A Mini Project Report On

PARAMETRIC ANALYSIS OF AFTERBURNING TURBOJET ENGINE FOR SMALL-SCALL SUPERSONIC UAV

Submitted To

JAWAHARLAL NEHRU TECHNOLOGICAL UNIVERSITY HYDERABAD

In partial fulfilment of the requirements for the award of the degree

BACHELOR OF TECHNOLOGY

IN

MECHANICAL ENGINEERING

Submitted by

MOHAMMED ABDULLAH GHORI (17RT1A0332)

Under the guidance of

MR. RAZA AHMED KHAN

(Associate Professor)



Department of Mechanical Engineering

NAWAB SHAH ALAM KHAN COLLEGE OF ENGINEERING AND TECHNOLOGY

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CERTIFICATE

This is to certify that the project report entitled **'Parametric Analysis of Afterburning Turbojet Engine for Small Scale Supersonic UAV'** being submitted by

Mohammed Abdullah Ghori **17RT1A0332**

in partial fulfilment for the award of the Degree of Bachelor of Technology in Mechanical Engineering to the Jawaharlal Nehru Technological University, Hyderabad is a record of bonafide work carried out under my guidance and supervision.

The results embodied in this project report have not been submitted to any other University or Institute for the award of any Degree or Diploma.



Mr Raza Ahmed Khan

Associate Professor,
Mechanical Engineering,
NSAKCET



Dr Syed Mujahed Hussaini

Head of the Department
Mechanical Engineering,
NSAKCET

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I express my profound gratitude to Dr Syed Mujahed Hussaini, Head of the Mechanical Engineering Department for being a great source of inspiration for me.

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I thank all the other staff members and friends who have taken time out to help and develop this project.

NOMENCLATURE & SYMBOLS

SYMBOL	UNITS	PARAMETER
W	kg/s	Air Flow Rate
W_F	kg/s	Fuel Flow Rate
CPR		Compressor Pressure Ratio
TPR		Turbine Pressure Ratio
TIT	K	Turbine Inlet Temperature
NER		Nozzle Expansion Ratio
A_9	m^2	Nozzle Exit Area
A*	m^2	Nozzle Throat Area
η_{prop}		Propulsive Efficiency
η_{th}		Thermal Efficiency
η_o		Overall Efficiency
TSFC	g/kg*s	Thrust Specific Fuel Consumption
F_N	kN	Net Thrust
F_G	kN	Gross Thrust
η_C		Compressor Isentropic Efficiency
η_T		Turbine Isentropic Efficiency
η_{poly}		Polytropic Efficiency
η_{mech}		Spool Mechanical Efficiency
W_c	kg/s	Corrected Flow Rate
PR		Pressure Recovery
M_0		Flight Mach No.
M_e		Exhaust Gas Mach No.

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1. INTRODUCTION

Today's aircraft industry is competing to build a highly capable & efficient aircrafts. Countries are racing each other to develop air superiority fighters that would dominate the skies. This led to increased demand for development of powerful air breathing engines required for massive thrusts for supersonic & hypersonic vehicles.

However, Unmanned Combat Aircrafts are the future of military aviation. Eliminating the risk of human life with autonomy, drones could be used in extreme operations. It will hold the capability to break the boundaries of flight speeds & high G-Forces giving the aircraft extreme combat maneuvers. A UAV equipped with supersonic speeds & enhanced maneuvers is believed to be an air superiority aircraft of the future.

As the jet engine technology is advancing, their manufacturing processes are become complex, requiring advanced materials with high thermal hence increasing the costs of development. This research is aimed to explore the possibilities of developing an engine for supersonic UAV by mitigating manufacturing complexities & cutting down of development costs. Engine development is a multidisciplinary process covering thermal design, aerodynamics and structural mechanics, where the requirements of one discipline may very well contradict the requirements of another, and trade-offs between performance, size, weight, and costs continuously need to be made. Engine development requires input from aircraft manufacturers & their development occurs simultaneously for seamless integration.

1.1 DESIGN OBJECTIVE

The primary objective of this project was to develop a parametric design of a single spool, afterburning Turbojet Engine without air bleed system for turbine & nozzle cooling. The engine was designed to produce sufficient thrust, enabling aircraft to reach Mach 2 super-cruise with operating ceiling of 5km. The engine performance is to be optimized for minimum possible TSFC.

This study will estimate maximum thrust, engine would produce to reach Mach 2, without showing significant thermal fatigue on engine turbine & other components.

1.2 WORK SCOPE & LIMITATIONS

The conceptual design of jet engine is limited to thermodynamic cycle analysis of turbojet engine along with parametric studies for Design Point & Off-Design conditions.

2. LITERATURE REVIEW

INTAKE

In [1], author discusses about approach to preliminary design for Supersonic intake & the importance of maximizing Ram Recovery. Higher pressure recovery is necessary for optimal performance of engine particularly at supersonic speed.

Higher number of oblique shocks formation are desirable as they result in better pressure recovery, however they are also accompanied by flow complexities & bulkiness of intake system as it is obtained by deploying multiple variable ramps making intake longer & complex with multiple actuators. The interactions of boundary layer with shocks result in dissatisfactory air flow performance as well. Selection of number of intake ramp depends on the choice of designer. Several selections & tradeoffs are made before deciding the right configuration for the engine.

There are three types of compression methods for Oblique Shock Intake:

- all external oblique shocks
- 2 external oblique shocks, 1 internal oblique shock, and 1 internal normal shock
- 1 external oblique shock, 2 internal oblique shocks, and 1 internal normal shock

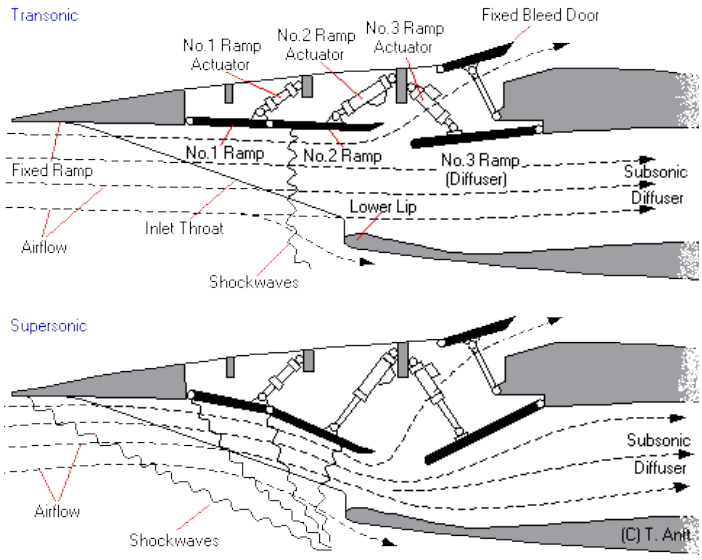


Fig 1: Oblique Shock Intake of F-15 Eagle

is followed by subsonic diffuser that further reduces velocity down to subsonic values to match compressor demand for inlet velocity & air flow rate.

2D Intake configuration is far better & capable than conventional spike inlet as it offers better adaptability over wide range of flow conditions, especially during climb with high angles of attack. It is tailored to give high performance and efficiency for supersonic & subsonic flights as well.

Air velocity reduces as it flows across multiple oblique shocks and reaches Mach 1 at throat which

PROPULSION NOZZLE

Propulsion Nozzle is what differentiates a Gas Turbine Power Plant & a Jet Engine. In the nozzle, gases having high temperature & pressure are expanded to increase the velocity of exhaust gases. It is the difference of flight velocity & exhaust gas velocity that causes momentum change & hence forward thrust.

There are several types of nozzles that can be broadly classified into Supersonic & Subsonic. For aircrafts flying at subsonic velocities, a convergent nozzle is used where maximum exhaust velocity that can be achieved is Mach 1. For supersonic aircrafts that has super-cruise capability use *de Laval* nozzle where a divergent section is added which further expands gases to supersonic velocities. Military aircrafts have sophisticated nozzle systems that must be able to vary throat area & exit area to ensure optimal performance over wide range of flight conditions. Afterburning engines require variable throat area nozzles to accommodate increased flow rates & to avoid nozzle damage.

In [2], author describes the process of selection of different techniques for varying NER. Passively controlled Area Nozzle are used in F-15 Eagle & F-16. For these nozzles, A9 is kinematically tied to A* through a slotted mode strut that allows a range of A9 to exist for a given throat area. The internal pressures acting on the divergent petals exert force necessary to passively adjust the A9 according to exhaust gas velocity & thrust.

This configuration offers multiple advantages, including light weight, design simplicity, and good performance over a wide range of flight conditions.

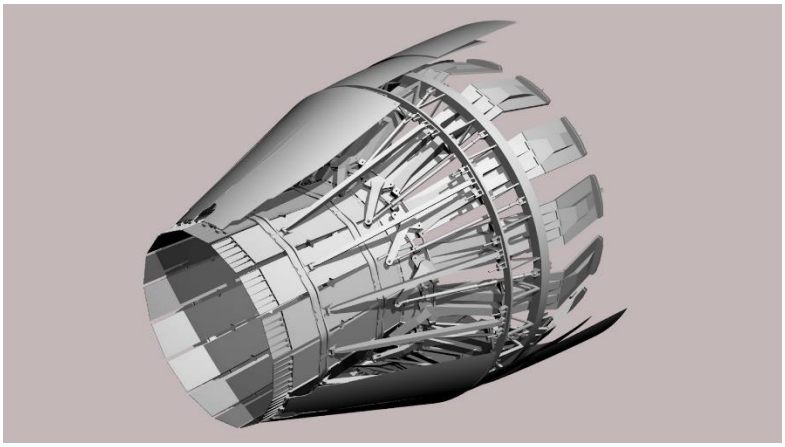


Fig 2: Passively Controlled Area Nozzle

TURBINES

The turbojet engine works by scooping in air & compressing it to maximum engine pressure followed by isobaric combustion process that adds energy to the flow & expanding it to high flow velocities in the nozzle to produce a forward thrust. Scramjet Engines acquire maximum engine pressure by compressing high-speed air across supersonic inlets & diffuser. However, turbojets operate at subsonic velocities as well, where upstream air velocity is less than Mach 1. Therefore, they require an axial compressor to pressurize air before the combustion process. This is achieved by coupling turbines with compressor, that partially expands high temperature exhaust gases from combustion chamber to drive the compressor. Supersonic turbojet engines require higher TITs to produce massive thrust to propel the aircraft & sustain its flight beyond Mach 1.

With the turbine blades subjected to extreme temperatures & pressures, they undergo thermal fatigue and require turbine overhauls. To mitigate thermal stresses & improve engine performance, various turbine blade cooling systems were developed along with advanced metallurgical technologies. Turbine cooling system works by drawing bleed air from the compressor &

distributing it to the turbine stages. The air-cooling methods used in the turbine blade can be divided into the following categories:

1. Convection cooling
2. Impingement cooling
3. Film cooling
4. Full-coverage film cooling.
5. Transpiration cooling.

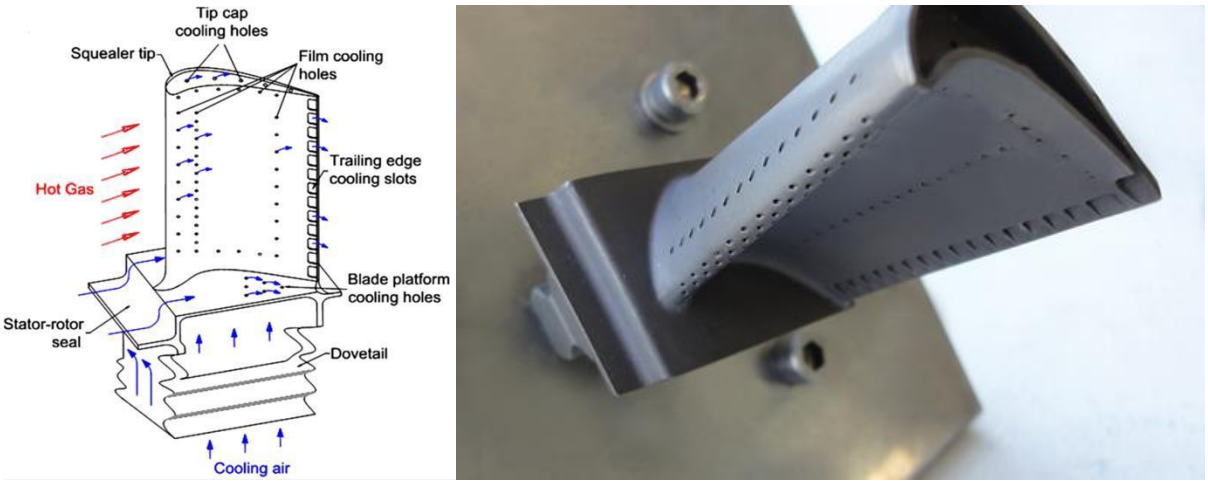


Fig 3: Turbine Section showing cooling air flow

Provided that turbine cooling is essential for mitigation of thermal stresses, they significantly increase the manufacturing complexities & hence costs of production.

A number of different types of efficiencies are used to characterize the performance of an aeroengine:

- **Propulsive efficiency** – the ratio of useful propulsive energy. This is the product of thrust and flight velocity divided by the kinetic energy of the jet.
  Propulsive Efficiency is zero for stationary engine. It increases proportionally with increase in flight velocity & becomes 100% when flight velocity & nozzle exhaust velocity are equal which is considered highly impractical.

- **Thermal efficiency** – the increase of the kinetic energy of the gas stream divided by the energy of the fuel, which is the product of the fuel flow rate and its Heating Value. Thermal efficiency can be improved by increasing OPR of the engine.
- **Overall efficiency** – the ratio of useful work done in overcoming the drag of the airplane, and the energy of the fuel. It is also obtained by product of thermal & propulsive efficiency.

  Higher overall efficiency can ensure high endurance & better fuel economy.
- **TSFC** – The mass flow of burnt fuel per unit of time per unit of output thrust

3. DESIGN OF PROPULSION CYCLE

The first step in the design process is the thermodynamic cycle analysis. The cycle parameters and the performance of the components are selected and evaluated to meet the given specification. This was followed by cycle optimization to get optimal values for TSFC, Compressor Pressure Ratio & engine efficiencies.

Once the thermodynamic design is successfully completed, the aerodynamic design of the turbomachinery and other components begins. Number of stages, engine sizing, spool operating RPM and other aerodynamic parameters can be determined.

This is followed by mechanical design where engine components such as bearings, spool, rotors, casing etc. are designed based on stress analysis. This is final & detailed phase of conceptual design process.

Engine design is highly iterative process i.e. all three phases of engine design are dependent on each other and as the design moves into next phases, optimization of previous phase is required to successfully complete the design according to requirements.

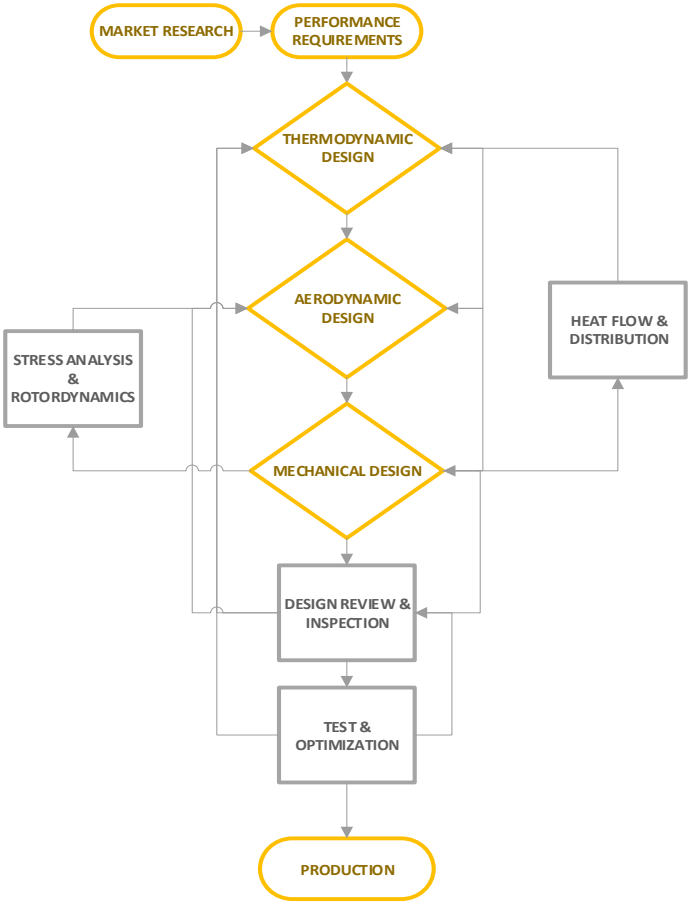
3.1 SOFTWARE DESCRIPTION

For the thermodynamic analysis of the engine, the GasTurb 13 code has been used. The program has a built-in solver for design point & off design analysis with engine modeling.

3.2 DESIGN POINT CALCULATIONS

The design point analysis is where several cycles are analyzed based on design operating conditions to get the required optimal performance. At design point, the geometry of the engine like dimensions, spool RPM etc. are defined which is later analyzed for off design performance.

The design point condition for project is super-cruise at Mach 2 at Altitude 5000m, where the aircraft is expected to spend most of its time. These conditions are used as a Reference point for engine design.



3.3 SELECTION OF DESIGN PARAMETERS

The engine design parameters were selected based on the performance requirement & technological trends which were associated with certain restrictions.

2D Oblique Shock Intake was considered for diffusion of supersonic airflow using single variable ramp with an estimate of 92% inlet pressure recovery. Single Ramp configuration has been selected to minimize weight & complexity of intake system.

With this configuration, a compromise on 1.58kN Net Thrust was done. Although higher TPR of up to 95% is much more suitable for ram recovery at Mach 2 but the tradeoff had to be made

between weight & thrust. Since the engine design is supposed to have minimum weight & complexity, choice of pressure recovery 92% is satisfactory.

Ram Recovery for single ramp intake system is estimated according to Military Specification 5008B.

$$\eta_r = \begin{cases} 1 & M_0 \leq 1 \\ 1 - 0.075(M_0 - 1)^{1.35} & 1 < M_0 < 5 \\ \dfrac{800}{M_0^4 + 935} & 5 < M_0 \end{cases}$$

Based on latest technological trend in turbomachinery, the polytropic efficiency for axial compressor was considered to be 90%. Other design parameters included inlet velocity of Mach 0.5 & compressor diameter of 0.6m. Based on design point conditions corrected flow rate at compressor face was estimated to be 47.14kg/s. CPR was computed based on isentropic relations for which corresponding thermal efficiency was maximum for engine TIT.

$$CPR_{opt} = \left(\frac{TIT}{T_{amb}}\right)^{\frac{\gamma}{2(\gamma-1)}}$$

This value was later optimized for maximum net thrust & minimum SFC.

Annular Combustion Chamber with 97% efficiency is selected with fuel preheat system. The fuel used for analysis was JP-4 with heating value of 43.32MJ/kg.

For this purpose, Thermal coated Inconel 718 was selected for turbine with maximum operating temperature restricted to 1400K. Inconel 718 is a nickel-based super alloy that is well suited for applications requiring high strength in temperature ranges from cryogenic up to 1250K. Isentropic efficiency for turbine was selected as 90%, based on latest technological trends.

A passive control Convergent-Divergent Nozzle with independent throat & exit area variation was considered having exhaust velocity of up to Mach 2.5 corresponding to NAR 2.763, considering ideal expansion to ambient temperature & pressure. However realistic estimation of

thrust was obtained by limiting overexpansion in nozzle. Following isentropic relations were used to estimate nozzle area ratio & pressure ratio.

$$\frac{A}{A^*} = \frac{1}{M}\left[\frac{1+\frac{\gamma-1}{2}M^2}{\frac{\gamma+1}{2}}\right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

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

Ideal nozzle thrust was determined by expanding the flow isentropically to ambient pressure. Dividing the thrust by the product of the total pressure and the effective throat area

$$\frac{F_{ideal}}{PtA_8^*} = \frac{mv_9 + (P_9 - P_{amb})A_9}{PtA_8^*}$$

Optimum performance is achieved at the expansion ratio providing $P_9 = P_{amb}$ referred to as the design expansion ratio.

This type of nozzle configuration was considered due to its light weight, simplicity & excellent performance over wide range of flight conditions.

Maximum temperature for thrust augmentation was limited due to metallurgical constraints for Nozzle & Jetpipe. Since nozzle cooling was not considered, it was safe to consider reheat temperature as 1600K

4. RESULTS

In this report a preliminary design of a Turbojet engine for a Supersonic UAV has been carried out. Based on the performance requirements, the thermodynamic cycle has been optimized and basic sizing of the main components were performed. The result is a more compact, simpler & light weight engine with optimal performance efficiencies.

Thermodynamic design of engine has been successfully completed and it satisfied all performance requirements. According to design & computation data of GasTurb 13, engine will produce net thrust of 29.07kN at design point condition with TSFC of 37.4407 g/kN*s. Thermal efficiency (η_t) has been estimated to 42.02% and Overall Efficiency (η_o) of 31.8% which dropped to 27.93% with active Afterburners.

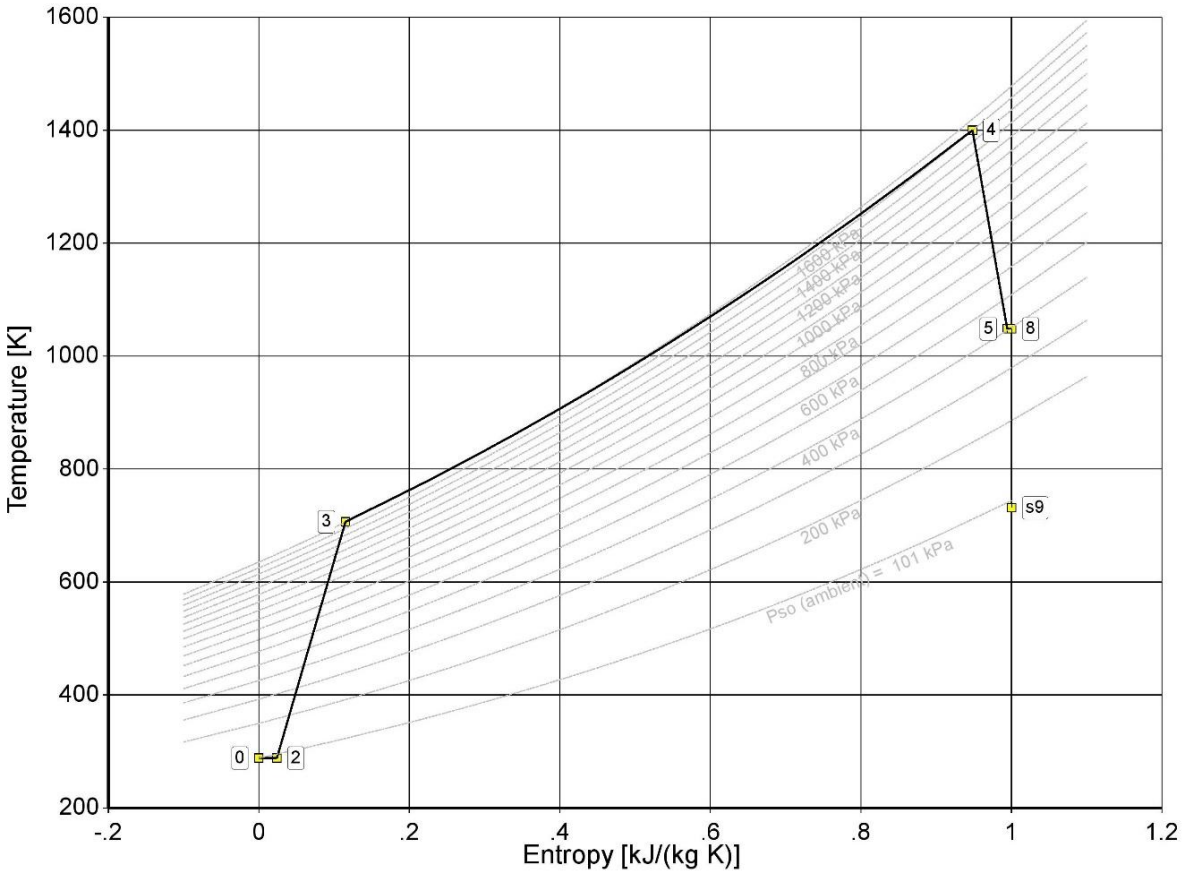
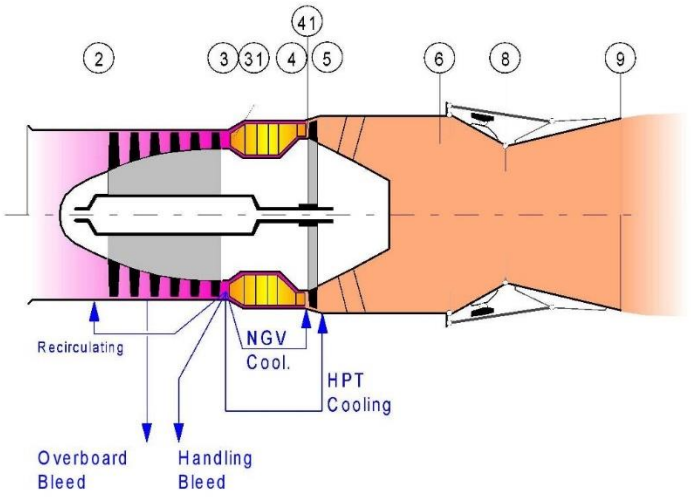


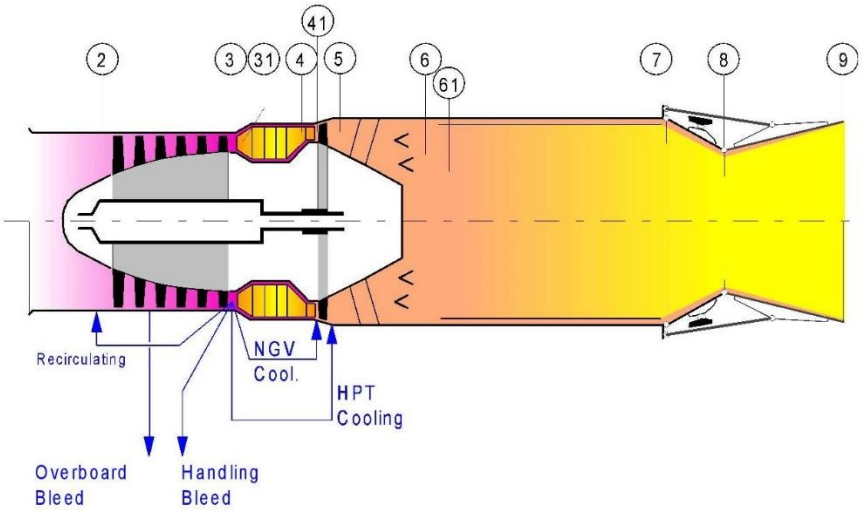
Fig 4: Temperature vs Entropy

	Units	St 2	St 3	St 4	St 5	St 6	St 8	St 9
Mass Flow	kg/s	43.3688	43.3688	44.2447	44.2447	44.2447	44.2447	44.2447
Total Temperature	K	288.15	706.667	1400	1048.23	1048.23	1048.23	1048.23
Static Temperature	K	274.407	675.638	1391.59	1041.58	1041.58	901.394	731.391
Total Pressure	kPa	93.219	1683.96	1633.44	408.392	400.224	400.224	400.224
Static Pressure	kPa	78.5826	1424.94	1591.59	397.786	389.83	216.067	94.7415
Velocity	m/s	166.054	257.456	144.22	125.678	125.678	587.003	854.332
Area	m²	0.26179	0.022927	0.077007	0.26464	0.270041	0.090273	0.114778
Mach Number		0.5	0.5	0.2	0.2	0.2	1	1.60552
Density	kg/m³	0.997637	7.34726	3.98387	1.33029	1.30368	0.83495	0.451208
Spec Heat @ T	J/(kg*K)	1004.52	1076.49	1243.96	1186.98	1186.98	1186.98	1186.98
Spec Heat @ Ts	J/(kg*K)	1004.19	1069.08	1242.81	1185.65	1185.65	1155.51	1113.02
Enthalpy @ T	J/kg	-10032.2	422330	1.25298E6	824895	824895	824895	824895
Enthalpy @ Ts	J/kg	-23819.3	389189	1.24258E6	816997	816997	652608	459953
Entropy Function @ T		-0.11924	3.09217	5.96224	4.7371	4.7371	4.7371	4.7371
Entropy Function @ Ts		-0.290042	2.92515	5.93628	4.71079	4.71079	4.12067	3.29623
Exergy	J/kg	-6896.78	399208	989915	548505	546833	546833	546833
Gas Constant	J/(kg*K)	287.05	287.05	287.087	287.087	287.087	287.087	287.087
Fuel-Air-Ratio		0	0	0.020196	0.020196	0.020196	0.020196	0.020196
Water-Air-Ratio		0	0	0	0	0	0	0



TJetCD.WMF GasTurb

	Units	St 2	St 3	St 4	St 5	St 6	St 8	St 9
Mass Flow	kg/s	43.3688	43.3688	44.2447	44.2447	44.2447	45.0108	45.0108
Total Temperature	K	288.15	706.667	1400	1048.23	1048.23	1548.77	1548.77
Static Temperature	K	274.407	675.638	1391.59	1041.58	1041.58	1353.95	1130.52
Total Pressure	kPa	93.219	1683.96	1633.44	408.392	400.224	395.106	395.106
Static Pressure	kPa	78.5826	1424.94	1591.59	397.786	389.83	216.145	98.1829
Velocity	m/s	166.054	257.456	144.22	125.678	125.678	708.641	1031.43
Area	m²	0.26179	0.022927	0.077007	0.26464	0.270041	0.114238	0.144272
Mach Number		0.5	0.5	0.2	0.2	0.2	1	1.5857
Density	kg/m³	0.997637	7.34726	3.98387	1.33029	1.30368	0.556007	0.302478
Spec Heat @ T	J/(kg*K)	1004.52	1076.49	1243.96	1186.98	1186.98	1301.49	1301.49
Spec Heat @ Ts	J/(kg*K)	1004.19	1069.08	1242.81	1185.65	1185.65	1275.2	1236.67
Enthalpy @ T	J/kg	-10032.2	422330	1.25298E6	824895	824895	1.47671E6	1.47671E6
Enthalpy @ Ts	J/kg	-23819.3	389189	1.24258E6	816997	816997	1.22562E6	944782
Entropy Function @ T		-0.11924	3.09217	5.96224	4.7371	4.7371	6.55402	6.55402
Entropy Function @ Ts		-0.290042	2.92515	5.93628	4.71079	4.71079	5.95082	5.1617
Exergy	J/kg	-6896.78	399208	989915	548505	546833	1.04724E6	1.04724E6
Gas Constant	J/(kg*K)	287.05	287.05	287.087	287.087	287.087	287.119	287.119
Fuel-Air-Ratio		0	0	0.020196	0.020196	0.020196	0.037862	0.037862
Water-Air-Ratio		0	0	0	0	0	0	0



TJetRHCD.WMF GasTurb

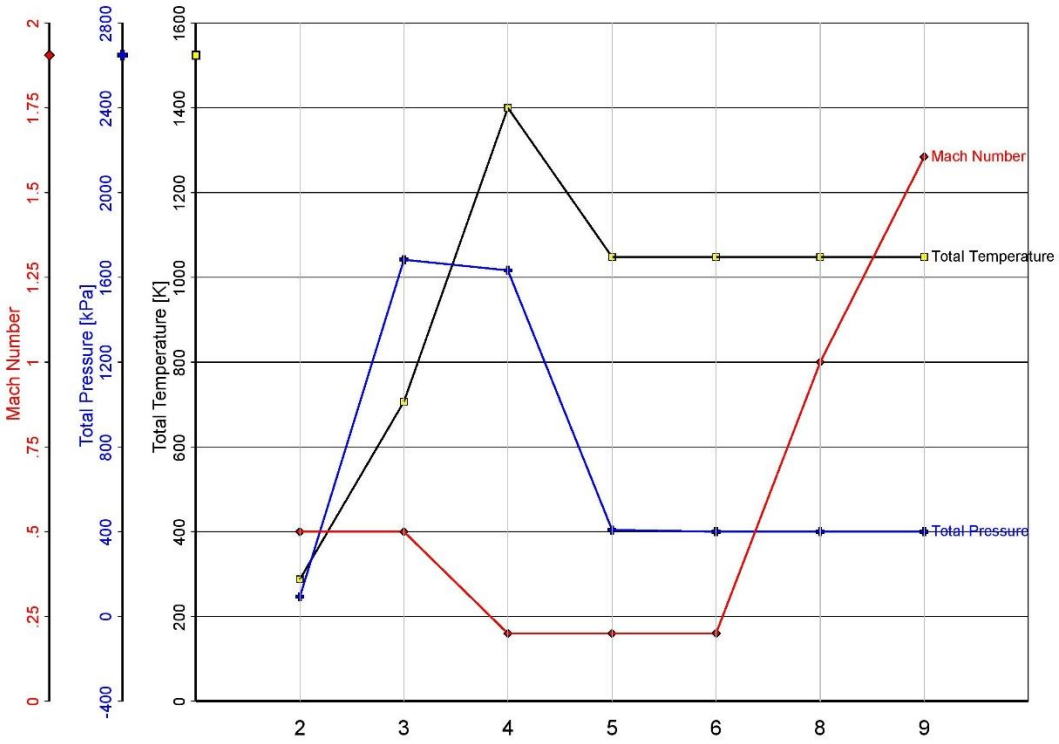


Fig 5: Engine parameters across engine stages

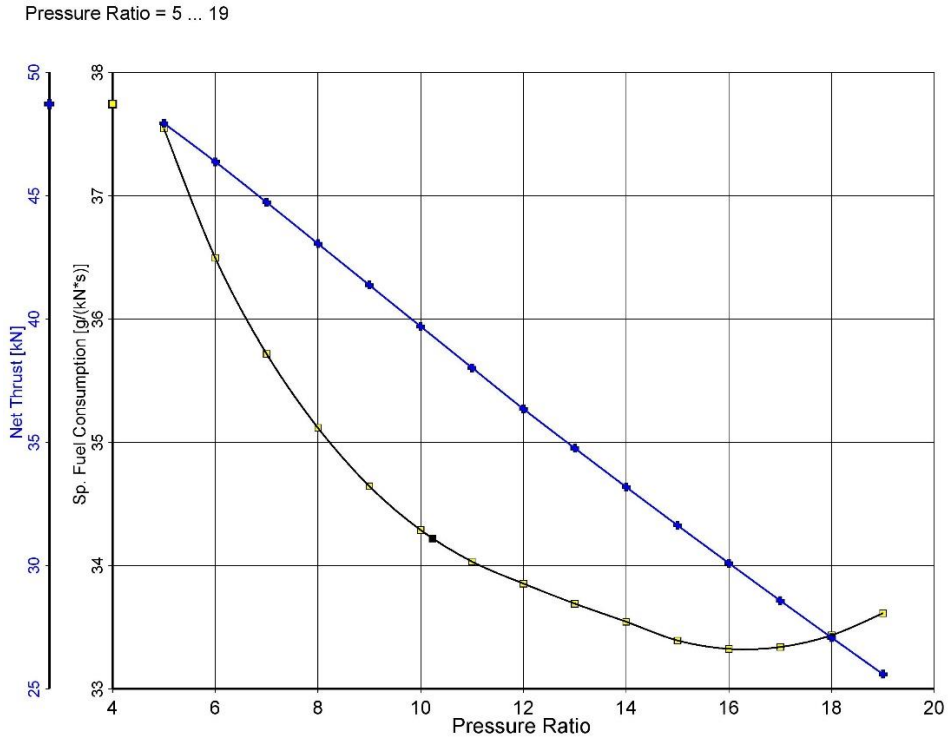


Fig 6: Net Thrust vs TSFC vs Pressure ratio

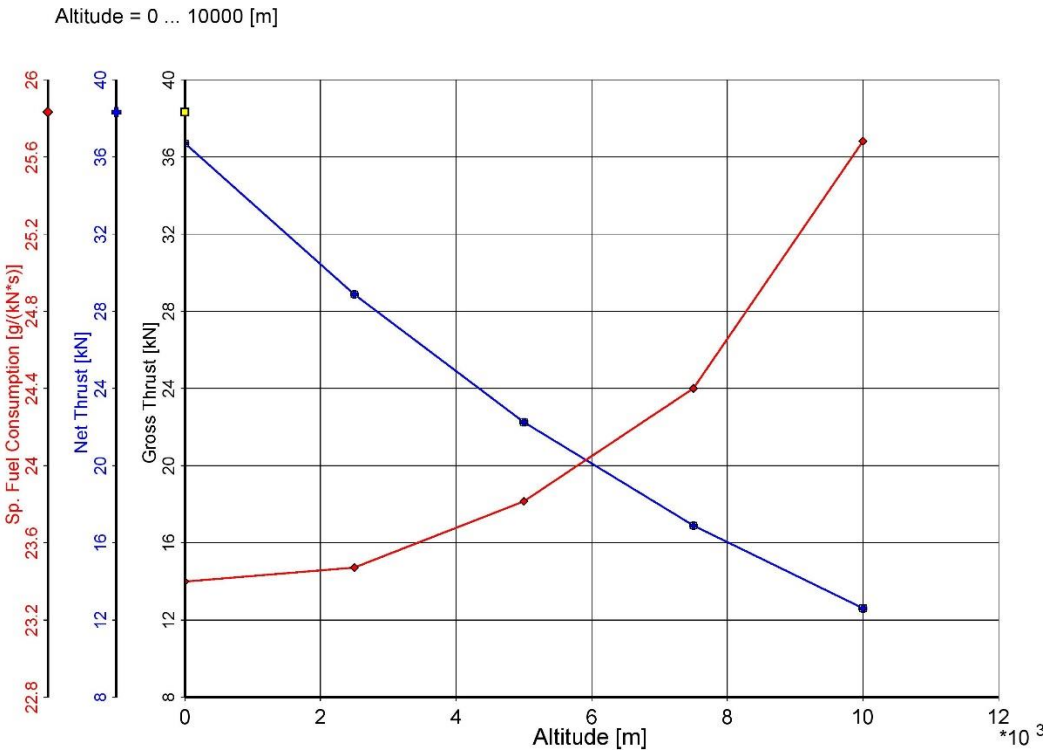


Fig 7: Engine Performance variation with Altitude

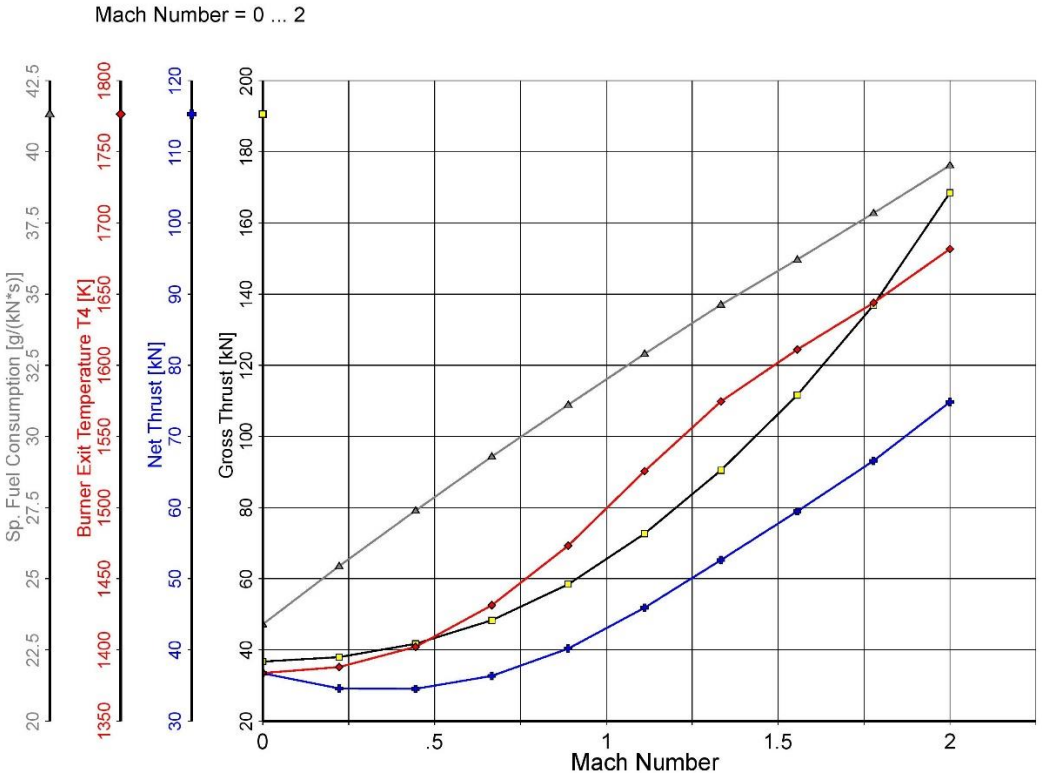


Fig 8: Engine Performance variation with Flight Mach Number

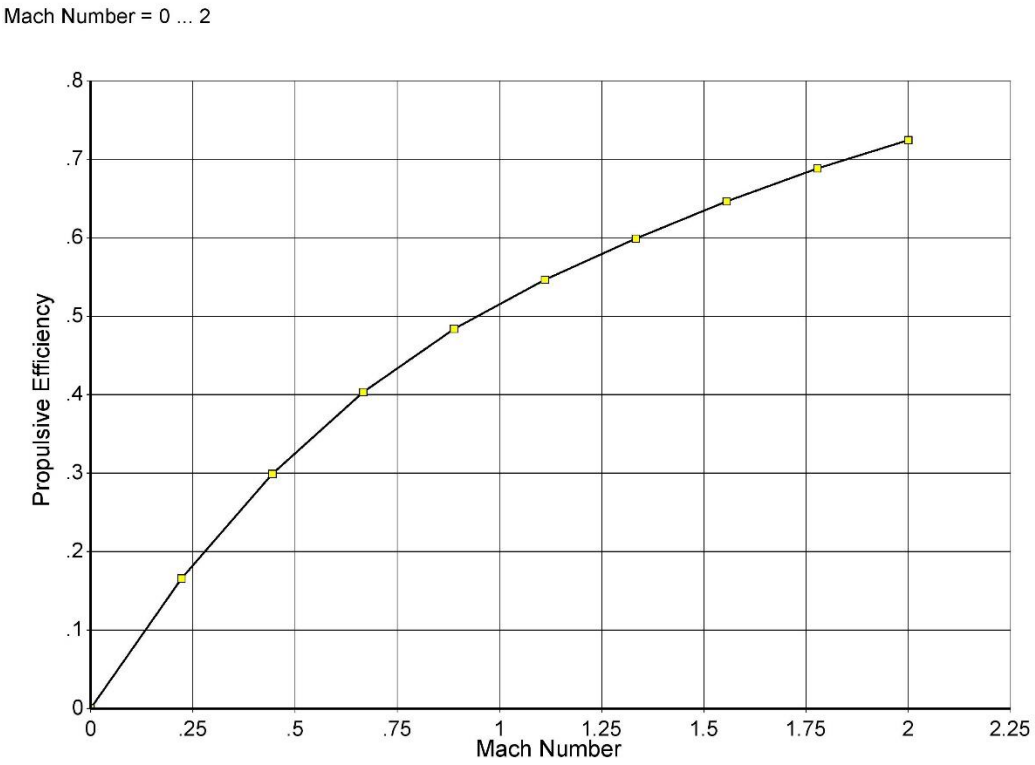


Fig 9: Propulsive Efficiency vs Flight Mach Number

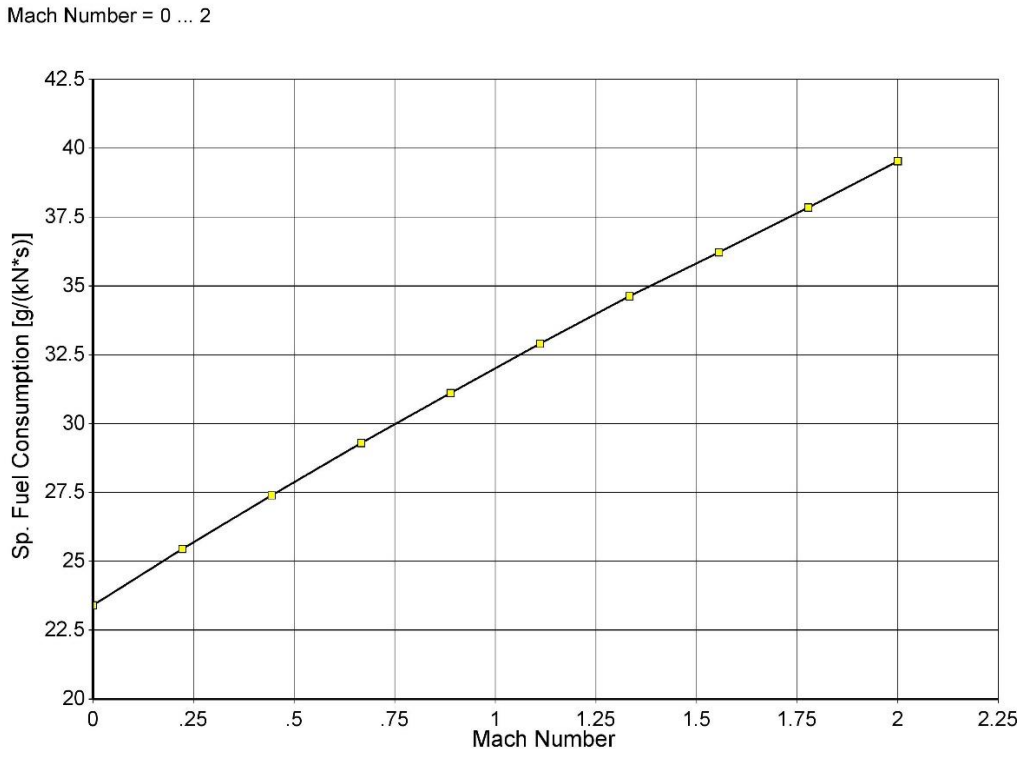


Fig 10: TSFC vs Flight Mach Number

5. CONCLUSION

The Jet Engine Design is a complex process which involve many different disciplines and demands firm understanding of concepts. There is no obvious solution as improvement of one parameter often comes at the expense of another.

Since the project is an independent initiative, therefore to cover all parts of the design process without supervision of propulsion experts may not be possible. A course going deeper in the theory of gas turbine technology, with a greater focus on the advanced metallurgical technology, could however be developed to provide a better direction for improvisation on a project like this.

6. FUTURE WORK

Only parametric analysis has been done and remaining phases are necessary to complete the conceptual design of engine. Further component analysis & simulations are to be performed for accurate prediction of performance over various flight conditions. Aerodynamic design including compressor & turbine stage estimation, component geometry & dimensions, burner design is to be completed. Mechanical design that includes weight estimations and stress calculations particularly the life period of Inconel 718 under thermal stress is to be estimated accurately.

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