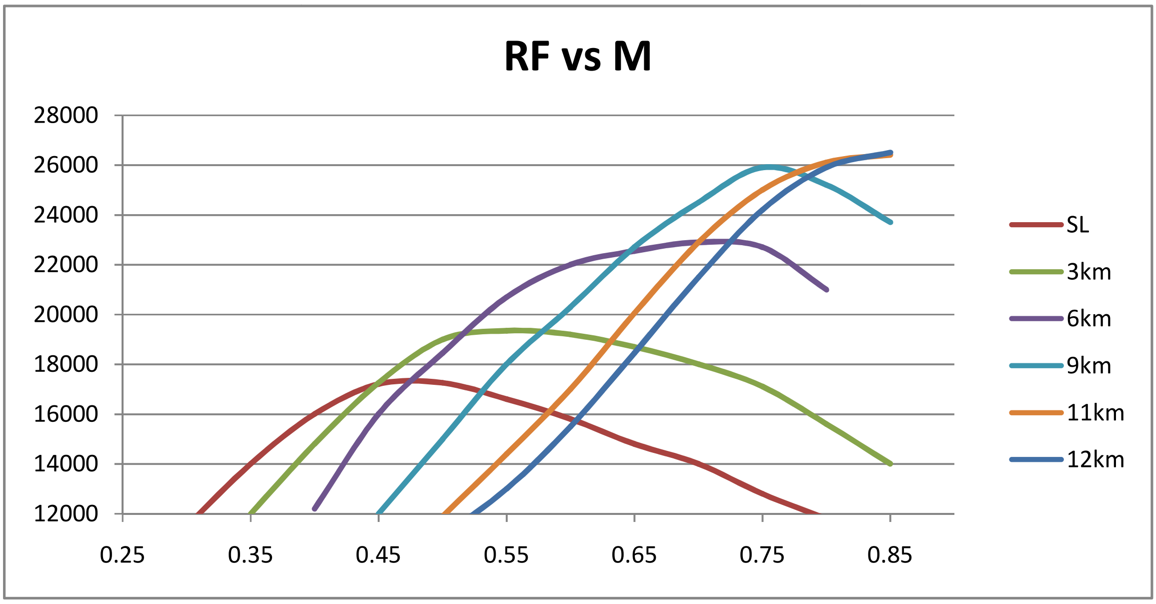


Figure 2.5 Range factor

The best cruise Mach number (minimum fuel consumption) increases with altitude and the best fuel consumption occurs at 11 km altitude and at a Mach number of about M=0.83

C_L/C_D - relationship

In case flight conditions require minimum fuel consumption , the optimum flight condition can be approximated by the one corresponding the maximum C_L/C_D

By taking the derivative of expression (2.3) ,setting it to equal 0 and solving it for C_L that gives minimum C_D/C_L we obtain maximum C_L/C_D :

$$\frac{C_D}{C_L} = K_1 C_L + K_2 + \frac{C_{D0}}{C_L} \qquad (2.22)$$

The lift coefficient that that gives maximum C_L/C_D (minimum C_D/C_L) is

$$C_L^* = \sqrt{\frac{C_{D0}}{K_1}} \qquad (2.23)$$

$$C_L^* = \sqrt{\frac{0.014}{0.056}} = 0.5$$

And maximum C_L/C_D is given by

$$(\frac{C_D}{C_L})^* = \frac{1}{2\sqrt{C_{D0}K_1}+K_2} \qquad (2.24)$$

On figure (2.7) we can see that the maximum C_L/C_D occurs at M=0,75.This is also the Mach number that corresponds to the point of minimum drag. Thus for M=0.75, C_L/C_D is maximum and EF is maximum.

The Mach number that corresponds to maximum range for a given altitude ,is usually higher than that corresponding to $(C_L/C_D)^*$.It can be explained because the velocity factor dominates the increase in TSFC.

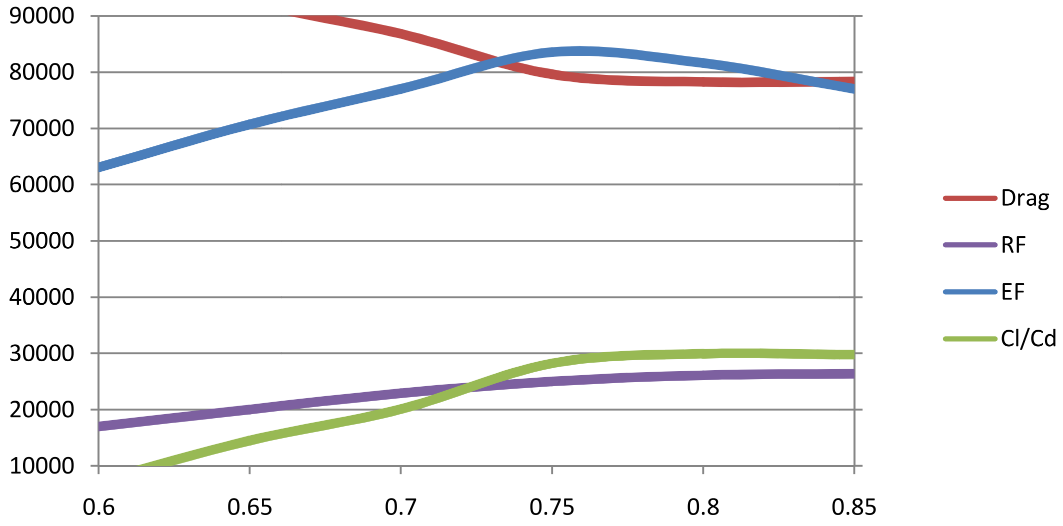
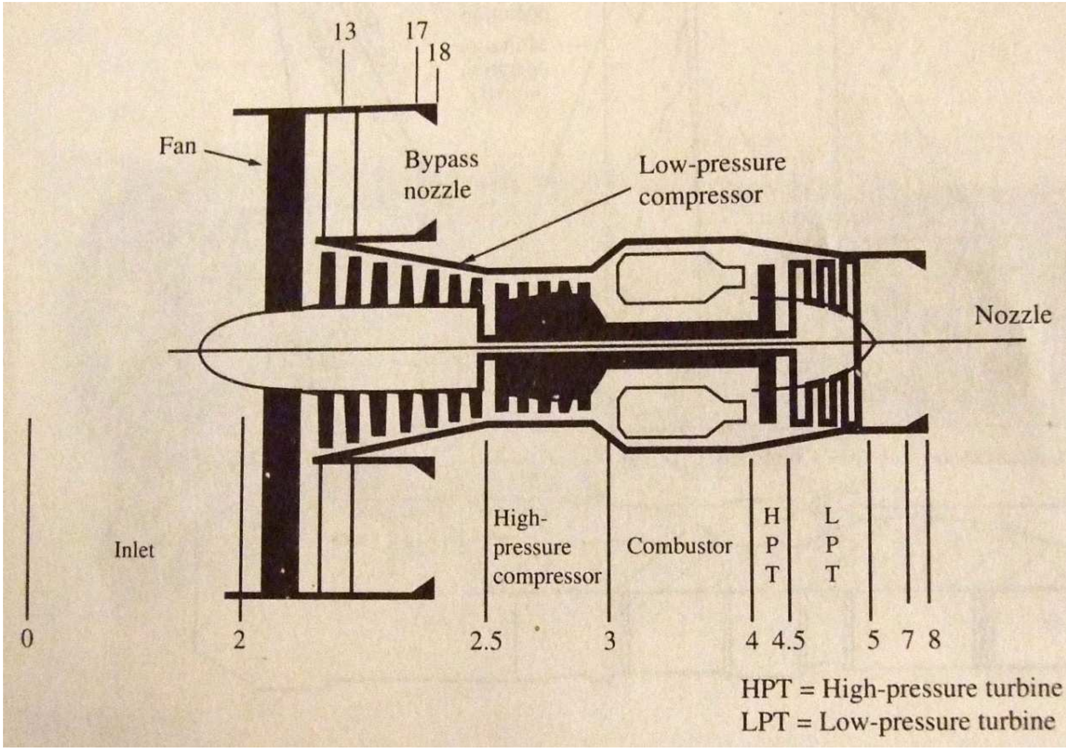


Table 2.6. Drag, Range factor, Endurance factor, Cl/Cd in function of Mach number, Source: Author

Schematic diagram of a high Bypass ratio turbofan engine.



0	undisturbed airflow
2	airflow through fan
2,5	compressed air leaving low pressure compressor (LPC) compressed air entering the high pressure compressor (HPC) \
3	compressed air leaving the high pressure compressor (HPC)/compressed air entering the combustion chamber
4	burnt gasses leaving the combustion chamber burnt gasses entering the high pressure turbine (HPT)
4,5	partially expanded gasses leaving HPT and entering low pressure turbine (LPT)
5	further expanded gasses leaving LPT and entering nozzle
7	continued expansion in nozzle

8	expansion to ambient pressure
13	compressed air leaving the fan and entering the bypass nozzle
17	continued expansion in by pass nozzle
18	expansion to ambient pressure

<u>Parametric cycle analysis of ideal engines</u>

Parametric cycle analysis determines the performance of engines at different flight conditions and chosen design values. The thermodynamic changes of the working fluid are studied in cycle analysis as it flows through the engine.

Parametric cycle analysis studies the engine in on-design point conditions.

Performance analysis determines the performance of a specific engine at all flight conditions and throttle settings. This means off design conditions. The components of an engine are characterized by the change in properties they produce, and a certain engine behavior is determined by its geometry.

The main objective of parametric cycle analysis is to relate the engine performance parameters (primarily thrust *F* and thrust specific fuel consumption *S)* to design choices (compressor pressure ratio, fan pressure ratio, bypass ratio, etc.), to design limitations (burner exit temperature, compressor exit pressure, etc.), and to flight environment (Mach number, ambient temperature, etc.). From parametric cycle analysis, we can determine which engine type and components are most suited for a particular need.

The value of parametric cycle analysis depends directly on the realism with which the engine components are characterized.

He engine represents a so called " rubber engine" , as a change in geometry is considered as a change of the thermodynamic parameters.

<u>Equations used in Parametric cycle analysis of ideal engines</u>

Total temperature

The total or stagnation temperature is the temperature when a steadily flowing fluid is put to stop adiabatically

$$T_t = T\left(1 + \frac{\gamma - 1}{2} M^2\right) \tag{4.1}$$

where :

T	static (thermodynamic) temperature
γ	gas constant
M	Mach number

Total pressure

The total or stagnation pressure is the temperature when a steadily flowing fluid is put to stop adiabatically and reversibly, thus isentropically.

$$P_t = P\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{4.2}$$

where:

P	static (thermodynamic) pressure

Ratios

As ratios of total temperature and pressure will be used frequently in this part, a special notation will be used for them. A subscript will indicate the component: d for diffuser (inlet), c for compressor, b for burner, t for turbine, n for nozzle, and f for fan.

$$\pi_a = \frac{total\ pressure\ leaving\ component\ a}{total\ pressure\ entering\ component\ a} \tag{4.3}$$

$$\tau_a = \frac{total\ temperature\ leaving\ component\ a}{total\ temperature\ entering\ component\ a} \tag{4.4}$$

Total / static temperature ratio – free stream

$$\tau_r = \frac{T_{to}}{T_0} = 1 + \frac{\gamma-1}{2} M_0^2 \tag{4.5}$$

Total / static pressure ratio – free stream

$$\pi_r = \frac{P_{to}}{P_0} = \left(1 + \frac{\gamma-1}{2} M_0^2\right)^{\frac{\gamma}{\gamma-1}} \tag{4.6}$$

Burner exit / ambient enthalpy ratio

$$\tau_\lambda = \frac{h_{t\ burner\ exit}}{h_0} = \frac{(c_p T_t)_{burner\ exit}}{(c_p T_o)} \tag{4.7}$$

where:

$h_{t\ \text{burner exit}}$ burner exit enthalpy

h_0 ambient enthalpy

table 4.1 contains most short form ratio’s that will be used in our analysis

Core stream		Bypass stream	
$\tau_d = \frac{T_{t2}}{T_{t0}}$	$\pi_d = \frac{P_{t2}}{P_{t0}}$	$\tau_f = \frac{T_{t13}}{T_{t2}}$	$\pi_f = \frac{P_{t13}}{P_{t2}}$
$\tau_c = \frac{T_{t3}}{T_{t2}}$	$\pi_c = \frac{P_{t3}}{P_{t2}}$	$\tau_{fn} = \frac{T_{t19}}{T_{t17}}$	$\pi_{fn} = \frac{P_{t19}}{P_{t17}}$

$\tau_b = \frac{T_{t4}}{T_{t3}}$	$\pi_b = \frac{P_{t4}}{P_{t3}}$		
$\tau_t = \frac{T_{t5}}{T_{t4}}$	$\pi_t = \frac{P_{t5}}{P_{t4}}$		
$\tau_n = \frac{T_{t9}}{T_{t7}}$	$\pi_n = \frac{P_{t9}}{P_{t7}}$		

Table 4.1 Source: Jack D Mattingly Elements of Gas Turbine Propulsion

Assumptions of ideal cycle analysis

Following assumptions are made for ideal cycle analysis:

1. There are isentropic (reversible and adiabatic) compression and expansion processes in the inlet (diffuser), compressor, fan, turbine, and nozzle resulting in these relationships:

$$\tau_d = \tau_n = 1 \qquad \pi_d = \pi_n = 1 \qquad \tau_c = \pi_c^{\frac{(\gamma-1)}{\gamma}} \qquad \tau_t = \pi_t^{\frac{\gamma-1}{\gamma}}$$

2. Constant pressure combustion (π_b =1)is idealized as a heat interaction into the combustor
   The fuel flow rate is a lot smaller than the airflow through the combustor so:

$$\frac{\dot{m}_f}{\dot{m}_c} \ll 1 \qquad \text{and} \qquad \dot{m}_c + \dot{m}_f \cong \dot{m}_c$$

3. The fluidum will be air, which will be considered as a perfect gas with constant specific heats
4. The engine exhaust nozzles expand the gas to the ambient pressure.(P_0=P_e)

Design inputs

All the possible inputs can be classified in one of four categories:

1. Flight conditions $P_0, T_0, M_0, c_p, \pi_r, \tau_r$
2. Design limits $(c_p Tt)_{burner\ exit}$
3. Component performance π_d, π_b, π_n,etc
4. Design choices π_c, π_f,etc

<u>Parametric cycle analysis for the ideal turbofan engine</u>

The propulsive efficiency of a turbojet engine can easily be increased by equipping it with a fan. This ducted ”propeller” - will be driven by the engine ‘s gas generator by extracting a potion of its energy.

The fan increases the mass flow rate and decreases the required propellant exit velocity for a given thrust. The net effect of this installation will be a decrease in wasted kinetic energy and an improvement in propulsive efficiency.

The ratio of the fan flow to the core flow is defined as the bypass ratio α

$$\alpha = \frac{\dot{m}_F}{\dot{m}_C} \tag{4.8}$$

The total gas flow can expressed as follows:

$$\dot{m}_0 = \dot{m}_F + \dot{m}_C = (1+\alpha)\,\dot{m}_C \tag{4.9}$$

<u>Assumption for ideal cycle analysis</u>

The thrust of the ideal turbofan engine is

$$F = \frac{\dot{m}_C}{g_c}(V_9 - V_0) + \frac{\dot{m}_F}{g_c}(V_{19} - V_0) \tag{4.10}$$

Substitution of equation (4.9) in (4.10) leads to:

$$\frac{F}{\dot{m}_0} = \frac{a_0}{g_c}\frac{1}{1+\alpha}\left[\frac{V_9}{a_0} - M_0 + \alpha\left(\frac{V_{19}}{a_0} - M_0\right)\right] \tag{4.11}$$

The core stream of the turbofan engine

The core stream will encounter the same elements as the ideal turbojet

$$\left(\frac{V_9}{a_0}\right)^2 = \frac{T_9}{T_0}M_9^2 \tag{4.12}$$

$$\frac{T_9}{T_0} = \tau_b M_9^2 = \frac{\tau_\lambda}{\tau_r \tau_c} \tag{4.13}$$

$$M_9^2 = \frac{2}{\gamma - 1}(\tau_r \tau_c \tau_t - 1) \tag{4.14}$$

Thus:

$$\left(\frac{V_9}{a_0}\right)^2 = \frac{T_9}{T_0} M_9^2 = \frac{2}{\gamma - 1} \frac{\tau_\lambda}{\tau_r \tau_c} (\tau_r \tau_c \tau_t - 1) \tag{4.15}$$

The fan has neither combustor nor turbine, adaption of the core stream formula gives:

$$\left(\frac{V_{19}}{a_0}\right)^2 = \frac{T_{19}}{T_0} M_{19}^2 \tag{4.16}$$

$$T_{19} = T_0 \tag{4.17}$$

$$M_{19}^2 = \frac{2}{\gamma - 1} \left(\tau_r \tau_f - 1\right) \tag{4.18}$$

Thus:

$$\left(\frac{V_{19}}{a_0}\right)^2 = M_{19}^2 = \frac{2}{\gamma - 1} \left(\tau_r \tau_f - 1\right) \tag{4.19}$$

The law of energy for the steady state flow applied on the burner

$$\dot{m}_c c_p T_{t3} + \dot{m}_f h_{PR} = (\dot{m}_c + \dot{m}_f) c_p T_{t4} \tag{4.20}$$

Defining fuel to air ratio

$$f = \frac{\dot{m}_f}{\dot{m}_c} = \frac{c_p T_0}{h_{PR}}(\tau_\lambda - \tau_r \tau_c) \quad (4.21)$$

Output power of the turbine

$$\dot{W_t} = \left(\dot{m}_c + \dot{m}_f \right) c_p (T_{t4} - T_{t5}) \cong \dot{m}_c c_p T_{t4}(1 - \tau_t) \quad (4.22)$$

Power required to drive the compressor

$$\dot{W}_c = \dot{m}_c c_p (T_{t3} - T_{t2}) = \dot{m}_c c_p T_{t2}(\tau_c - 1) \quad (4.23)$$

Power required to drive the fan

$$\dot{W}_f = \dot{m}_F c_p (T_{t13} - T_{t2}) = \dot{m}_F c_p T_{t2}(\tau_f - 1) \quad (4.24)$$

Since $\dot{W}_t = \dot{W_c} + \dot{W_f}$ for the ideal turbofan , then

$$T_{t4}(1 - \tau_t) = T_{t2}(\tau_c - 1) + \alpha T_{t2}(\tau_f - 1) \quad (4.25)$$

$$\tau_t = 1 - \frac{T_{t2}}{T_{t4}}\left[\tau_c - 1 + \alpha(\tau_f - 1)\right] \quad (4.26)$$

$$\tau_t = 1 - \frac{\tau_r}{\tau_\lambda}\left[\tau_c - 1 + \alpha(\tau_f - 1)\right] \quad (4.27)$$

Combination of equation (4.15) and (4.27) gives

$$\left(\frac{V_9}{a_0}\right)^2 = \frac{2}{\gamma-1}\frac{\tau_\lambda}{\tau_r\tau_c}\left(\tau_r\tau_c\left\{1-\frac{\tau_r}{\tau_\lambda}\left[\tau_c-1+\alpha(\tau_f-1)\right]\right\}-1\right) \qquad (4.28)$$

After simplifying this becomes

$$\left(\frac{V_9}{a_0}\right)^2 = \frac{2}{\gamma-1}\left\{\tau_\lambda-\tau_r\left[\tau_c-1+\alpha\left(\tau_f-1\right)\right]-\frac{\tau_\lambda}{\tau_r\tau_c}\right\} \qquad (4.29)$$

Specific thrust of the ideal turbofan

As result of equations, (4.11), (4.19), (4.27):

$$S = \frac{\dot{m}_f}{F} = \frac{f}{F/\,\dot{m}_0} = \frac{f}{(\,\dot{m}_0/\,\dot{m}_c)(F/\,\dot{m}_0)} \qquad (4.30)$$

$$S = \frac{f}{(1+\alpha)(F/\,\dot{m}_0)} \qquad (4.31)$$

Thermal efficiency of the ideal turbofan

$$\eta_T = 1-\frac{1}{\tau_r\tau_c} \qquad (4.32)$$

The thermal efficiency of the turbofan engine is the same as that of the turbojet engine , and this because the power that is extracted from the core stream of the turbofan engine is added to the bypass stream without loss in the ideal case, resulting in the net output power.

Thermal efficiency of the ideal turbofan

$$\eta_P = 2\frac{\frac{V_9}{V_0}-1+\alpha\left(\frac{V_{19}}{V_0}-1\right)}{\frac{V_9^2}{V_0^2}-1+\alpha\left(\frac{V_{19}^2}{V_0^2}-1\right)} \tag{4.33}$$

Thrust ratio

This is the ratio of specific thrust per unit of mass flow and is a useful performance parameter

$$FR \equiv \frac{F_C/\dot{m}_C}{F_F/\dot{m}_F} \tag{4.34}$$

Optimum Bypass Ratio

There can be stated if all other parameters are fixed, there exists an optimum bypass ratio. This optimum bypass ratio gives the minimum TSFC

$$\propto^* = \frac{1}{\tau_r(\tau_f-1)}\left[\tau_\lambda - \tau_r(\tau_c-1) - \frac{\tau_\lambda}{\tau_r\tau_c} - \frac{1}{4}\left(\sqrt{\tau_r\tau_f-1}+\sqrt{\tau_r-1}\right)^2\right] \tag{4.35}$$

When the bypass ratio is chosen to give the minimum fuel consumption, the thrust per unit mass flow one half that of the fan, this means that for an optimum bypass ratio ideal turbofan engine, the thrust ratio is 0.5

$$FR = \frac{V_9-V_0}{V_{19}-V_0} = \frac{1}{2} \tag{4.36}$$

Using this formula , the specific thrust becomes the following:

$$\left(\frac{F}{\dot{m}_0}\right)_{\alpha^*} = \frac{a_0}{g_c}\frac{1+2\alpha^*}{2(1+\alpha^*)}\left[\sqrt{\frac{2}{\gamma-1(\tau_r\tau_f-1)}} - M_0\right] \tag{4.37}$$

The propulsive efficiency at optimum bypass ratio given by:

$$(\eta_P)_{\min S} = \frac{4(1+2\alpha^*)V_0}{(3+4\alpha^*)V_0+(1+4\alpha^*)V_{19}} \tag{4.38}$$

Optimum Fan pressure Ratio

For all other parameters fixed there exists an optimum fan pressure ratio which gives minimum fuel specific fuel consumption and maximum specific thrust.

There can be concluded that optimum fan pressure ratio will be obtained when

$$V_9 = V_{19} \tag{4.39}$$

and

$$FR = 1 \tag{4.40}$$

giving:

$$\tau_f^* = \frac{\tau_\lambda \tau_r(\tau_c - 1) - \frac{\tau_\lambda}{\tau_r \tau_c} + \alpha\tau_r + 1}{\tau_r(1+\alpha)} \tag{4.41}$$

The specific thrust for an optimum fan pressure ratio turbofan becomes:

$$\left(\frac{F}{\dot{m}_0}\right)_{r_f^*} = \frac{a_0}{g_c}\left[\sqrt{\frac{2}{\gamma - 1}\left(\tau_r\tau_f^* - 1\right)} - M_0\right] \tag{4.42}$$

The propulsive efficiency at optimum fan pressure ratio turbofan is:

$$(\eta_P)_{r_f^*} = \frac{2}{\frac{V_{19}}{V_0} + 1} \tag{4.43}$$

<u>Description of the Computer calculated performance for an ideal turbofan engine using the PARA v3.10 program</u>

For the parametric analysis of the turbofan engine , we will use the PARA software to obtain the necessary calculations .For the result plots,the PARA program and excel program will be used.

Figure 4.1 shows the basic interface of the PARA program

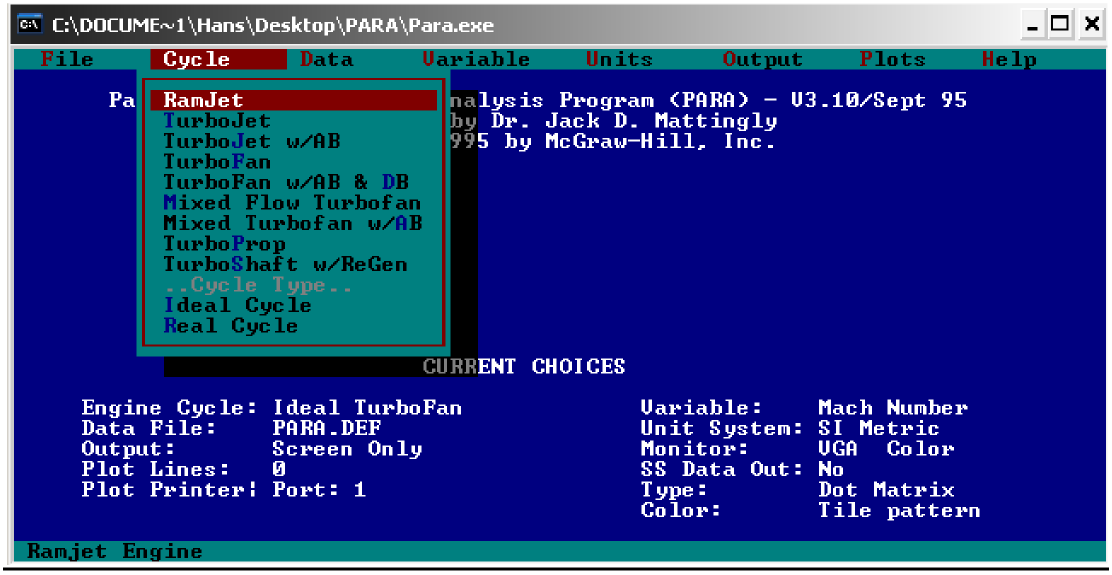


Figure 4.1. Basic interface of program Source: PARA Software

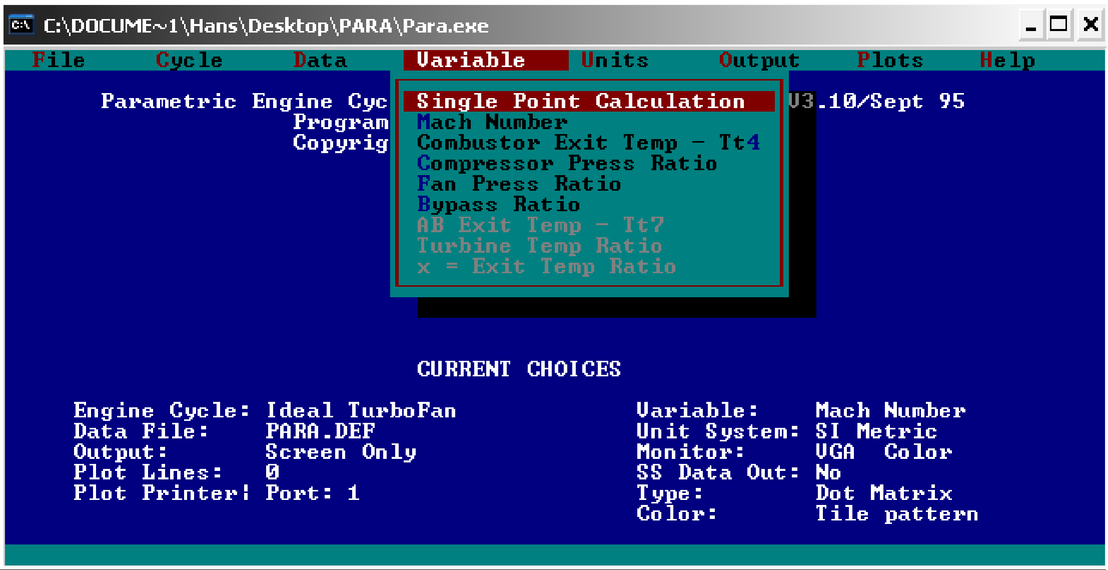


Figure 4.2 Basic interface of program Source: PARA Software

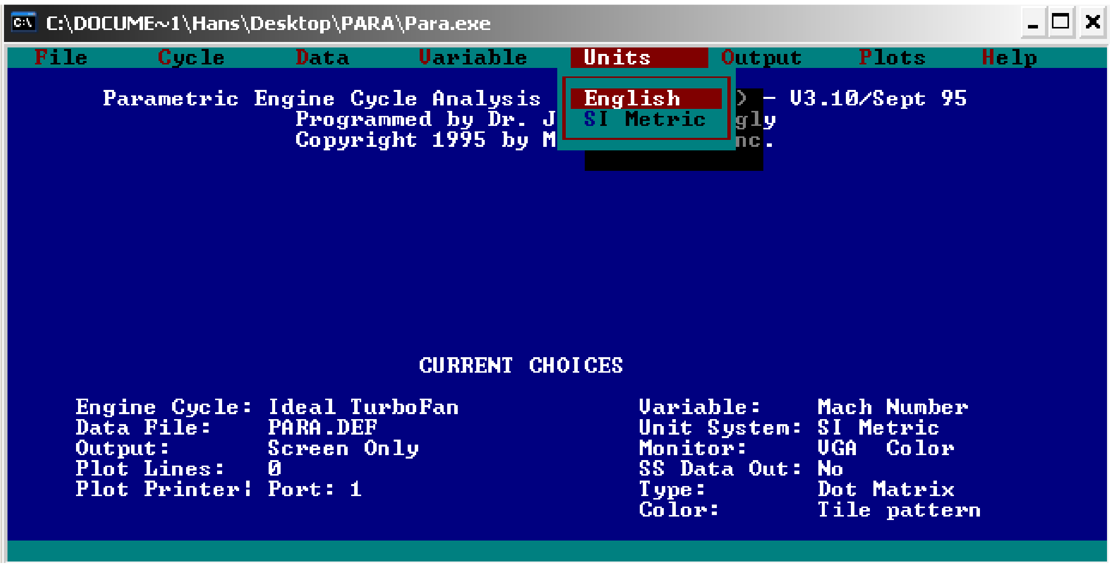


Figure 4.3. Basic interface of program Source: PARA Software

First thing we have to do is to select the **Type of cycle.** In our case we will select ”turbofan”. By selecting this type of cycle the basic data corresponding to this configuration will be uploaded in the program.

In the assumption for this cycle analysis is stated that the fluidum - air will be considered as a perfect gas , meaning constant specific heats. This is the second selection we make under the option **ideal (perfect)gas model.**

Under the option **Unit system**, SI metric will be selected , as we are interested to have the results in SI units.

Last thing we have to select in the basis scope of this program is the **Iteration variable**, define maximum and minimum value, as well as the iteration steps.

When this done, we proceed to **view data** , where we further can set the parameters corresponding to the flight conditions as well as engine design variables and efficiencies.

The input data for the turbofan engine is presented in figure 4.4

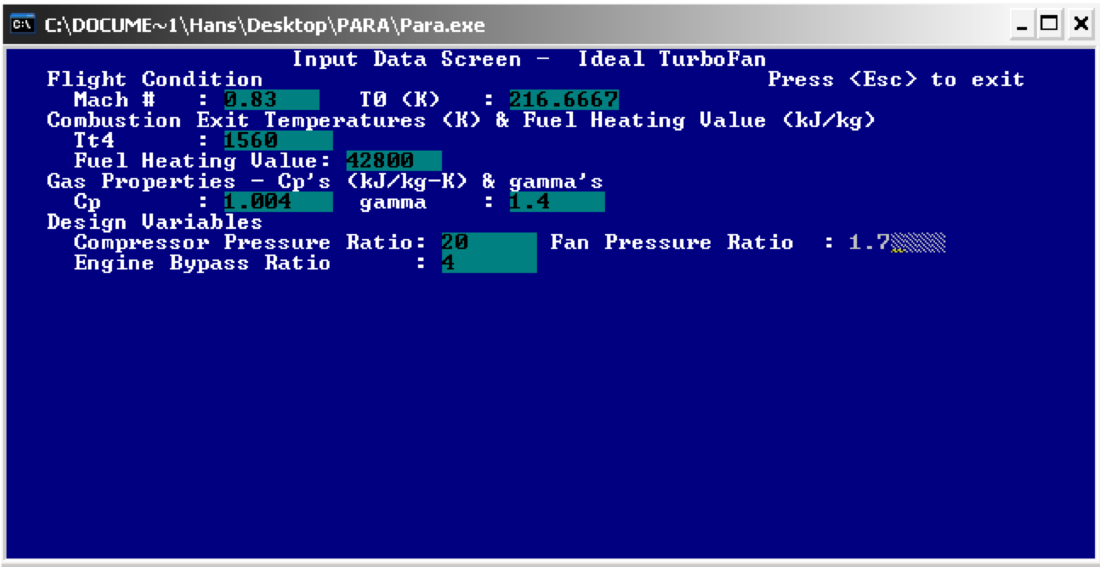


Figure 4.4. Basic interface of program Source: PARA Software

In this part of the program follwing data will be set:

For the flight conditions:

- Mach number
- Altitude
- Ambient temperature
- Ambient pressure
- Specific heat with constant pressure
- Gas constant gamma
- low heating value of fuel
- temperature before entering turbine

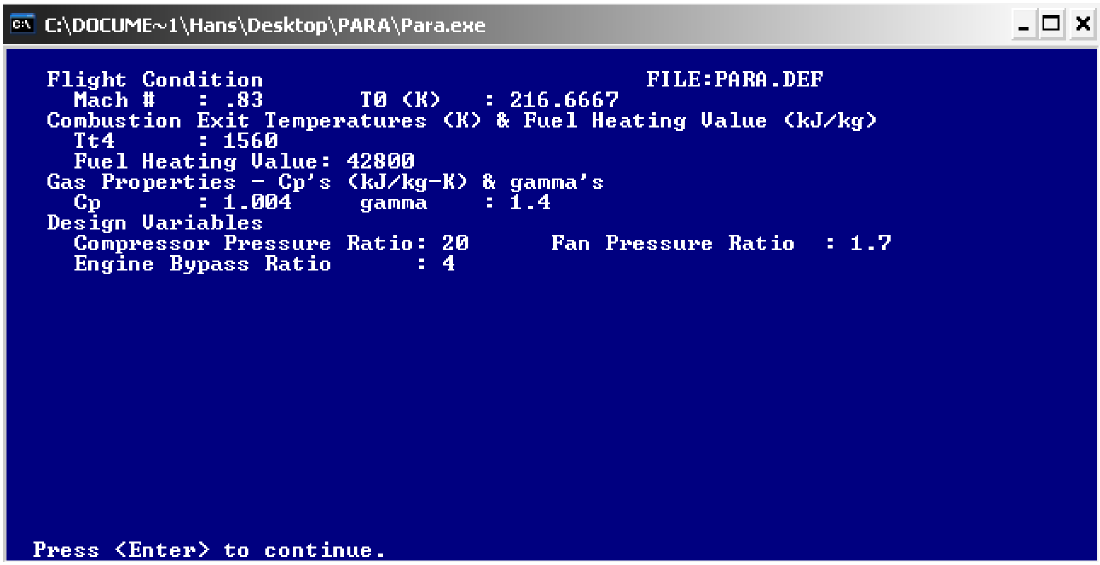
for the engine design variables:

- compressor pressure ratio
- LPC pressure ratio
- Fan pressure ratio
- Bypass ratio

Ideal component efficiencies will be used.

After all parameters are set, we choose **perform calculations.**

Figure 4.5 shows the output file of the PARA program for the chosen settings

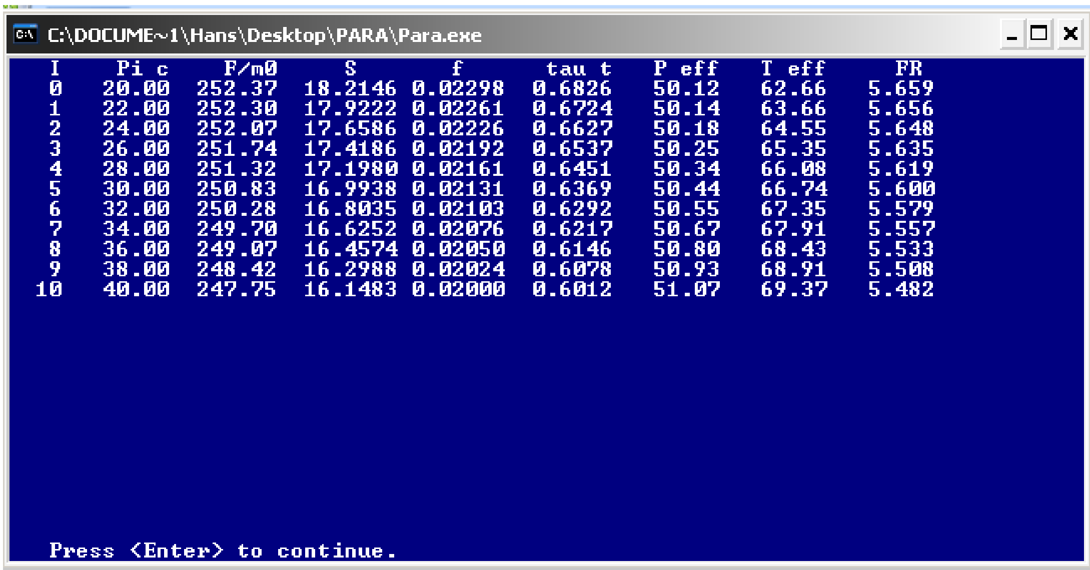


```
C:\DOCUME~1\Hans\Desktop\PARA\Para.exe

 Flight Condition                            FILE:PARA.DEF
   Mach #    : .83        T0 (K)    : 216.6667
 Combustion Exit Temperatures (K) & Fuel Heating Value (kJ/kg)
   Tt4       : 1560
   Fuel Heating Value: 42800
 Gas Properties - Cp's (kJ/kg-K) & gamma's
   Cp        : 1.004      gamma     : 1.4
 Design Variables
   Compressor Pressure Ratio: 20       Fan Pressure Ratio   : 1.7
   Engine Bypass Ratio       : 4

Press <Enter> to continue.
```

Figure 4.5. Basic interface of program Source: PARA Software



C:\DOCUME~1\Hans\Desktop\PARA\Para.exe

I	Pi c	F/m0	S	f	tau t	P eff	T eff	FR
0	20.00	252.37	18.2146	0.02298	0.6826	50.12	62.66	5.659
1	22.00	252.30	17.9222	0.02261	0.6724	50.14	63.66	5.656
2	24.00	252.07	17.6586	0.02226	0.6627	50.18	64.55	5.648
3	26.00	251.74	17.4186	0.02192	0.6537	50.25	65.35	5.635
4	28.00	251.32	17.1980	0.02161	0.6451	50.34	66.08	5.619
5	30.00	250.83	16.9938	0.02131	0.6369	50.44	66.74	5.600
6	32.00	250.28	16.8035	0.02103	0.6292	50.55	67.35	5.579
7	34.00	249.70	16.6252	0.02076	0.6217	50.67	67.91	5.557
8	36.00	249.07	16.4574	0.02050	0.6146	50.80	68.43	5.533
9	38.00	248.42	16.2988	0.02024	0.6078	50.93	68.91	5.508
10	40.00	247.75	16.1483	0.02000	0.6012	51.07	69.37	5.482

Press <Enter> to continue.

Figure 4.6. Basic interface of program Source: PARA Software

Calculation of the ideal engine parameters

In this section an overview of the input data used for the PARA program, results and graphs of the results will be presented.

Input data for the PARA program

Constants:

- Mach number M=0.83
- Altitude H=11000m
- Ambient temperature T= 216.65 K
- Ambient pressure P = 22.632 kPa
- Specific heat with constant pressure C_p=1.004 $\frac{kJ}{kg \times K}$
- Gas constant gamma γ = 1.4
- low heating value of fuel 42 800 $\frac{kJ}{kg}$
- temperature before entering turbine T_{t4} = 1560 k

Variable:

- Evaluation of bypass ratio's 4 ; 6 ; 8 ; 10 ;12 and optimum bypass ratio 11,7
- Variation of the compressor pressure ratio with a minimum of 20 to a maximum of 40 with increments of 2.

Results obtained from the PARA program

In figure 4.7, the trust specific fuel consumption is plotted versus versus the specific thrust.

In order to calculate the minimum uninstalled specific thrust, we need the drag value obtained from value obtained from figure

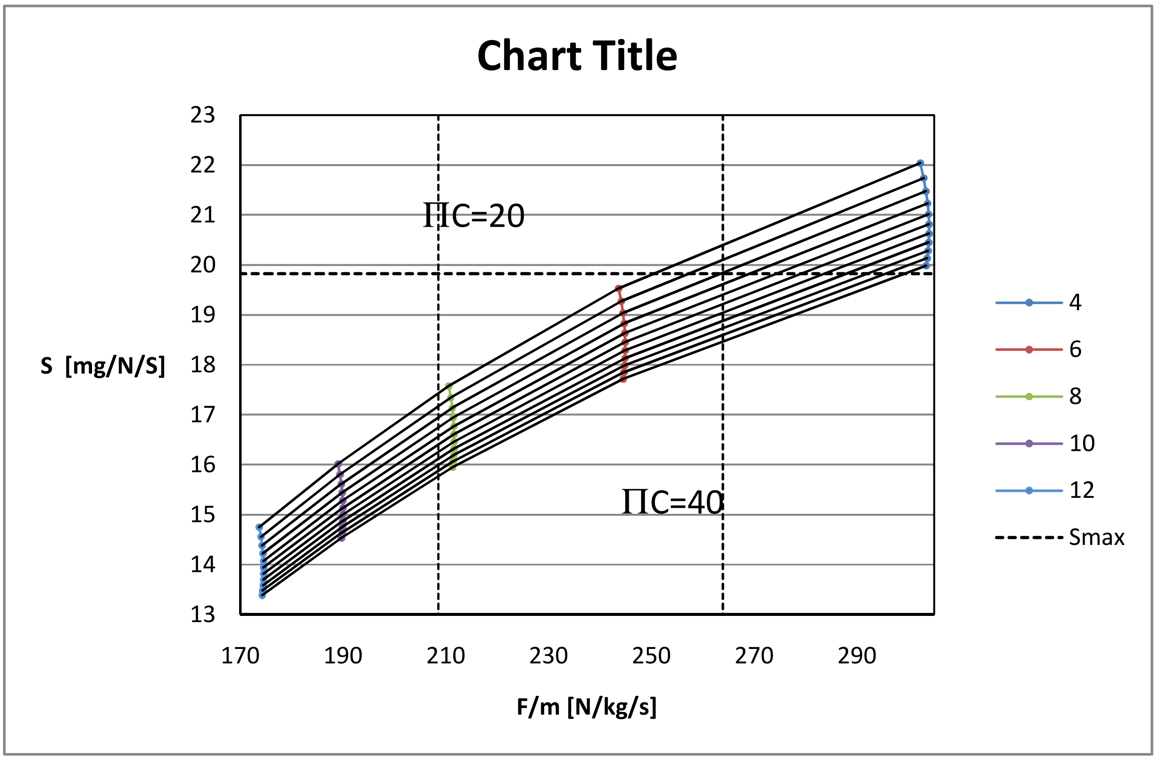


Figure 4.7. Specific trust versus specfic fuel consumption

to apply in following formula:

From this graph we know that the best engine performance which corresponds to the specific minimum fuel consumption maximum specific thrust for the following range of parameters

$$4.5 \leq \alpha \leq 5.5$$

$$24 \leq \pi_c \leq 40$$

$$2m \leq dt \leq 2.25\, m$$

And the specific thrust will have the following range:

$$\frac{F}{\dot{m}} = 267 \; to \; 272 \frac{N}{kg}/s$$

If we compare the values if with the ideal cycle

$$\alpha \approx 10$$

$$32 \leq \pi_c \approx\leq 40$$

$$dt = 2.5\, m$$

We can observe that the compressor ratio , the bypass ratio and the fan diameter have greater values than in case of the ideal cycle .These differences can be explained by the energy losses that occur during the real cycle here.

Calculation of the minimum specific thrust at cruising speed M = 0,83

$$\frac{T-(D+R)}{W} = \frac{P_S}{V} \tag{4.44}$$

where:

R additional drag (landing gear, antennas) and is considered zero

W= 1563472 N (95% of the maximum gross take off weight .)

D = 78000 N

P_s=1,5 m/s

Now the minimum required installed thrust can be determined

T = 87598 N

Assuming that $\phi_{nozzle} + \phi_{inlet} = 0,02$ and knowing following relationship:

$$T = F(1 - \phi_{inlet} - \phi_{nozzle}) \tag{4.45}$$

We can calculate the required uninstalled thrust

F= 89386 N

Determination of the maximum airflow in the engine

The subsonic inlet is constructed in such a way that in the throat M=0.8 is not exceeded.

This is a limiting Mach number which will correspond with the maximum corrected engine mass flow that must pass the throat.

Using this information the throat diameter can be calculated with:

$$d_t = 0,1636\sqrt{\dot{m}_{c0max}} \quad (4.46)$$

where:

d_t throat diameter

$\dot{m}_{c0max}$ maximum corrected engine mass flow

For a given diameter, the maximum airflow that passes the engine can now be calculated

$$\dot{m}_{c0max} = \left(\frac{d_t}{0,1636}\right)^2 \quad (4.47)$$

The result of this formula will be in English units, meaning d_t in feet and the mass flow in pound mass per second.

A conversion will be required to SI units.

$$\dot{m}_{c0max}(SI) = \quad \dot{m}_{c0max}(ENG) \times 0,454$$

The maximum airflow through the engine in function of the diameter is shown in table 4.2

dt (m)	dt (ft)	m (kg/s)	F/m (N/kg/s)
1.8	5.9	275.1	325.8
2	6.6	339.5	263.9
2.25	7.4	429.7	208.5
2.5	8.2	530.5	168.9
2.75	9.0	641.9	139.6
3	9.8	763.9	117.3
3.25	10.7	896.6	99.9

Table 4.2 Maximum airflow in function of diameter

Calculation of the maximum allowable uninstalled thrust specific fuel consumption

From figure 2.5 we can determine the installed thrust specific fuel consumption in cruise conditions, TSFC = 0.71 lbm/(lbf .hr) (H=11000m ;M=0.83) and with following formula the uninstalled thrust specific fuel consumption

$$S = TSFC(1 - \phi_{inlet} - \phi_{nozzle}) \quad (4.48)$$

S_{max} = 19.8 mg/ (N.s)

Engine component performance

Before the parametrical cycle analysis of the entire engine can be made, engine component performance must be analyzed first.

The value of characteristic parts in a real engine will differ from those of the ideal engine.

This difference can be explained by the fact that a the real cycle is subject to energy losses.

Apart from assigning values to pressure ratio's π and temperature ratio's τ, we now also must assign values to the isentropic efficiencies η and the polytrophic efficiencies e

For the calculation we can use the data from table 5.1which contains engine data for engines used at present as well as data for engines to be used in the future.

In following calculations ,the data for level of technology 4 will be used

TABLE 6-2
Component figures of merit for different technological levels

Component	Figure of merit	Type	Level of technology 1	2	3	4
Diffuser	$\pi_{d\,max}$	*A*	0.90	0.95	0.98	0.995
		B	0.88	0.93	0.96	0.98
		C	0.85	0.90	0.94	0.96
Compressor	e_c		0.80	0.84	0.88	0.90
Fan	e_f		0.78	0.82	0.86	0.89
Burner	π_b		0.90	0.92	0.94	0.95
	η_b		0.85	0.91	0.98	0.99
Turbine	e_t	Uncooled	0.80	0.85	0.89	0.90
		Cooled		0.83	0.87	0.89
Afterburner	π_b		0.90	0.92	0.94	0.95
	η_b		0.85	0.91	0.96	0.99
Nozzle	π_n	*D*	0.95	0.97	0.98	0.995
		E	0.93	0.96	0.97	0.98
		F	0.90	0.93	0.95	0.97
Maximum T_{t4}		(K)	1110	1390	1780	2000
		(°R)	2000	2500	3200	3600
Maximum T_{t7}		(K)	1390	1670	2000	2220
		(°R)	2500	3000	3600	4000

A = subsonic aircraft with engines in nacelles
B = subsonic aircraft with engine(s) in airframe
C = supersonic aircraft with engine(s) in airframe
D = fixed-area convergent nozzle
E = variable-area convergent nozzle
F = variable-area convergent-divergent nozzle

Source : Elements of gas turbine propulsion ,Jack D. Mattingly, McGraw-Hill 1996

Parameteric cycle analysis of the real engine

In this chapter the formulas will be presented to calculate the parameters of the real turbofan engine, and permit real cycle analysis.

The results of the real cycle will be compared to the results of the ideal cycle.

Assumptions for the turbofan engine cycle analysis with losses

The steps of the cycle analysis will be applied on both the fan stream and the engine core stream

1.Perfect gas upstream of main burner with constant properties γ_c ,R_c ,c_{pc}, etc

2. Perfect gas downstream of main burner with constant properties γ_c ,R_c ,c_{pc}, etc

3.All components are adiabatic (no turbine cooling)

4.The efficiencies of the compressor , fan and turbine are described through the use of (constant) polytrophic efficiencies e_c, e_f and e_t respectively.

Formulas used in the analysis of the real engine

Engine fan stream

For the fan stream following equation is valid

$$\frac{F_f}{\dot{m}_F} = \frac{1}{(1+\alpha)}\frac{a_0}{g_c}\left[\frac{V_{19}}{a_0} - M_0 + \frac{\frac{T_9}{T_0}}{\frac{V_{19}}{a_0}}\frac{1-\frac{P_0}{P_9}}{\gamma_c}\right] \tag{6.1}$$

We know that

$$\left(\frac{V_{19}}{a_0}\right)^2 = \frac{T_{19}}{T_0}M_{19}^2 \tag{6.2}$$

The Mach number in the bypass nozzle is

$$M_{19}^2 = \frac{2}{\gamma_c - 1}\left[\left(\frac{P_{t19}}{P_{19}}\right)^{\frac{\gamma_c - 1}{\gamma_c}}\right] \tag{6.3}$$

Where

$$\frac{P_{t19}}{P_0} = \pi_r \pi_d \pi_f \pi_{fn} \tag{6.4}$$

and

$$\frac{T_{19}}{T_0} = \frac{\tau_f \tau_r}{\left(\frac{P_{t19}}{P_{19}}\right)^{\frac{\gamma_c - 1}{\gamma_c}}}$$

(6.5)

where

$$\frac{T_{19}}{T_0} = \tau_f \tau_r \tag{6.6}$$

Engine core stream

For the core stream following equation is valid

$$F_c = \frac{1}{g_c}(\dot{m}_c V_9 - \dot{m}_c V_0) + A_9(P_9 - P_0) \tag{6.7}$$

$$\frac{F_c}{\dot{m}_c} = \frac{a_0}{g_c}\left[(1+f)\frac{V_9}{V_0} - M_0 + (1+f)\frac{R_t}{R_c}\frac{\frac{T_9}{T_0}}{\frac{V_9}{a_0}}\frac{1-\frac{P_0}{P_9}}{\gamma_c}\right] \tag{6.8}$$

and the fuel /air ratio for the main burner is defined as

$$f \equiv \frac{\dot{m}_f}{\dot{m}_c} \tag{6.9}$$

We know that

$$\left(\frac{V_9}{a_0}\right)^2 = \frac{\gamma_t R_t T_9}{\gamma_c R_c T_0} M_9^2 \tag{6.10}$$

The Mach number in the core nozzle is

$$M_9^2 = \frac{2}{\gamma - 1}\left[\left(\frac{P_{t9}}{P_9}\right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1\right] \tag{6.11}$$

where

$$\frac{P_{t9}}{P_9} = \frac{P_0}{P_9}\pi_r \pi_d \pi_c \pi_b \pi_t \pi_n \tag{6.12}$$

And

$$\frac{T_9}{T_0} = \frac{T_{t9}/T_0}{(P_{t9}/P_9)^{(\gamma_c - 1/\gamma_c)}} \tag{6.13}$$

where

$$\frac{T_9}{T_0} = \tau_r \tau_d \tau_c \tau_b \tau_t \tau_n = \frac{C_{pc}}{C_{pt}} \tau_\lambda \tau_t \tag{6.14}$$

Application of the first law of thermodynamics to the burner

$$\dot{m}_C C_{pc} T_{t3} + \eta_b \dot{m}_f h_{PR} = \dot{m}_4 C_{pt} T_{t4} \qquad (6.15)$$

By using the definitions of temperature ratio's and fuel /air ratio , this equation becomes:

$$\tau_r \tau_c + f \frac{\eta_b h_{PR}}{C_{pc} T_0} = (1+f)\tau_\lambda \qquad (6.16)$$

Solving for f gives

$$f = \frac{\tau_\lambda \tau_r \tau_c}{\frac{\eta_b h_{PR}}{C_{pc} T_{t3}} - \tau_\lambda} \qquad (6.17)$$

The power balance between fan , compressor and turbine with a mechanical efficiency η_m of the coupling between turbine ,compressor and fan gives

$$\dot{m}_c c_{pc}(T_{t3} - T_{t2}) + \dot{m}_F c_{pc}(T_{t13} - T_{t2}) = \eta_m \dot{m}_4 c_{pt}(T_{t4} - T_{t5}) \qquad (6.18)$$

power into compressor power into fan net power from turbine

Dividing equation (6.18) by $\dot{m}_C C_{pc} T_{t2}$ and using the definitions of temperature ratios, fuel /air, and the bypass ratio , we obtain

$$\tau_c - 1 + \alpha(\tau_f - 1) = \eta_m (1+f) \frac{\tau_\lambda}{\tau_r}(1 - \tau_t) \qquad (6.19)$$

Solving for the turbine temperature ratio gives

$$\tau_t = 1 - \frac{1}{\eta_m (1+f)} \frac{\tau_r}{\tau_\lambda}\left[\tau_c - 1 + \alpha(\tau_f - 1)\right] \qquad (6.20)$$

Turbine pressure ratio

$$\pi_t = \tau_t^{\gamma_c/[(\gamma_c-1)e_t]} \tag{6.21}$$

Isentropic turbine efficiency

$$\eta_t = \frac{1-\tau_t}{1-\tau_t^{1/e_t}} \tag{6.22}$$

Compressor Temperature ratio in faction of compressor pressure ratio

$$\tau_c = \pi_c^{(\gamma_c-1)/(\gamma_c e_c)} \tag{6.23}$$

Isentropic compressor efficiency

$$\eta_c = \frac{\pi_c^{(\gamma_c-1)/\gamma_c}-1}{\tau_c-1} \tag{6.24}$$

Temperature ratio of the combustion chamber

$$\tau_\lambda = \frac{C_{pt}T_{t4}}{C_{pc}T_0} \tag{6.25}$$

Fan temperature ratio

$$\tau_f = \pi_f^{(\gamma_c-1)/(\gamma_c e_f)} \tag{6.26}$$

Fan polytropic efficiency

$$\boldsymbol{\eta_f = \frac{\pi_f^{(\gamma_c-1)/\gamma_c}-1}{\tau_f-1}} \tag{6.27}$$

If we combine the thrust equation for the fan stream and the engine core stream, we get:

$$\frac{F_C}{\dot{m}_C} = \frac{1}{1+\alpha}\frac{a_0}{g_C}\left((1+f)\frac{V_9}{a_0} - M_0 + (1+f)\frac{R_t}{R_c}\frac{\frac{T_9}{T_0}}{\frac{V_9}{a_0}}\frac{1-\frac{P_0}{P_9}}{\gamma_c}\right) + \frac{\alpha}{1+\alpha}\frac{a_0}{g_c}\left(\frac{V_{19}}{a_0} - M_0 + \frac{\frac{T_{19}}{T_0}}{\frac{V_{19}}{a_0}}\frac{1-\frac{P_0}{P_{19}}}{\gamma_c}\right) \tag{6.28}$$

The thrust specifis fuel consumption S is

$$S = \frac{\dot{m}_f}{F} = \frac{\dot{m}_f/\dot{m}_C}{(\dot{m}_f/\dot{m}_C)(F/\dot{m}_0)} \tag{6.29}$$

or

$$S = \frac{f}{(1+\alpha)(F/\dot{m}_0)} \tag{6.30}$$

Engine thermal efficiency

$$\eta_T = \frac{a_0^2[(1+f)(V_9/a_0)^2+\alpha(V_{19}/a_0)^2-(1+\alpha)M_0^2]}{2g_c f h_{PR}} \tag{6.31}$$

Engine propulsive efficiency

$$\eta_P = \frac{2M_0[(1+f)(V_9/a_0)+\alpha(V_{19}/a_0)-(1+\alpha)M_0}{(1+f)(V_9/a_0)^2+\alpha(V_{19}/a_0)^2-(1+\alpha)M_0^2} \tag{6.32}$$

Description of the Computer calculated performance for a real turbofan engine using the PARA v4.0 program

For the parametric analysis of the turbofan engine , we will use the PARA software to obtain the necessary calculations .For the result plots,the PARA program and excel program will be used.

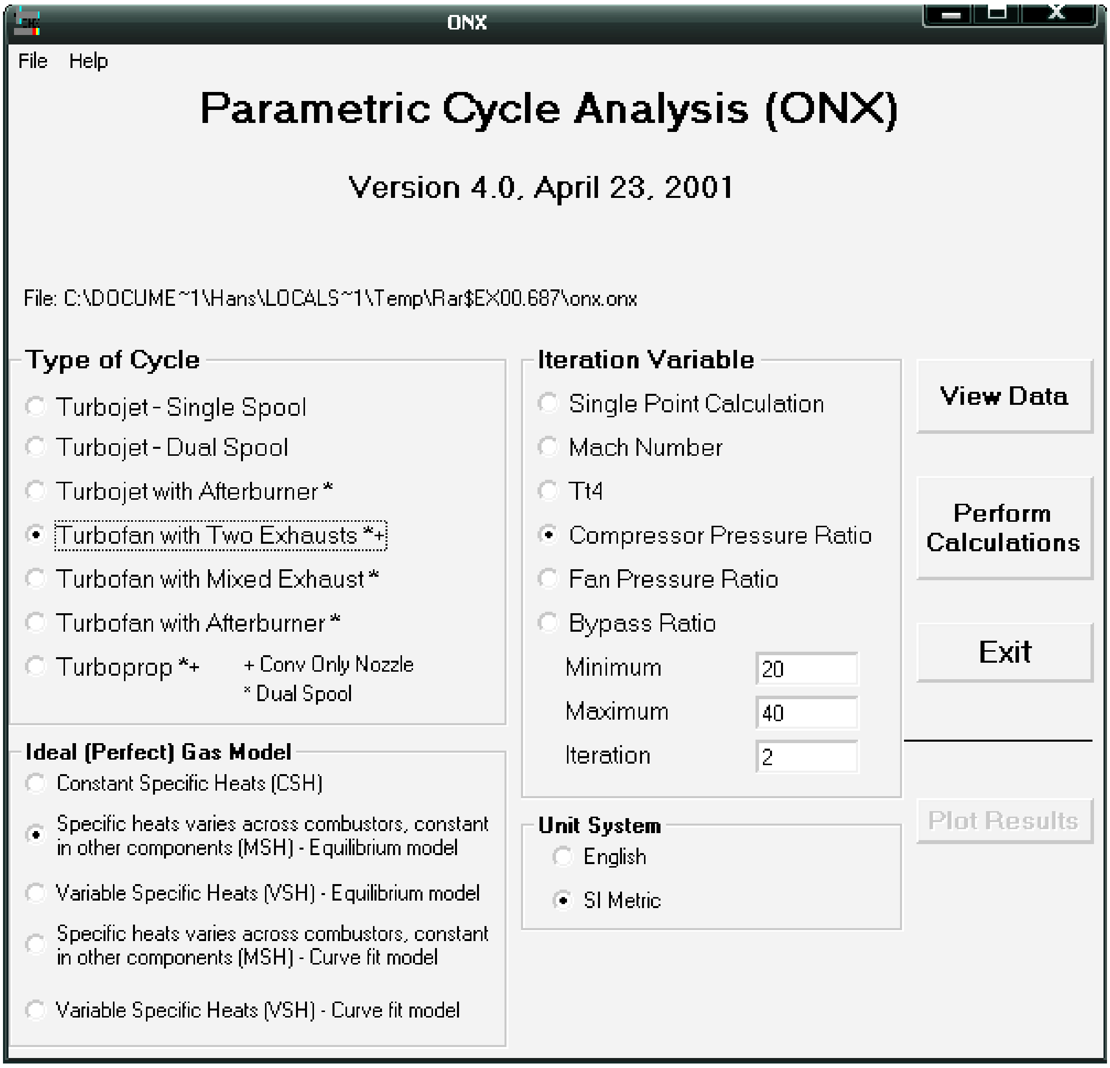


Figure 6.1. Basic interface of the PARA program

First thing we have to do is to select the **Type of cycle.** In our case we will select ”turbofan with two exhausts”. By selecting this type of cycle the basic data corresponding to this configuration will be uploaded in the program.

In the assumption for this cycle analysis is stated that the fuidum – air will be considered as a perfect gas , with constant specific heat c_{pc} upstream of the combustor , and a perfect gas with constant specific heat c_{pt} downstream of the combustor. This is the second selection we make under the option **ideal (perfect)gas model.**

Under the option **Unit system**, SI metric will be selected , as we are interested to have the results in SI units.

Last thing we have to select in the basis scope of this program is the **Iteration variable**, define maximum and minimum value ,as well as the iteration steps.

When this done, we proceed to **view data**, where we further can set the parameters corresponding to the flight conditions as well as engine design variables and efficiencies.

The input data for the turbofan engine is presented in figure 6.2

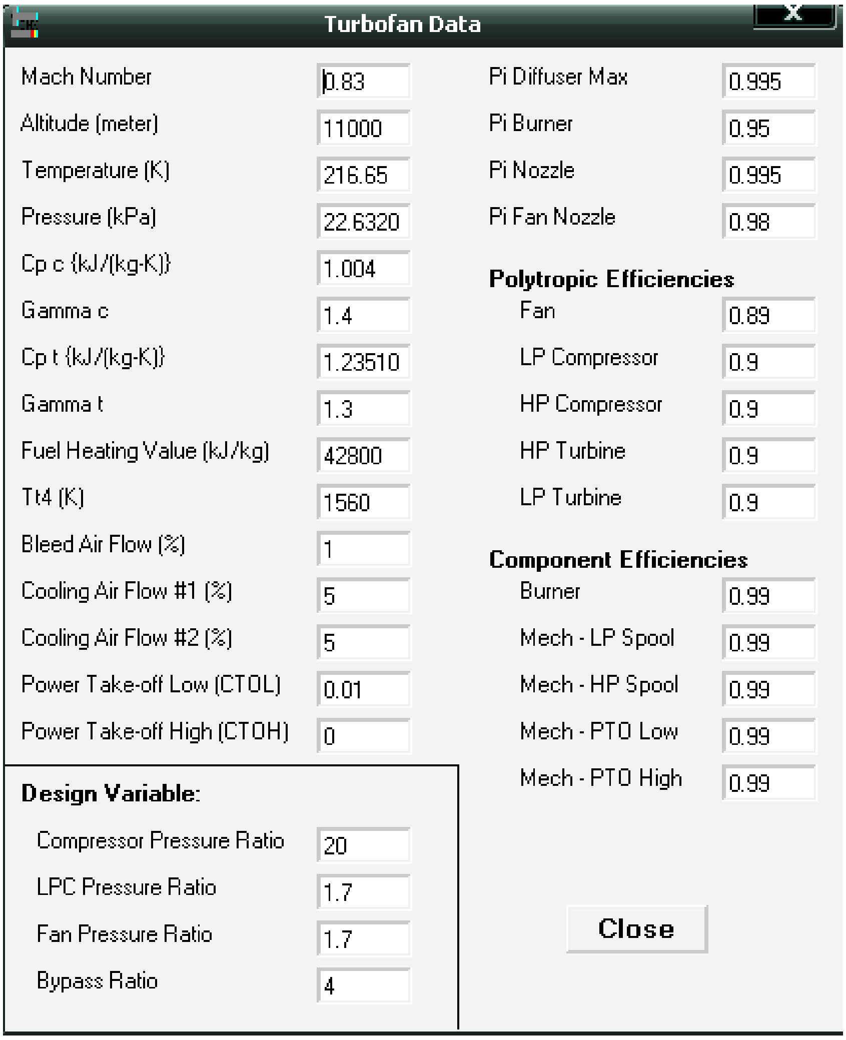


Figure 6.2 input data turbofan engine

Results

```
    On-Design Calcs (ONX V4.0)              Date: 5/24/2008 7:06:46 PM
    File: C:\Documents and Settings\Hans\Desktop\Turbofan_engine\BPR4.onx
             Turbofan Engine with Two Exhausts & Conv. Nozzles
                    with Constant Specific Heats Model
    **********************        Input Data       **********************
False            (kg/m^3)                            Efficiency
          Cp c       = 1.0040 kJ/kg-K       Burner        =  0.990
          Cp t       = 1.2351 kJ/kg-K     Mech Hi Pr    =  0.990
          Gamma c    = 1.4000               Mech Lo Pr    =  0.990
          Gamma t    = 1.3000               Fan/LP Comp   =0.890/0.900 (ef/ecL)
          Tt4 max    = 1560.0 K             HP Comp       =  0.900 (ecH)
          h - fuel   =  42800 kJ/kg         HP Turbine    =  0.900 (etH)
          CTO Low    = 0.0100               LP Turbine    =  0.900 (etL)
          CTO High   = 0.0000               Pwr Mech Eff L =  0.990
          Cooling Air #1 =  5.000 %         Pwr Mech Eff H =  0.990
          Cooling Air #2 =  5.000 %         Bleed Air      =  1.000 %
    *************************    RESULTS    *************************
          Tau r      =  1.138               a0 (m/sec)    =  295.0
          Pi r       =  1.571               V0 (m/sec)    =  244.8
          Tau L      =  8.858
          Tau f      = 1.1857               Pt19/P0       =  2.658
          Eta f      = 0.8815               Tt19/T0       = 1.3491
          Pt19/P19   = 1.8929               P0/P19        = 0.7268
          M19        = 1.0000               V19/V0        = 1.2775
      Pi c F/mdot    S    Pt9/P9  V9/V0   T Eff   P Eff  P0/P9
     20.00 220.4725.6752  1.832 2.3017 53.123 55.091 0.4166
     22.00 219.7825.3614  1.832 2.2847 53.727 55.151 0.4097
     24.00 218.9825.0825  1.832 2.2686 54.221 55.262 0.4047
     26.00 218.0924.8319  1.832 2.2533 54.625 55.416 0.4014
     28.00 217.1424.6046  1.832 2.2387 54.955 55.602 0.3994
     30.00 216.1424.3969  1.832 2.2246 55.220 55.816 0.3984
     32.00 215.1124.2059  1.832 2.2111 55.432 56.053 0.3984
     34.00 214.0424.0294  1.832 2.1981 55.597 56.309 0.3993
```

Save   Print   Done

Calculation of the real engine parameters

In this section an overview of the input data used for the PARA program, results and graphs of the results will be presented.

Input data for the PARA program

Constants:

- Mach number M=0.83

- Altitude H=11000m
- Ambient temperature T= 216.65 K
- Ambient pressure P = 22.632 kPa
- Specific heat with constant pressure in compressor C_{pc}=1.004 $\frac{kJ}{kg\times K}$
- Specific heat with constant pressure in turbine C_{pt}=1,235 $\frac{kJ}{kg\times K}$
- Fan pressure ratio π_f =1.7
- Gas constant gamma compressor γ_c = 1.4
- low heating value of fuel h_{pr} 42 800 $\frac{kJ}{kg}$
- Gas constant gamma compressor γ_t = 1.3
- temperature before entering turbine T_{t4} = 2000 k

values obtained from table 5.1,component figures of merit for different levels of technology, level 4 components will be used

- diffuser pressure ratio π_{dmax} , burner pressure ratio π_b ,core nozzle pressure ratio π_n ,bypass nozzle pressure ratio π_{fn}
- polytropic efficiencies ; compressor e_c , turbine e_t ,fan e_f
- isentropic efficiency of the burner η_b
- Mechanical efficiency η_m

Variable:

- Evaluation of bypass ratio's 4 ; 6 ; 8 ; 10 ;12 and optimum bypass ratio 11,7
- Variation of the compressor pressure ratio with a minimum of 20 to a maximum of 40 with increments of 2.

<u>**Results obtained from the PARA program**</u>

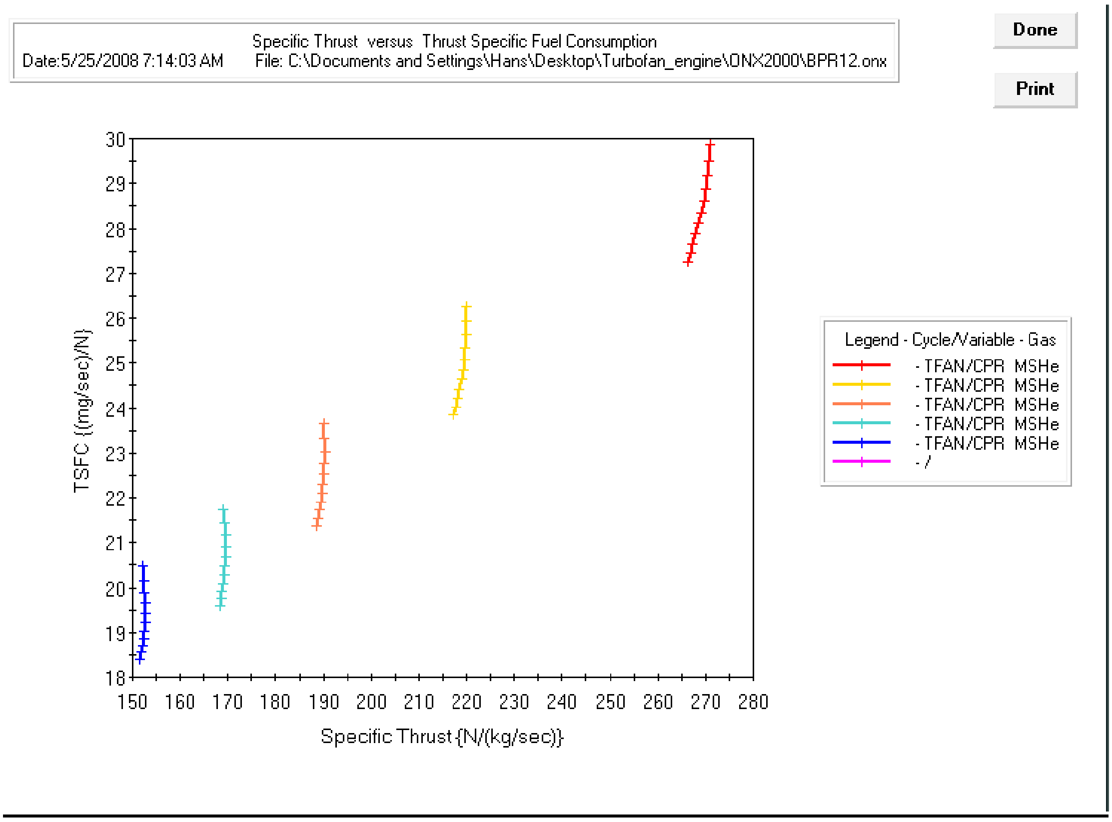


Figure 6.4. Specific Thrust versus specific fuel consumption, Source: PARA Software

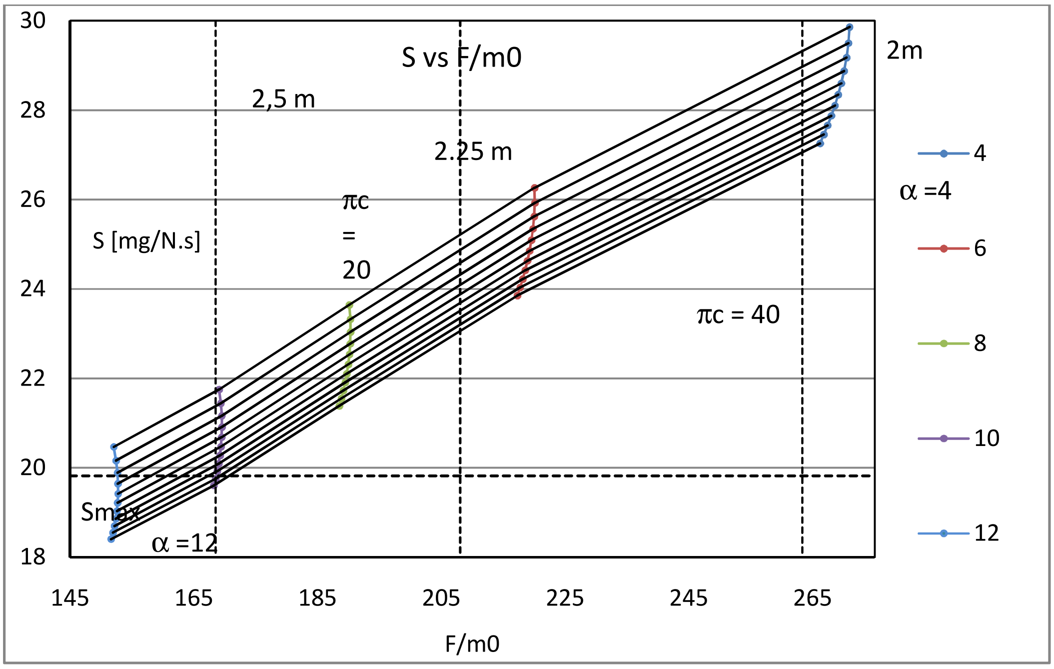


Figure 6.5. Specific Thrust versus specific fuel consumption, Source: PARA Software

<u>Engine performance analysis</u>

The required analysis to obtain engine performance is related , but very different from the parametric cycle analysis studied in earlier paragraphs.

When using parametric cycle analysis of a turbofan engine, values will be selected independently for the compressor pressure ratio, main burner exit temperature, flight conditions, etc.

When using the engine performance analysis ,the performance will be considered of an engine that was built (physical or mathematical construction),with a selected pressure ratio an d a corresponding pressure ratio and its corresponding turbine temperature ratio.

The turbine temperature remains constant (essential)for a turbojet engine, and its compressor pressure ratio is function of the throttle setting (main burner exit temperatureT_{t4}) and flight condition (M_0 and T_0).

Table 7.1 lists a comparison of analysis variables

Table 7.1 comparison, of analysis variables

Variable	**Parametric cycle**	**Engine performance**
Flight condition (M_0,T_0 and P_0)	independent	independent
Compressor pressure ratio π_c	Independent	dependent
Main burner exit temperature T_{t4}	independent	Independent
Turbine temperature ratio τ_t	dependent	Constant

Table 7.1. Source: Elements of gas turbine propulsion ,Jack D. Mattingly, McGraw-Hill 1996

In the parametric cycle analysis the main burner exit temperature and aircraft flight conditions were specified via the design inputs M_0,T_0 ,P_0 and T_{t4}.

The engine cycle was selected as well as the compressor pressure ratio, and the polytrophic efficiencies of the engine components etc.

The result of these design input variables will determine the design or reference point of the engine. The obtained specific engine thrust and fuel consumption are only valid for the chosen engine cycle and values of M_0,T_0 ,P_0 ,T_{t4}, π_c, τ_t ,η_c ,etc.

The engine we studied in the parametrical cycle analysis was a so called " rubber" engine, meaning when we change any of the above mentioned values ,the engine will change its shape and component design to meet the thermodynamical and other requirements.

When constructing and designing a turbofan engine , the degree of variability depends on the available technology, the main use of the engine, and the demands set by the designers. Most gasturbines have constant area flow passages and limited variability.

When a gasturbine is installed on an aircraft, its performance varies with flight conditions and throttle setting and is limited by the engine control system.

A good approximation of an engine's performance can be made by assuming that the component efficiencies remain constant.

The analysis of engine performance requires a model for the behavior of each engine component over its actual range of operation. Thus the more accurate and complete the model, the more accurate the results will be.

We will also take a closer look at the engine's performance in off design conditions as the engine is designed to perform in these conditions. However, the engine is likely to also operate in conditions other than design conditions and this will of course affect engine performance and the reliability of the engine

Reference values for engine performance analysis

By the application of the laws of conservation of mass, energy ,momentum and entropy for a steady one dimensional flow of a perfect gas to an engine steady-state operating point, functional relationships can be deduced which can be used to predict engine performance at different throttle settings and flight conditions.

Therefore

$$f(\pi,\tau) = constant$$

represents a relationship between the engine variables π and τ at a steady state operating point.

This constant can be compared to a reference condition (subscript R) :

$$f(\pi,\tau) = f(\pi_R,\tau_R) = constant$$

Sea level static is the is the normal reference condition (design point) for the value of the gas turbine engine variables.

Assumptions for engine performance analysis

1. The flow is choked at the high pressure turbine entrance nozzle,LPT entrance nozzle , primary exit nozzle and bypass duct nozzle.
2. The total pressure ratio's of the main burner , primary exit nozzle , and bypass stream exit nozzle (π_b, π_n *and* π_{fn}) do not change from their reference values.
3. The component efficiencies (η_c, η_f , η_b , η_{tH} , η_{tL}, η_{mH} and η_{mL}) do not change from their reference values.
4. Turbine cooling and leakage effects can be neglected.
5. No power is taken from the turbine to drive accessories (or alternately , η_{mH} or η_{mL} includes the power removed but is still constant .
6. It will be assumed that the gases are calorically perfect up- and downstream of the main burner ,and that γ_t and c_{pt} do not vary with the power setting (T_{t4}).
7. The term unity plus the fuel air ratio (1+f) will be considered as a constant.

Performance of the turbofan engine

The turbofan engines used on commercial subsonic aircraft typically have two spools and separate exhaust nozzles of the convergent type.

The flow in the nozzle will be choked when $\frac{P_{t9}}{P_0} \geq \left[\frac{(\gamma+1)}{2}\right]^{\gamma/(\gamma-1)}$

In case the exhaust nozzle is not choked , the nozzle exit pressure equals the ambient pressure and the exit Mach number is subsonic.

To obtain a choked flow at stations 4 and 4.5 of the high pressure spool during engine operation requires constant values of π_{tH}, τ_{tH} , $\dot{m}_{c4}$ and $\dot{m}_{c4.5}$.

Because the exhaust nozzles have fixed areas, this engine has 4 independent variables M_0,T_0 ,P_0 and T_{t4}

In case both nozzles are unhooked, this will result in 11 dependent variables.

The performance variables and constants are listed in table 7.1

Component	Variables		
	Independent	**Constant or known**	**Dependent**
Engine	M_0, T_0, P_0		$\dot{m}_0, \alpha$
Diffuser		$\pi_d = f(M_0), \tau_d$	
Fan			$\pi_f \tau_f$
High-pressure compressor			π_{cH}, τ_{cH}
Burner	T_{t4}	π_b	
High-pressure turbine		π_{tH}, τ_{tH}	
Low-pressure turbine			π_{tL}, τ_{tL}
Mixer		$\pi_{M\,max}$	$\pi_M, \tau_M, M_6, M_{16}, \alpha', M_{6A}$
Afterburner	T_{t7}	π_{AB}	
Nozzle	$\frac{P_0}{P_9}$ or $\frac{A_9}{A_8}$	π_n, τ_n	
Total number	6		14

Table 7.2. Source: Elements of gas turbine propulsion ,Jack D. Mattingly, McGraw-Hill 1996

Formulas used for engine performance analysis

Mass flow parameter (MFP)

The mass flow parameter is defined as follows

$$MFP \equiv \frac{\dot{m}\sqrt{T_t}}{P_t A} \tag{7.1}$$

where

$$\frac{\dot{m}}{A} = \rho V = \frac{PV}{RT} = \frac{V}{\sqrt{\gamma g_c RT}} \frac{P\sqrt{\gamma g_c}}{\sqrt{RT}} = M\sqrt{\frac{\gamma g_c}{R}} \frac{P}{\sqrt{T}} \tag{7.2}$$

multiplying equation (7.2) by $\sqrt{T_t}/P_t$ gives

$$\frac{\dot{m}\sqrt{T_t}}{P_t A} = M\sqrt{\frac{\gamma g_c}{R}} \frac{P/P_t}{\sqrt{T/T_t}} \tag{7.3}$$

replacing the static/ total property ratios with equations (4.1) and (4.2) results in

$$MFP(M) = \frac{\dot{m}\sqrt{T_t}}{P_t A} = \frac{M\sqrt{\gamma g_c/R}}{\{1+[(\gamma-1)/2]M^2\}^{(\gamma+1)/[2(\gamma-1)]}} \tag{7.4}$$

Low pressure compressor ($\boldsymbol{\tau_{cL}}$, $\boldsymbol{\pi_{cL}}$)

$$\tau_{cL} = 1 + (\tau_f - 1)\frac{\tau_{cLR}-1}{\tau_{fR}-1} \tag{7.5}$$

$$\pi_{cL} = [1 + \eta_{cL}(\tau_{cL} - 1)]^{\gamma_c/(\gamma_c-1)} \tag{7.6}$$

Low pressure turbine ($\boldsymbol{\tau_{tL}}$, $\boldsymbol{\pi_{tL}}$)

$$\tau_{tL} = 1 - \eta_{tL}(1 - \pi_{tL}^{(\gamma_t-1)/\gamma_t}) \tag{7.7}$$

$$\pi_{tL} = \pi_{tLR}\sqrt{\frac{\tau_{tL}}{\tau_{tLR}}}\frac{MFP(M_{9R})}{MFP(M_9)} \qquad (7.8)$$

Bypass ratio α

An expression for the engine bypass ratio can be obtained at any operating condition by relating the mass flow rates of the core and fan streams to their reference values

Engine core stream

$$\dot{m}_C = \frac{\dot{m}_4}{1+f} = \frac{P_{t4}A_4}{\sqrt{T_{t4}}}\frac{MFP(M_4)}{1+f} \qquad (7.9)$$

so

$$\frac{\dot{m}_C}{\dot{m}_{CR}} = \frac{P_{t4}}{P_{t4R}}\sqrt{\frac{T_{t4R}}{T_{t4}}} \qquad (7.10)$$

Engine Fan stream

$$\dot{m}_F = \frac{P_{t19}A_{19}}{\sqrt{T_{t19}}}MFP(M_{19}) \qquad (7.11)$$

so

$$\frac{\dot{m}_F}{\dot{m}_{FR}} = \frac{P_{t19}}{P_{t19R}}\sqrt{\frac{T_{t19R}}{T_{t19}}}\frac{MFP(M_{19})}{MFP(M_{19R})} \qquad (7.12)$$

thus combination of equation (7.9) and (7.11) leads to

$$\alpha = \alpha_R \frac{\pi_{cLR}\pi_{cHR}/\pi_{fR}}{\pi_{cL}\pi_{cH}/\pi_f}\sqrt{\frac{\tau_\lambda/(\tau_r\tau_f)}{[\tau_\lambda/(\tau_r\tau_f)]_R}}\frac{MFP(M_{19})}{MFP(M_{19R})} \qquad (7.13)$$

Engine mass flow

The engine mass flow can be written as follows

$$\dot{m}_0 = \dot{m}_{0R} \frac{1+\alpha}{1+\alpha_R} \frac{P_0 \pi_r \pi_d \pi_{cL} \pi_{cH}}{(P_0 \pi_r \pi_d \pi_{cL} \pi_{cH})_R} \sqrt{\frac{T_{t4R}}{T_{t4}}} \tag{7.14}$$

High pressure compressor ($\boldsymbol{\tau_{cL}}$, $\boldsymbol{\pi_{cL}}$)

$$\tau_{cH} = 1 + \frac{T_{t4}/T_0}{(T_{t4}/T_0)_R} \frac{(\tau_r \tau_{cL})_R}{\tau_r \tau_{cL}} (\tau_{cH} - 1)_R \tag{7.15}$$

$$\pi_{cH} = [1 + \eta_{cH}(\tau_{cH} - 1)]^{\gamma_c/(\gamma_c - 1)} \tag{7.16}$$

Fan ($\boldsymbol{\tau_{cL}}$, $\boldsymbol{\pi_{cL}}$)

The relationship of the power balance between the fan and the LPT is

$$\eta_{mL} \dot{m}_{4.5} Cpt(T_{t4.5} - T_{t5}) = \dot{m}_f Cpc(T_{t13} - T_{t2}) + \dot{m}_c Cpc(T_{t2.5} - T_{t2}) \tag{7.17}$$

Rewriting this equation in terms of temperature ratios ,rearranging the variable and constant terms , and equating the constant values to the reference values gives

$$\frac{\tau_r[(\tau_{cL}-1)+\alpha(\tau_f-1)]}{(T_{t4}/T_0)(1-\tau_{tL})} = \eta_{mL}(1+f)\tau_{tH} = \left\{\frac{\tau_r[(\tau_{cL}-1)+\alpha(\tau_f-1)]}{(T_{t4}/T_0)(1-\tau_{tL})}\right\}_R \tag{7.18}$$

Combination with equation (7.5) results in

$$\tau_f = 1 + (\tau_{fR} - 1)\left[\frac{1-\tau_{tL}}{(1-\tau_{tL})_R} \frac{\tau_\lambda/\tau_r}{(\tau_\lambda/\tau_r)_R} \frac{\tau_{cLR}-1+\alpha_R(\tau_{fR}-1)}{\tau_{cLR}-1+\alpha(\tau_{fR}-1)}\right] \tag{7.19}$$

where

$$\pi_f = \left[1 + (\tau_f + 1)\eta_f\right]^{\frac{\gamma_c}{\gamma_c - 1}} \quad (7.20)$$

Description of the Computer calculated performance for a real turbofan engine using the PERF v 3.11 program

In this part the computer program PERF (Perfomanc cycle analysis) will be used to to determine engine performance in off-design conditions and different flight conditions.

As this is DOS software , the results will be processed with excel to obtain performance charts.

Figure 7.1 shows the PERF interface.

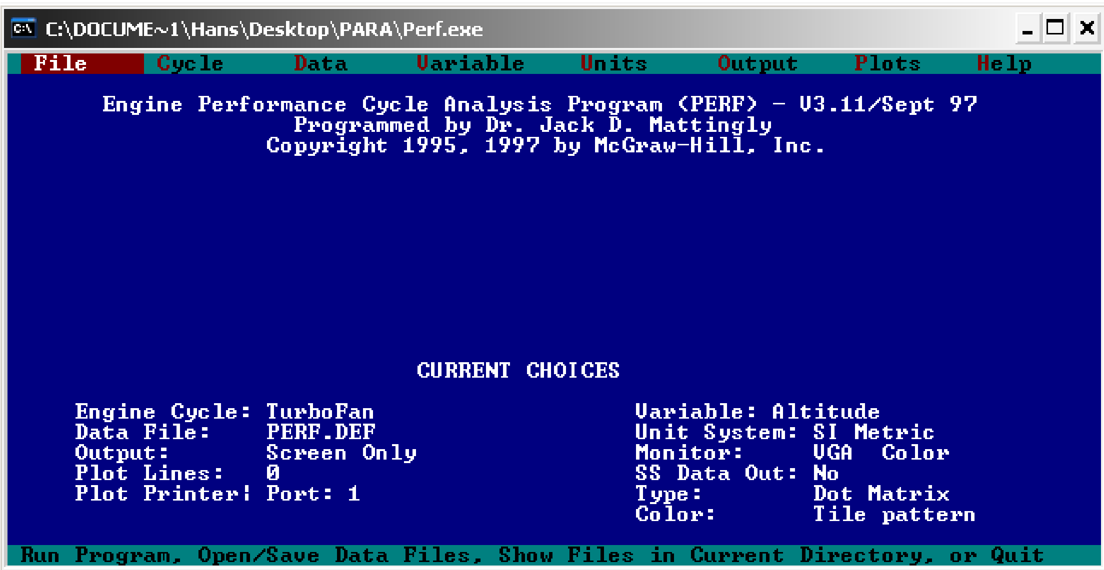


Figure 7.1. PERF interface

Short summary of the PERF menu.

File – permits you to open files and save files,as SSD extension , and the program must have short directory name to work.

Cycle – Permits selection of the engine type and configuration

DATA- allows u to change and view the input data of the engine .Here we select our input data

Variable – allows us to select an iteration variazle

Units – SI or ENG

Calculation of the ideal engine parameters

In this section an overview of the input data used for the PARA program, results and graphs of the results will be presented .The results will be analyzed and compared to the assumptions made for the initial engine.

Input data for the PERF program

Constants:

- Mach number M=0.83
- Altitude H=11000m
- Ambient temperature T= 216.65 K
- Ambient pressure P = 22.632 kPa
- Specific heat with constant pressure in compressor C_{pc}=1.004 $\frac{kJ}{kg \times K}$
- Specific heat with constant pressure in turbine C_{pt}=1,235 $\frac{kJ}{kg \times K}$
- Fan pressure ratio π_f =1.7
- Gas constant gamma compressor γ_c = 1.4
- low heating value of fuel h_{pr} 42 800 $\frac{kJ}{kg}$
- Gas constant gamma compressor γ_t = 1.3
- temperature before entering turbine T_{t4} = 2000 k

- by pass ratio α =10
- mass airflow of 340 kg/s

values obtained from table 5.1,component figures of merit for different levels of technology

- diffuser pressure ratio π_{dmax} , burner pressure ratio π_b ,core nozzle pressure ratio π_n ,bypass nozzle pressure ratio π_{fn}
- polytropic efficiencies ; compressor e_c , turbine e_t ,fan e_f
- isentropic efficiency of the burner η_b
- Mechanical efficiency η_m

Variable:

- Evaluation of altitudes 0 to 11000 m
- Mach numbers 0 to 0.9

```
C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe
                  TurboFan   using   Data File: PERF.DEF
 Flight Condition                        Altitude (m): 0
   Mach #    : .1          T0 (K)    : 288.166   P0 (kPa) : 101.3
 Combustion Exit Temperatures (K) & Fuel Heating Value (kJ/kg)
   Tt4       : 2000
   Fuel Heating Value: 42800
 Gas Properties - Cp's (kJ/kg-K) & gamma's
   Cp c      : 1.004       Cp t      : 1.235
   gamma c   : 1.4         gamma t   : 1.3
 Component Total Pressure Ratios
   Pi D max : .995         Pi B      : .95       Pi N      : .995
   Pi fN     : .98
 Turbomachinery Polytropic Efficiencies
   LPC   : .9       LPT   : .9       HPC   : .9       HPT   : .9       Fan   : .89
 Combustion Efficiencies
   Burner    : .995
 Mechanical Efficiencies
   LP Shaft : .99          HP Shaft : .99
 Exhaust Nozzle Static Pressure Ratio  (0 = Convergent Only Nozzle)
   P0/P9     : 1           P0/P19    : 1
 Design Variables               Mass Flow Rate of Air  (kg/s): 340
   LPC Press Ratio: 1.7       HPC Press Ratio: 20       Fan Press Ratio: 1.7
   Bypass Ratio    : 10       Mach Number @ 2: .5       Mach Number @ 5: .5

 Press <Enter> to continue.
```

```
C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe
 ****************************** RESULTS ********************************
   Tt0/T0    =   1.0020                        Pt0/P0    =   1.0070
   Tt4/T0    =   6.9404                        f         = 0.039435
   Tt25/Tt2  =   1.1835                        Pt25/Pt2  =   1.7000
   Tt3/Tt25  =   2.5884                        Pt3/Pt25  =  20.0000
   Etta cL   =  89.2251 %                      Etta cH   =  85.2164 %
   Tt13/Tt2  =   1.1857                        Pt13/Pt2  =   1.7000
   Etta f    =  88.1479 %
   Tt45/Tt4  =   0.7856                        Pt45/Pt4  =   0.3129
   Tt5/Tt45  =   0.7037                        Pt5/Pt45  =   0.1842
   Etta tH   =  91.1628 %                      Etta tL   =  91.6636 %

 Press <Enter> to continue._
```

```
C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe
 ****************************** RESULTS ********************************
   V19/a0      =   0.8995                      Pt19/P19  =   1.6693
   V9/a0       =   1.7717                      Pt9/P9    =   1.8562
   M9          =  1.01134                      M19       =  0.88787
   A9/A0       = 0.017628                      A9/A8     =   1.0051
   A19/A0      = 0.103727                      A19/A18   =   1.0000
   FR          =   2.1785
   a0          =   340.19 m/s                  Prop Eff  =  17.3194 %
   F/m0        =   301.10 N/(kg/s)             Ther Eff  =  38.5450 %
   S           =   11.906 (mg/s)/N             Over Eff  =   6.6758 %
   Thrust      =   102375 N                    Fuel      =   1.2189 kg/s

 Press <Enter> to continue.
```

C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe

```
 ****************************** RESULTS ********************************
Station      Pt        Tt    Pressure     Temp   Mach  Velocity      Area    Area*
Number     (kPa)      (K)      (kPa)      (K)           (m/s)      (m^2)   (m^2)
  0       102.011    288.7   101.300    288.2  0.1000    34.0     8.1557  1.4009
  2       101.501    288.7    85.567    275.0  0.5000   166.2     1.8864  1.4079
  2.5     172.551    341.7   149.449    328.0  0.4578   166.2     0.1171  0.0819
  13      172.551    342.4   149.490    328.6  0.4574   166.2     1.1730  0.8198
  3      3451.028    884.5  3266.884    870.7  0.2810   166.2     0.0142  0.0066
  4      3278.476   2000.0  1789.155   1739.1  1.0000   802.7     0.0111  0.0111
  4.5    1025.861   1571.2   559.841   1366.3  1.0000   711.5     0.0314  0.0314
  5       188.973   1105.7   161.109   1065.7  0.5000   314.2     0.1928  0.1430
  8       188.973   1105.7   103.128    961.5  1.0000   596.9     0.1430  0.1430
  9       188.028   1105.7   101.300    958.6  1.0113   602.7     0.1438  0.1437
  18      169.100    342.4   101.300    295.7  0.8879   306.0     0.8460  0.8366
  19      169.100    342.4   101.300    295.7  0.8879   306.0     0.8460  0.8366

  Press <Enter> to continue._
```

C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe

```
                        Performance Calculations of
                   TurboFan   using    Data File: PERF.DEF
  Input Constants:
  Pid max= 0.9950    Cp c    = 1.0040    Cp t    = 1.2350    Pi b    = 0.9500
  Etta b = 0.9950    gam c   = 1.4000    gam t   = 1.3000    Pi n    = 0.9950
  Etta cL= 0.8923    Etta cH= 0.8522     Etta mL= 0.9900     Etta mH= 0.9900
  Pi tH  = 0.3129    Tau tH = 0.7856     Etta tL= 0.9166     Pi fn   = 0.9800
  Etta f = 0.8815
  TRatio = 1.0000    pic max=   0.00     Pt3 max=    0.0kPa  Tt3 max=    0.0K
  RPMLmax=   0.0%    RPMHmax=   0.0%     P0/P9   = 1.0000    P0/P19 = 1.0000

  M0 =  0.00    Alt =      0 m     T0 = 288.2 K    P0 = 101.300 kPa
  Tt4 = 1777.8

   Press <Enter> to continue._
```

C:\DOCUME~1\Hans\Desktop\PARA\Perf.exe

I	Mach	Thrust	Air	Fuel	F/m0	S	pic	Alpha	pif	L
0	0.000	85393	297.98	0.8487	286.57	9.939	25.963	10.9867	1.5377	0
1	0.100	76175	299.78	0.8504	254.11	11.164	25.841	11.0310	1.5344	0
2	0.200	68585	305.14	0.8555	224.77	12.474	25.483	11.1615	1.5248	0
3	0.300	62344	314.03	0.8642	198.53	13.861	24.908	11.3703	1.5099	0
4	0.400	57242	326.39	0.8766	175.38	15.314	24.144	11.6446	1.4910	0
5	0.500	53074	342.02	0.8931	155.18	16.827	23.226	11.9650	1.4694	0
6	0.600	49704	360.70	0.9138	137.80	18.385	22.191	12.3089	1.4466	0
7	0.700	47040	382.30	0.9393	123.04	19.967	21.076	12.6569	1.4238	0
8	0.800	44964	407.83	0.9693	110.25	21.556	19.906	13.0368	1.4001	0
9	0.900	43379	438.18	1.0039	99.00	23.143	18.705	13.4655	1.3753	0
10	1.000	42220	473.93	1.0435	89.08	24.716	17.500	13.9413	1.3497	0
11	1.100	40933	514.09	1.0846	79.62	26.497	16.315	14.4646	1.3241	0
12	1.200	39684	559.98	1.1289	70.87	28.447	15.169	15.0359	1.2992	0
13	1.300	38526	612.71	1.1773	62.88	30.559	14.076	15.6533	1.2754	0
14	1.400	37434	673.23	1.2303	55.60	32.866	13.044	16.3147	1.2530	0
15	1.500	36372	742.54	1.2881	48.98	35.415	12.080	17.0176	1.2319	0
16	1.600	35293	821.75	1.3510	42.95	38.280	11.185	17.7588	1.2123	0
17	1.700	34148	912.07	1.4191	37.44	41.559	10.359	18.5347	1.1943	0
18	1.800	32881	1014.79	1.4926	32.40	45.395	9.600	19.3416	1.1777	0
19	1.900	31432	1131.35	1.5716	27.78	50.001	8.907	20.1751	1.1626	0
20	2.000	29734	1263.30	1.6561	23.54	55.697	8.274	21.0308	1.1489	0

Press <Enter> to continue.

Results obtained by the PERF program

As mentioned before, we will use the Excel program to process the results of our calculations . a study is made of the real engine off-design performance in relation to the variation to Mach number and altitude

In previous chapter we calculated that the required thrust should 89386 N calculated for following flight conditions : M=0.83 ,altitude = 11000 m.

From figure 7.4, we can see that the thrust in these conditions will be 10700 N.

As we have 2 engines , The total thrust will be 214000N

.   we must also be able to have a 2.4% climb gradient for single engine performance on takeoff.

At M=0.25 , and at sea level this means the take of speed, we have an engine thrust of about 68000N, enough to satisfy the 2.4% climb gradient.

In next graph , the specific thrust is plotted in function if altitude and Mach number.

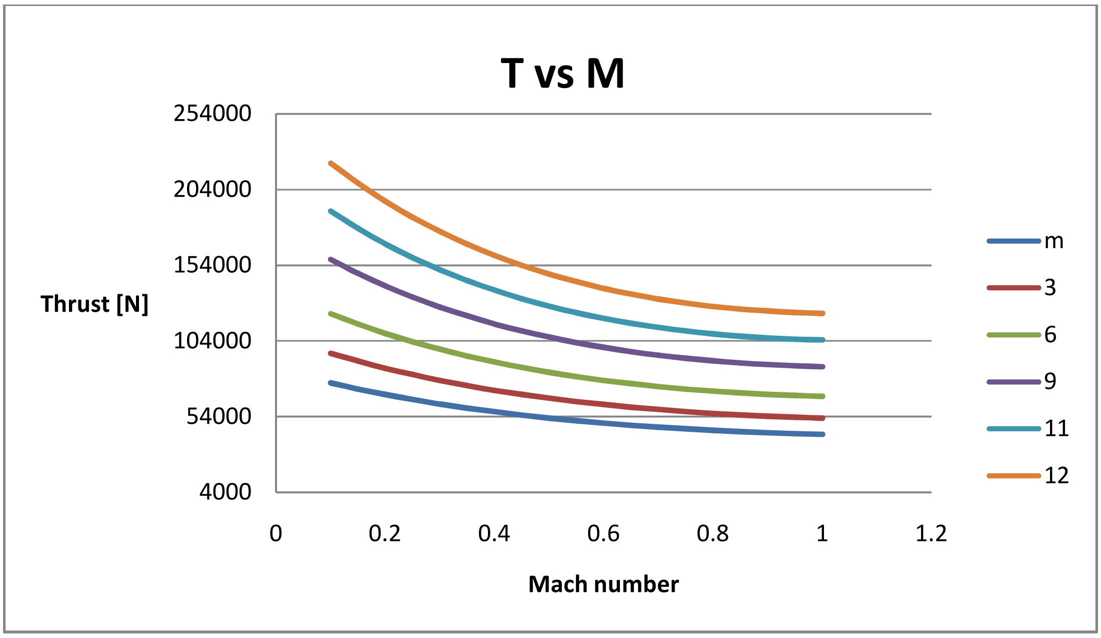


Figure 7. Function if altitude and Mach number

Conclusions.

Designing an engine is not an easy feature !

One most well be aware of all the pitfalls that can occur during calculations, such as mistakes in conversion of units , or calculation and static pressure ratio instead of total pressure ratio.

Throughout this work ,from the start , we made certain assumptions , it is evident that these assumptions must be chosen must be chosen with great care , to obtain plausible results.

In the first part we first took a look at the properties of the aircraft , as the aerodynamic qualities and the weight of the aircraft will determine the primary requirements for designing the engine .

Once we have determined the basic design requirements we can begin with the parametrical cycle analysis.

In order to design our high bypass turbofan engine , We start out with the basic ideal model , with perfect efficiencies , no pressure losses by the components , etc.

Once this basic perfect model is created, we can “mould “ the model ,from perfect to real by adjusting polytrophic efficiencies of the components , which will of course affect overall performance .The quality of the used components is therefore of great importance

A “reference “ situation is created in order to evaluate performance of different engines in the same conditions.

Our parametrical cycle analysis only permits us to analyze performance in on design conditions, but allows us to optimize engine parameters, which will eventually lead to a stable engine that should be able to operate over a wide range of power settings in off-design conditions.

The use of corrected variables is also introduced to make comparisons between different engines easier. This property is used to reduce 5 dependent physical parameters to 2 independent dimensionless or corrected parameters.

Technology plays also a major part in these developments, as new material or cooling techniques allows to further increase the thrust.

The results obtained in engine design should not be the same for a real engine than for an ideal engine, however , it should be in there same trend.

We have also learned that for a set of given parameters, there exists one optimum bypass ratio or one optimum compressor ratio, which corresponds to minimum fuel consumption.

This aspect (and safety) is probably the second most important in aviation industry. If we can optimize the specific fuel consumption, this will lead to a larger range and endurance for the same amount of fuel , which means cost saving and less pollution of the environment.

A thorough background in aerodynamics and thermodynamics is necessary to fully grasp the details of engine design, especially for a correct interpretation for the vast amount of data that influences engine performance.

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