Aircraft Engine Design Project

Abhijit Venkat (22110008), Aditya Prasad (22110018), Hari Balaji (22110092), Mutyala Pardha Sai (23210062)

Department of Mechanical Engineering, IIT Gandhinagar

Instructor: Prof. Dilip S. Sundaram

Contents

1 Introduction I
2 Part A: Mission Analysis I
2.1 Problem Statement and Given Information I
2.2 Mission Information I
2.3 Calculations for Thrust and Fuel Burn for the Mission I
Taxi • Takeoff • Climb and Acceleration • Cruise • Descent • Loiter • Landing and Taxi
3 Part B: Parametric Cycle Analysis III
4 Part C: Engine Performance Analysis IV
4.1 Engine Failure Analysis V
5 Results and Discussions VI
5.1 Calculation of Ticket Cost VI
6 Conclusion VI
7 Acknowledgements VII
References VII

1. Introduction

Aircraft engine design stands at the forefront of aerospace engineering, embodying the marriage of innovation and efficiency to propel aircraft through the skies. In modern aviation, the quest for more efficient engines is paramount, driven by the imperative to enhance fuel economy, reduce emissions, and extend operational range. Particularly in specific missions, such as long-range flights or high-speed transport, the imperative for efficiency becomes even more pronounced, as every incremental gain in engine performance translates into tangible benefits in fuel savings, environmental sustainability, and mission success. Thus, pursuing more efficient engines represents a technical challenge and a strategic imperative for advancing aerospace technology.

2. Part A: Mission Analysis

2.1. Problem Statement and Given Information

A request for proposal (RFP) has been published by an aircraft manufacturer inviting proposals from gas turbine engine manufacturers to supply gas turbine engines for a specified modern civilian aircraft. Imagine that your team is working for a specific engine manufacturer, such as Pratt & Whitney, General Electric. We are required to design a gas-turbine engine for the civilian aircraft given below:

$$C_D = K_1 C_L^2 + K_2 C_L + C_{D0}$$

Other given information:

- Cruise speed= $0.83M_0$
- Pressure altitude= 1.6km
- Runway length= 3650m
- Number of passengers= 253, with 90 kg
- Still air distance to be travelled= 11,120 m
- Fuel reserve at the end= 30 min
- Initial altitude attained= 11km
- Maximum diameter of the engine inlet= 2.2m
- Single-engine climb gradient= 2.4%

Maximum Gross Takeoff Weight	1,645,760 N
Empty Weight	822,880 N
Maximum Landing Weight	1,356,640 N
Maximum Payload	420,780 N (253 passengers + 196,000 N of cargo)
Maximum Fuel Capacity	716,706 N
Wing Area	282.5 m^2
Engine Needed	Turbofan engine (two engines per aircraft)
Maximum Lift Coefficient, $C_{L,max}$	2.0

Table 1. Aircraft Characteristics

M_0	K_1	K_2	C_{D0}
0.00	0.056	-0.004	0.0140
0.40	0.056	-0.004	0.0140
0.75	0.056	-0.008	0.0140
0.83	0.056	-0.008	0.0150

Table 2. Drag Coefficients

2.2. Mission Information

Phase	**Distance (km)**
Taxi	1.5
Takeoff	3.65
Climb and Acceleration	330
Cruise	10,650
Descent	140
Loiter (30 min at 9 km altitude)	-
Land and taxi	1.5
Total	11,120

Table 3. Distances traveled in various phases of flight

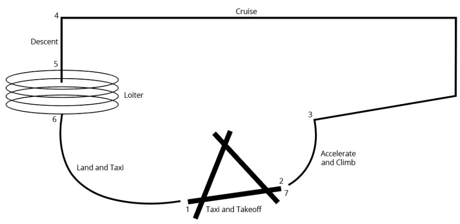


Figure 1. Mission Phases

2.3. Calculations for Thrust and Fuel Burn for the Mission

To calculate the thrust requirements, we have chosen a value of TSFC of an aircraft that can perform similar missions. We considered the Boeing 777 as our reference aircraft. Thus, the TSFC we assumed was 16 g/kN.s.

2.3.1. Taxi

We assume that the aircraft taxis at a constant velocity, therefore we can say that $T = D$.

$$v_{taxi} = M_{taxi} a_0 \tag{1}$$

where $M_{taxi} = 0.05$.

$$t_{taxi} = \frac{d_{taxi}}{v_{taxi}} \tag{2}$$

where $d_{taxi} = 1.5$ km.

$$T_{taxi} = \frac{1}{2} \rho_{grd} A v_{taxi}^2 C_{D,taxi} \tag{3}$$

where $C_{D,taxi} = K_1 C_{L,max}^2 + K_2 C_{L,max} + C_{D0}$. The values of K_1, K_2, $C_{L,max}$ and C_{D0} are taken from table 2 according to the mach number.
To calculate the mass of the fuel burnt during the taxi,

$$m_{f,taxi} = TSFC \cdot T_{taxi} \cdot t_{taxi} \tag{4}$$

2.3.2. Takeoff

We are assuming that the aircraft accelerates with a constant acceleration.
For the wings to generate just enough to counter its weight, the aircraft's velocity should be 1.2 times it stall velocity, i.e. $v_{to} = 1.2 \cdot v_{stall}$.
It is also assumed that the entire runway length is used for acceleration, i.e., 3650 m. Here,

$$v_{stall} = \sqrt{\frac{W_0}{\frac{1}{2} \rho_{grd} A C_{L,max}}} \tag{5}$$

We now calculate the acceleration during takeoff (assuming that the plane starts from rest before it begins accelerating on the runway):

$$a_{to} = \frac{v_{stall}^2 - 0^2}{2 d_{to}} \tag{6}$$

We then apply a simple force balance to calculate the thrust during takeoff:

$$T_{to} = m_{to} a_{to} + D_{to} \tag{7}$$

Here,

$$D_{to} = \frac{1}{2} \rho_1 A v_{to}^2 C_{D,to} \tag{8}$$

and the values for $C_{D,to}$ are obtained using table 2.
To find the time required for takeoff, we apply the following equation:

$$t_{to} = \frac{1.2 v_{stall}}{a_{to}} \tag{9}$$

We now calculate the mass of fuel burnt during takeoff:

$$m_{f,to} = TSFC \cdot T_{to} \cdot t_{to} \tag{10}$$

2.3.3. Climb and Acceleration

It is given that we must reach the operating cruise altitude of 11 km within a horizontal displacement of 330 km. Using trigonometry, this gives us a climb gradient of 2.84%.
The final climb velocity is $v_{f,climb} = M_0 a_0$, where M_0 & a_0 are the Mach number and speed of sound at cruise.
We apply the equations of motion once more to get:

$$a_{climb} = \frac{v_{f,climb}^2 - v_{to}^2}{2 d_{climb}} \tag{11}$$

where $d_{climb} = 330$ km

$$t_{climb} = \frac{v_{climb} - v_{to}}{a_{climb}} \tag{12}$$

For calculating thrust, we have used a slightly modified equation as shown below:

$$T_{climb} = D_{climb} + W_{climb} \sin\theta + m \cdot a \tag{13}$$

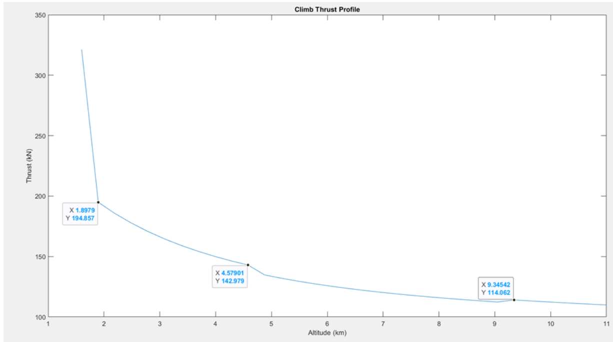


Figure 2. Climb Thrust vs. Altitude

Here, D_{climb} was calculated in the same way as15. W_{climb} is the updated weight of the aircraft after the taxi and takeoff phases.
We now calculate the mass of fuel burnt during the climb:

$$m_{f,climb} = TSFC \cdot T_{climb} \cdot t_{climb} \tag{14}$$

2.3.4. Cruise

During a cruise, the cruise travels at a constant velocity. Therefore, we can say that $T_{cr} = D_{cr}$
Also, the cruise velocity (v_{cr}) is equal to $0.83 a_0$. Therefore, the thrust during the cruise is:

$$T_{cr} = D_{cr} = \frac{1}{2} \rho_1 A v_{cr}^2 C_{D,cr} \tag{15}$$

Here, D_{cr} was calculated in the same way as15.
Now, calculating the time taken for the cruise,

$$t_{cr} = \frac{d_{cr}}{v_{cr}} \tag{16}$$

Calculating the mass of fuel burnt during the climb,

$$m_{f,climb} = TSFC \cdot T_{cr} \cdot t_{cr} \tag{17}$$

Since the cruise is the cruise of the flight, which has the most significant impact on fuel consumption and travel time, we need to calculate the aircraft's weight before and after the cruise phase.

$$W_{i,cr} = W_0 - (m_{f,to} - m_{f,climb}) g \tag{18}$$

$$W_{f,cr} = W_{i,cr} - m_{f,cr} g \tag{19}$$

Calculating the lift coefficient for the cruise,

$$C_{L,cr} = \frac{W_{i,cr}}{\frac{1}{2} \rho_{cr} A v_{cr}^2} \tag{20}$$

Calculating the mass outflow during the cruise,

$$\dot{m}_{cr} = 2 \rho_{cr} A v_{cr} \tag{21}$$

Calculating the specific thrust during the cruise,

$$ST_{cr} = \frac{T_{cr}}{\dot{m}_{cr}} \tag{22}$$

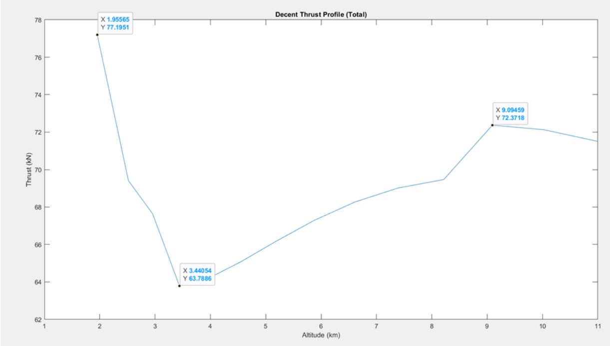


Figure 3. Descent Thrust vs. Altitude

2.3.5. Descent

For the descent segment, we first update the aircraft's weight by subtracting the weight of fuel lost in all the previous segments. This weight at the start of the descent is given by W_{dt}.
It is also given that the aircraft must descend by 2 km while traveling 140 km horizontally. This gives us an angle of descent of θ.

$$\theta = tan^{-1}(\frac{9.4}{140}) \tag{23}$$

Also, we assume that descent occurs at the same Mach Number/velocity since temperature does not vary much at that altitude.

$$t_{dt} = \frac{d_{dt}}{0.83a_0} \tag{24}$$

where, $d_{dt} = 140$ km, and a_0 is the speed of sound at 9km altitude. Now, to find the Thrust requirement, we apply a simple force balance. Since velocity is constant,

$$L - W = m_{aircraft} \cdot a_y \tag{25}$$

$$T - D = m_{aircraft} \cdot a_x \tag{26}$$

We know the initial and final velocities of the aircraft, and the distances, therefore we can find the net deceleration for descent. a_x and a_y are the resolved components of this net acceleration. Finally, using the thrust and time requirements, we find the fuel mass burnt during the descent phase.

$$m_{f,dt} = TSFC \cdot T_{dt} \cdot t_{dt} \tag{27}$$

2.3.6. Loiter

For the loitering phase, we have assumed that the aircraft travels at its cruising Mach number of 0.83. However, since the altitude drops, we take $\rho_{loiter} = 0.43 \frac{kg}{m^3}$. It is given that the aircraft loiters for 30 minutes, therefore $t_{Loiter} = 1800$ s.
Also, since the velocity is constant, Thrust must be equal to the drag force.

$$T_{Loiter} = D_{Loiter} = \frac{1}{2}\rho_{Loiter} A v_{cr}^2 C_{D,cr} \tag{28}$$

Now, the mass of fuel lost during loitering is given by:

$$m_{f,Loiter} = TSFC \cdot T_{Loiter} \cdot t_{Loiter} \tag{29}$$

2.3.7. Landing and Taxi

We now approach the final phase of our mission, which is the landing and deceleration. We analyze this in the same way as we did for takeoff. Firstly, we update the aircraft's weight by subtracting all the weights of fuel lost in the previous segments. The initial velocity before landing is taken to be $1.15v_{stall}$, and the final velocity is 0.
We assume that the entire length of the runway is used for landing. From this information, we can find out the aircraft's deceleration and hence the Reverse Thrust required for landing.

$$a_{landing} = \frac{1.15v_{stall}^2}{2 \cdot d_{runway}} \tag{30}$$

$$-T_{landing} - D_{landing} = -m \cdot a_{landing} \tag{31}$$

$$t_{landing} = \frac{1.15v_{stall}}{a_{landing}} \tag{32}$$

Similar to the above segments, we calculate the mass of fuel burnt using:

$$m_{f,landing} = TSFC \cdot T_{landing} \cdot t_{landing} \tag{33}$$

For the taxi segment after landing, we assume it is the same as the one just before takeoff. Therefore, we can say that all the thrust requirements and mass of fuel burnt are the same for both the taxi segments.

After calculating the thrust requirements for various sections of the mission using this TSFC, we obtain the values for fuel burn and the thrust requirement for each of the sections:

Phase	Distance (km)	Thrust Requirement (kN)	Fuel Burn (kg)
Taxi	1.5	10	13
Takeoff	3.65	333	418
Climb	330	333	5
Cruise	10650	80	53355
Descent	140	78	899
Loiter	-	70	2008.5
Land	3.65	70	223
Taxi	1.5	10	13
Total	11,120		60,961

Table 4. Thrust Requirements and Fuel Burn for the Different Mission Phases

3. Part B: Parametric Cycle Analysis

We now have to perform a Parametric Cycle Analysis to arrive at the optimum parameters for the engine design. The design point we chose was the cruise segment since the aircraft spends most of its time on Cruise. This corresponds to the 0.83 March, 11 km altitude cruise condition. The primary objective is to obtain the ranges of compressor pressure ratios, bypass ratios, etc., that best meet the design requirements. We have assumed $T_{t4} = 1560$ K and $h_{PR} =$ 42,800 kJ/kg for the analysis.

To arrive at possible ranges of design variables, we must first determine the minimum uninstalled specific thrust and maximum allowable uninstalled thrust specific fuel consumption. The minimum specific thrust corresponds to the maximum flow rate achievable for the given maximum engine inlet size. In contrast, the maximum specific fuel consumption results from the mission analysis performed.

We also plot the TSFC vs. Specific Thrust graph while varying the compressor pressure and bypass ratios. The resulting graph is a carpet plot for the design point, i.e., the cruise condition. We then choose our points on the graph, which can be probable engine designs. Later, we analyze each engine in the third section of the project and iteratively perform PCA to arrive at the best engine point.

We obtained The first carpet plot for $\pi_f = 1.65$. The maximum flow rate for cruise conditions is 327.7 kg/s. Also, the minimum specific thrust required during cruise is 122 N s/kg.

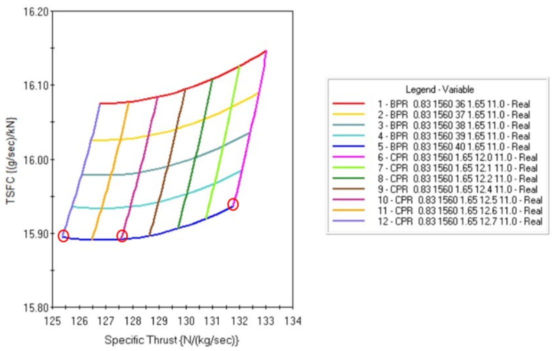


Figure 4. Carpet plot for $\pi_f = 1.65$

In the above plot, the bypass ratio (α) varies from 12 to 12.7, while the compressor pressure ratio (π_c) varies from 36 to 40. The nearly vertical lines correspond to decreasing α while going from left to right. The nearly horizontal lines increase π_c from top to bottom. We tested the marked engine design points

However, the engine with these parameters (at lower TSFC) is larger than required. So, at this TSFC, The engine constraints are unmet (PERF (Performance Analysis) results). The larger bypass ratio may also cause an increase in engine size while reducing the thrust. From these results, we decide to iterate the results with a reduced bypass ratio and increased TSFC to meet engine size constraints. We then plot the carpet plot corresponding to $\pi_f = 2$, while the maximum flow rate during cruise and specific thrust remain the same.
Due to the reduced bypass ratio, the TSFC was increased. To minimize the increase in TSFC, the fan pressure ratio was increased from 1.65 to 2.

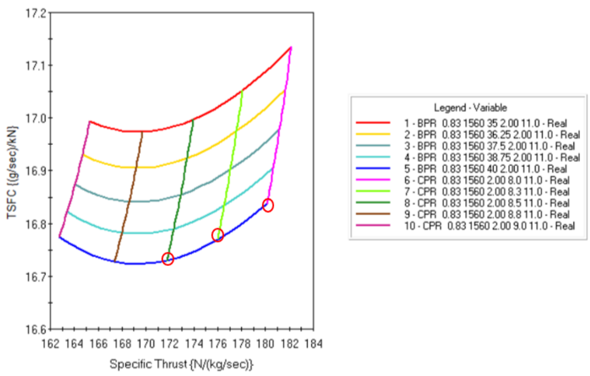


Figure 5. Carpet plot for $\pi_f = 2$

As we can see from the above graph, the Specific Thrust has increased. This particular graph is obtained by varying the bypass ratio (α) (vertical lines) from 8 to 9 while varying the compressor pressure ratio (π_c) from 35 to 40.

We then choose three engine design points corresponding to $\alpha =$ 8.5, 8.3, and 8, on the bottom-most curve corresponding to $\pi_c = 40$.

4. Part C: Engine Performance Analysis

Now, we perform the Engine performance Analysis on PERF for all three optimum engines to see if the engines meet the performance requirements for all the flight segments. Firstly, we conducted PERF analysis for the engines with bypass ratio (α)= 8.5 and 8.3, however those engines had failed for climb and engine failure operations. We take the engine with bypass ratio (α)= 8, compressor pressure ratio (π_c)= 40 and fan pressure ratio (π_f)= 2. In PERF, we set the reference engine at the takeoff. Here, we assume the combustor temperature to be 1890 K, which is higher than the temperature at cruise. This is because takeoff is for a relatively shorter duration and the turbine blades can withstand these high temperatures for a short time. This higher value of T_{t4} gives us more thrust. We observe the takeoff thrust to be 234 kN per engine, which easily comes out to be greater than the required value of 167 kN per engine.

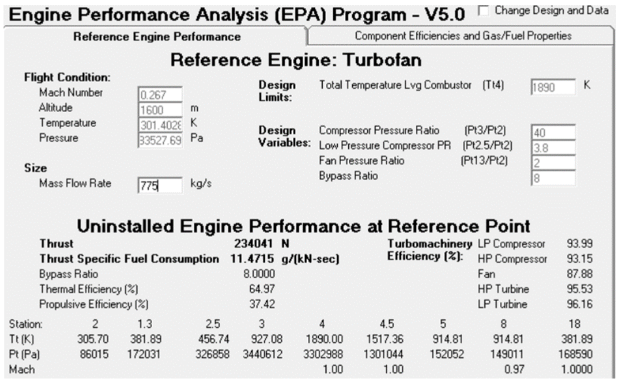


Figure 6. Reference Engine

We test this reference engine for cruise to see if the engine size is met.
We test this reference engine for all the sub-segments of the climb at various combinations of altitudes and corresponding Mach number values. We use the Thrust vs. altitude during the climb from PARA software.

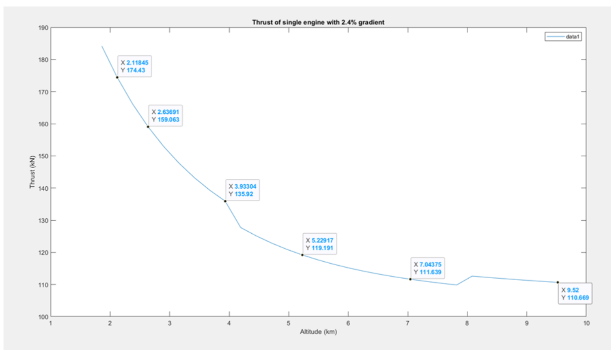


Figure 7. Thrust Vs Altitude

1. **1.89 km altitude:**

   - Required Thrust= 195 kN
   - Mach number= 0.3

   We get the single-engine thrust value to be 122 kN against the requirement of 97 kN. Hence, the engine passes the test.

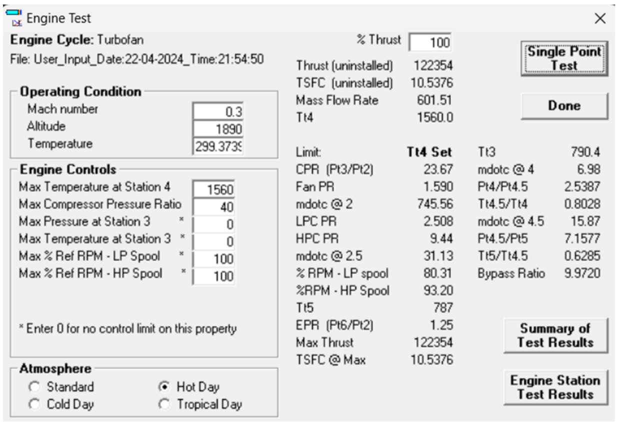


Figure 8. Engine Test for an Altitude of 1.89 km

2. **4.58 km altitude:**

   - Required Thrust= 143 kN
   - Mach number = 0.45

   We get the single-engine thrust value to be 94 kN against the requirement of 72 kN. Hence, the engine passes the test.

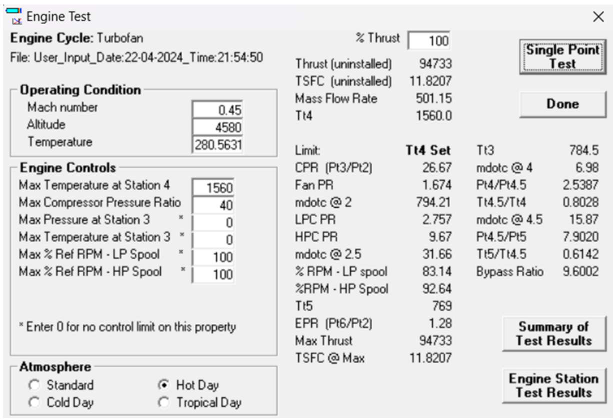


Figure 9. Engine Test for an Altitude of 4.58 km

3. **9.34 km altitude:**

   - Required Thrust= 112 kN
   - Mach number= 0.75

   We get the single-engine thrust value to be 61 kN against the required 56 kN. Hence, the engine passes the test.

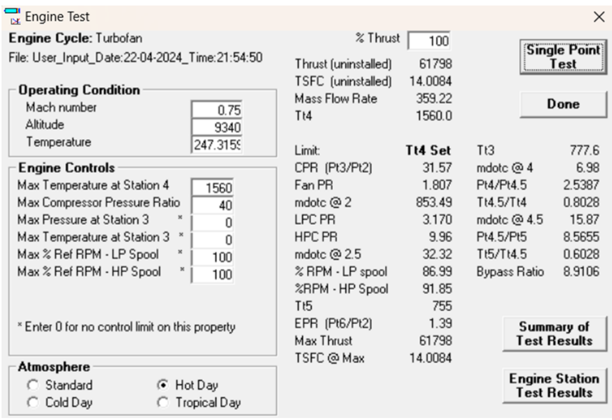


Figure 10. Engine Test for an Altitude of 9.34 km

We test this engine at cruise where the thrust value comes out to be 53 kN. This is comfortably higher than the single-engine thrust requirement of 40 kN at cruise.

Also, we can clearly see that from fig. 11, the mass flow rate during cruise comes Cruise be 313.6 kg/s.

When we apply $m_{dot} = \rho \cdot A \cdot v$, we can find the area of the inlet because we know the density and velocity at cruise conditions.

This in turn gives us the diameter of the engine, which comes out to be 2.1 m from the above calculation.

This is smaller than the maximum allowable engine diameter, which is 2.2m. Therefore, our engine meets the size constraints, while taking care of the takeoff and cruise constraints as well.

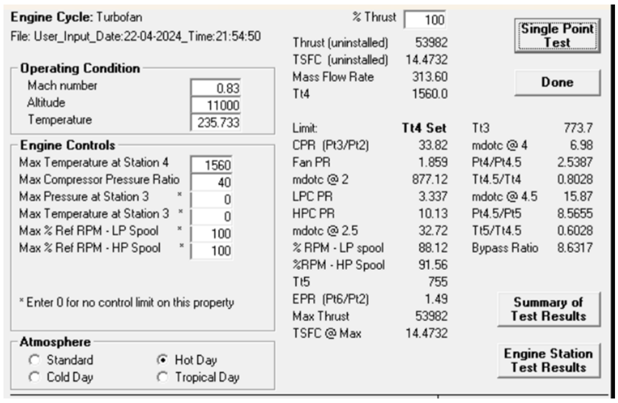


Figure 11. Engine Test at cruise

Lastly we test this reference engine for the loiter phase. The resulting single-engine thrust value comes out to be 60 kN against the required value of 35 kN.

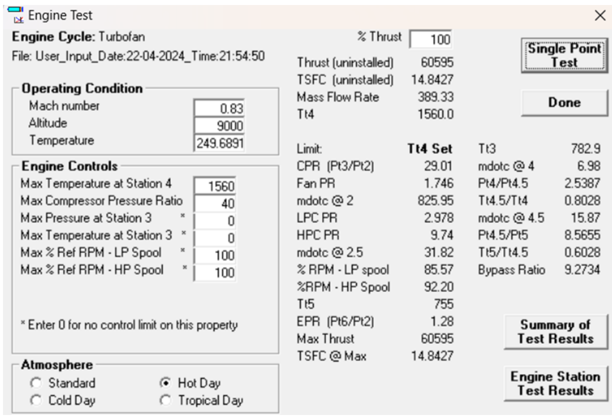


Figure 12. Engine Test at Loiter

The remaining sections (such as descent and taxi) shall not be analyzed in PERF as we know that the thrust requirements for these segments are lower than that of the cruise and taxi. Therefore, we know that our engine will satisfy the criteria for these phases.

4.1. Engine Failure Analysis

Apart from the general constraints, we also have to design the engines such that the aircraft can safely return back without any accidents in the unlikely case of one of the two engines failing. One of these constraints is that the aircraft should be able to climb at a climb gradient of 2.4%.

Similar to the climb analysis we did for two engines, we now perform an analysis for a single engine. The thrust requirements during climb should be met even by a single engine at 100% Thrust.

From the analysis, we get that the aircraft can reach a maximum altitude of approximately 7000 m, while maintaining a 2.4% Climb gradient.

This means that the aircraft travels much more horizontally than before and takes more time to reach its maximum altitude. The figure above shows the comparison.

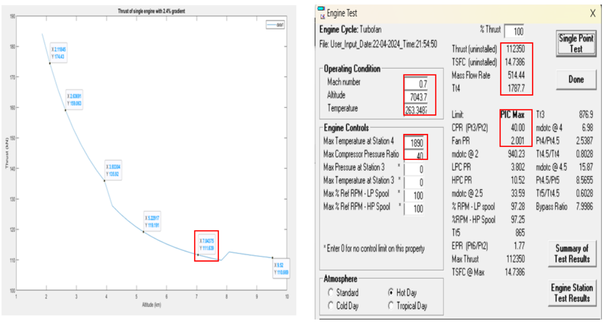


Figure 13. Engine Failure Analysis

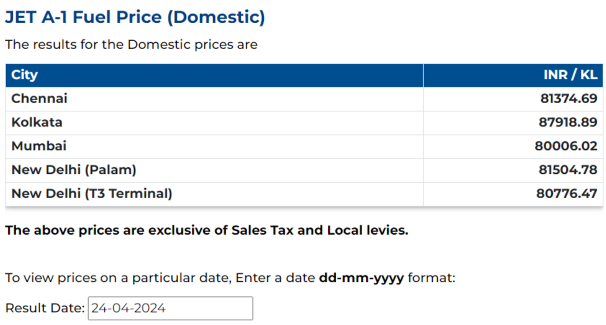
5. Results and Discussions

After ensuring that the aircraft is able to "fly", we perform single point calculations on PARA to find the corrected TSFC for each segment. Then, we update our calculations and find out the mass of fuel burnt, which comes out to be lower than the earlier assumed values.

Phase	Distance (km)	Thrust Requirement (kN)	Corrected TSFC	Fuel Burn (kg)
Taxi	1.5	10	16	13
Takeoff	3.65	333	12.77	333
Climb	330	333	11.7	2957
Cruise	10650	80	16	53355
Descent	140	78	12.77	717.7
Loiter		70	16	2008.5
Land	3.65	70	12	156
Taxi	1.5	10	16	13
Total	11,120			59553.5

Table 5. Thrust Requirements and Fuel Burn for the Different Mission Phases, After Applying the Corrected Values of the TSFC for Each Stage

5.1. Calculation of Ticket Cost



JET A-1 Fuel Price (Domestic)

The results for the Domestic prices are

City	INR / KL
Chennai	81374.69
Kolkata	87918.89
Mumbai	80006.02
New Delhi (Palam)	81504.78
New Delhi (T3 Terminal)	80776.47

The above prices are exclusive of Sales Tax and Local levies.

To view prices on a particular date, Enter a date dd-mm-yyyy format:

Result Date: 24-04-2024

Figure 14. Cost of Jet Fuel [4]

Calculating the Total Cost of Fuel for the Mission

- The cost for Jet A-1 fuel (C_{A1}) is approximately about ₹80000 per m^3.
- The average density of Jet A-1 fuel (ρ_f)= 800 kg/m^3 [5]
- Total volume of fuel required (V_f)= 74.441 m^3
- Maximum Gross Takeoff Weight (W_{GTO})= 1645760 N
- Empty Weight (W_E)= 822880 N
- From table 5, we get the total fuel burn (FB_{tot})= 59553.5 kg
- Therefore, total fuel cost (FC_{tot})= $V_f \times C_{A1}$, which comes out to be ₹59,55,280.
- The final ticket cost (C_{ticket})= FC_{tot}/N_p, which comes out to be ₹23538.65.

Calculating the Payload Characteristics

- Total Weight of the Carriable Payload (W_P)= $M_{GTO} - W_E - FB_{tot} \times g$, which comes out to be 238660.165 N.
- We assume the average weight of an Indian passenger (W_p) to be 60 kg, i.e. 588.6 N, and the number of passengers (N_p) was given to be 253.
- Therefore the weight of cargo (W_C)= $W_P - N_p \times W_p \times g$, which comes out to be 89744.365 N or 9148.25 kg, and when we divide this by N_p, we find that each passenger can carry luggage equivalent to 36.2 kg.

Now, 36.2 kg is not a lot of luggage allowance for a passenger travelling such a long distance. Hence, if we want to accommodate a higher luggage allowance per passenger, we reduce the total number of passengers.
We found that a good luggage allowance about 63 kg.

Calculation of the Final Ticket Cost

Calculating the ticket cost and number of passengers for a luggage allowance of 63 kg, we get:

- Final ticket cost (C_{ticket})= Approximately ₹30,000
- Number of passengers (N_p)= 200

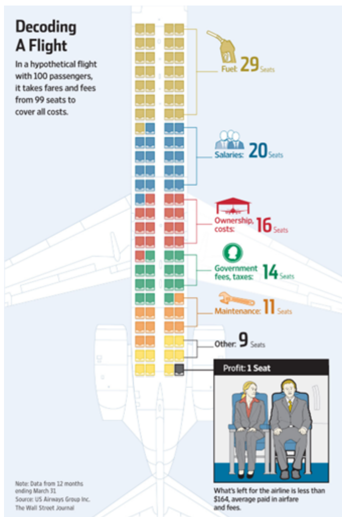


Figure 15. Cost Breakdown of a Hypothetical Flight [3]

- Now, we consider further expenses like taxes, labour costs, material costs, agency profits, company profits, etc.
- Making this consideration, we assume the cost of the fuel to be about 29% of the total flight cost, and therefore the revised ticket cost comes out to be **₹1,03,450**.

6. Conclusion

By working on this project, we learnt and understood a lot of things about designing an aircraft engine. Our work was directed only towards determining an optimal TSFC and ensuring that an economical ticket price was achieved. We encountered a lot of problems while trying to achieve our objective, and solving them made us realise the level of complexity of engine design.

We understood how to analyse an RFP and the required mission and extract the useful details. We learnt to make approximations for different parts of the mission. The approximations could then be made precise through the process of iteration. We realised that there were some trade offs which need to be made if we want to achieve the required conditions of the proposal.

7. Acknowledgements

In the course of completing this project, we faced several challenges. We would like to thank Prof. Dilip S. Sundaram for guiding us through the entire process of doing this project, and also for readily giving us advice when we needed it.

We would also like to thank IIT Gandhinagar for providing us the platform and the resources opportunity to complete this project effectively.

Engine Type	Twin Turbofan with Separate Exhausts
Engine Size	2.1m
Fan Pressure Ratio	2
Compressor Pressure Ratio	40
Bypass Ratio	8
Maximum Tt4 (During Takeoff and engine failure)	1890 K
Maximum Thrust	333 kN
Maximum TSFC	16 g/kN s
Total Fuel Used	59553.5 kg
Maximum Passengers (for 36.2 kg luggage)	253
Fuel Price Per Passenger (for 36.2 kg luggage)	₹23538.65
Maximum Passengers (for 63 kg luggage)	200
Fuel Price Per Passenger (for 63 kg luggage)	₹30000
Maximum Ticket Cost Including All Expenses (for 200 passengers)	₹1,03,450

Table 6. Summary

References

[1] D. T. P. Jack D. Mattingly William H. Heiser, *Aircraft Engine Design*, 2nd. Reston, VA, USA: American Institute of Aeronautics and Astronautics, 2002.

[2] J. D. Mattingly, *Elements of Gas Turbine Propulsion*, 1st. Reston, VA, USA: American Institute of Aeronautics and Astronautics, 2005.

[3] S. McCartney, "How airlines spend your airfare", *The Wall Street Journal*, Jun. 2012, Accessed: April 24, 2024. [Online]. Available: https://www.wsj.com/articles/SB10001424052702303296604577450581396602106.

[4] *Jet fuel price - domestic*, https://hindustanpetroleum.com/jet-fuel-price-domestic, Accessed: April 24, 2024, Hindustan Petroleum Corporation Limited.

[5] Wikipedia contributors. "Aviation fuel". Accessed: April 24, 2024, Wikipedia, The Free Encyclopedia. (current year), [Online]. Available: https://en.wikipedia.org/wiki/Aviation_fuel.