

Developing a Programme for Engine Design Calculations of a Commercial Airliner

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ABSTRACT: This project leads to a path of understanding the necessary fundamental calculations that need to be done during an engine design of a commercial airliner. These calculations are hand based calculations that are done based on the parameters of the airframe data provided by the airline manufacturers. These calculations are a little tedious and require a paper and a pen to carry out the procedures. This project will enable the following outcomes for the students: providing a fundamental understanding of the aircraft engine design, more from the grounds up approach and an automated way (program) of doing the above, enabling faster iterations and making it easy to achieve the required parameters for designing an engine.

Keywords: Bypass ratio, Efficiency, Flow properties, HTML, PHP, Turbofan engine.

I. INTRODUCTION

As we all know that an engine is the heart of any vehicle, without the engine there wouldn't be a vehicle. This heart in an aircraft is really big and there may be not one but four hearts. Designing these engines is really tricky and involves some serious work. Well it would basically require any engine designer to fill up this design sheet (fig 1.1). It is really a tedious work and requires a lot of time. This project basically allows any aeronautical engineering student to get acquainted with the theory behind designing of an aircraft engine and so as to save time with calculation a program for calculating has been developed.

NEW LARGE CIVIL AIRCRAFT Design sheet for the engine

This table gives quantities which are likely to be needed repeatedly. Numbers in parenthesis indicate the exercise in which the data is evaluated.

Initial Cruise altitude = 31000 ft.	Cruise Mach number = 0.85		
Ambient temperature = 226.7 K,	ambient pressure = 28.7 kPa	(Standard Atmosphere)	
Speed of sound at cruise altitude $a =$	m/s;	cruise speed of aircraft $V =$	m/s (1.2)
Aircraft mass at start of cruise = $635.6 \cdot 10^3$ kg,	aircraft lift/drag ratio at cruise = 20.		
Thrust per engine for level flight at 31000ft (assume 4 engines) $F_N =$			kN. (2.3)
Entering engine: Stagn. temp. and press. $T_{02}=T_{01} =$	K,	$p_{02}=p_{01} =$	kPa (6.2)
Turbine inlet stagnation temperature at cruise, $T_{04} = 1450$ K			
Engine pressure ratio at cruise for flow through core, $p_{03}/p_{02} = 40$			
Gas turbine cycle efficiency	$\eta_{cy} =$		(4.3e)
Entering core (after fan): stagn. temp. and press.	$T_{023} =$	K, $p_{023} =$	kPa (5.1)
Leaving core compressor: stagn. temp. and press.	$T_{03} =$	K, $p_{03} =$	kPa
Leaving HP turbine: stagn. temp. and press.	$T_{045} =$	K, $p_{045} =$	kPa (5.1)
Leaving LP turbine: stagn. temp. and press.	$T_{05} =$	K, $p_{05} =$	kPa (7.1)
Velocity of core jet (taken equal to vel. of bypass jet) $V_j =$		m/s	(7.1)
Bypass ratio	$bpr =$		(7.1)
Bypass leaving fan: stagn. temp. and press,	$T_{013} =$	K, $p_{013} =$	kPa (7.1)
sfc of bare engine	$=$	kg/hkg	(7.1)
sfc of installed engine	$=$	kg/hkg	(7.2)

II. LITERATURE SURVEY

The Boeing 747-400 was the largest civil aircraft that was introduced in service in the year 1989 but it is a derivative of the 747-100 which entered service in 1969. Presently Airbus A380 is the largest civil aircraft in this world. The 747-400 incorporated some aerodynamic improvements, including some improvements to the existing engines, but a much more radical redesign is needed to take the full advantage of the developments in aerodynamics and materials since 1969. Boeing and Airbus industries have separately and jointly discussed proposals for a much larger plane, with anywhere from about 600 to about 800 seats.

Boeing at that time had announced plans for a major derivative of the 747 to incorporate new wings and to be propelled by new engines. Two versions were proposed, the 747-500 and 747-600, with a common wing. The 747-600, which it is proposed should be produced first, was intended to carry 548 passengers over a range of up to 7890nm, whilst the 747-500 was to carry 462 passengers over a range of up to 8830nm. Airbus then proposed the A3XX with an expected capacity of 555 passengers for a range of up to 7450nm.

It takes several years to design, develop and certificate a new plane, though the length of time is becoming shorter. It seems to take even longer to develop the engines, but until the specifications of the plane are settled it is not clear what engine is needed. There are currently three major engine manufactures and they are, Rolls Royce in Britain, Pratt & Whitney and General Electrics in the USA and it is their aim to have an engine already for whatever new large aircraft is decided to build.

A specifications for the 'paper' airplanes alter, the 'paper' engine design to power them will also change; literally hundreds of potential engines will be tried to meet a large number of proposals for the new airplane before any company finally commits itself.

It is essential to realize that both the new plane, and the engines which power it, will depend heavily on the experience gained in earlier products. In Table 2.1 below, the proposed specifications for the New Large Aeroplane are therefore set beside those achieved for the 747-700.

Table 2.1 Comparison of some salient aircraft parameters

	New Large Aeroplane NLA	Boeing 747-400
No. of passengers	620	400
Range(nautical miles)	8000	7300
Payload at this range (a)	58.5tonne	38.5tonne
Max take-off weight (d = a + b + c)	635.6tonne	395.0tonne
Empty weight (b)	298.7tonne	185.7tonne
Fuel capacity (c)	275.4tonne	174.4tonne
Cruise Mach No.	0.85	0.85
Initial cruise Altitude	31000ft	31000ft
Cruise Lift/Drag	20	17.5
Wing Area	790m ²	511m ²

The atmosphere through which the plane flies depends on the altitude, with the pressure, temperature and density falling as altitude increases. At high altitude the variations with season, location and time of day is much less than at ground level and it is normal to use a standard atmosphere in considering aircraft and engine performance. According to International Civil Aviation Organization the values at altitudes are expressed as a ratio to the values at sea level. Standard sea-level atmospheric conditions are defined for a standard day: $T_{sl} = 288.15K$, $P_{sl} = 101.3 \text{ kPa}$, $\rho_{sl} = 1.225 \text{ kg/m}^3$. In Standard atmosphere temperature is assumed to decrease linearly with altitude at 6.5K per 1000m below the tropopause, but to remain constant above this altitude at 216.65K. Cruise often begins at 31000ft, and the corridors are separated by 2000ft. below table (2.2) gives information about different conditions of standard atmosphere.

Table 2.2 Useful values of ICAO standard atmosphere

ALTITUDE		TEMPERATURE	PRESSURE	DENSITY
Feet	Km	K	10 ⁵ Pa	Kg/m ³
0	0	288.15	1.013	1.225
31000	9.45	226.73	0.287	0.442
33000	10.05	222.28	0.260	0.336
35000	10.67	218.80	0.238	0.380
37000	11.28	216.65	0.214	0.285
39000	11.88	216.65	0.197	0.316
41000	12.50	216.65	0.179	0.287
51000	15.54	216.64	0.110	0.179

The below fig (2.1) gives the ICAO standard atmosphere:

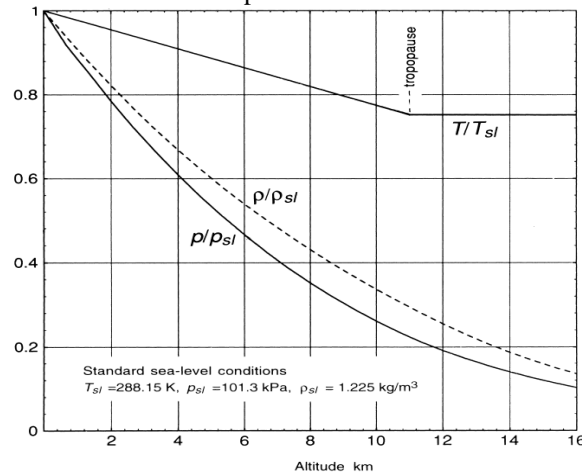


Fig 2.1: ICAO Standard Atmosphere

In order to calculate the speed of vehicle that is required for further calculation part is given by the formula;

$$\text{Mach no} = \frac{\text{SPEED OF VECHILE}}{\text{SPEED OF SOUND}} \quad (1)$$

The speed of sound formula can be given by;

$$a = \sqrt{\gamma RT} \quad (2)$$

Where: a = Speed of sound

$$\gamma = 1.4$$

$$R = 287 \text{ J/kg K}$$

T = Temperature at given altitude (i.e. Ta)

Hence in order to find out the speed of the vehicle;

$$V = M \times a \quad (3)$$

Where: V = speed of sound

M = Mach no

a = speed of sound

III. THEORY BEHINDTHE CALCULATION OF ESSENTIAL PARAMETERS

There are three main critical legs of a flight that affect the engine performance, they are: Take-off, Climb and Cruise. In any engine the first parameter to be fixed would be thrust. There has to be a compromise between a machine that can travel fast at cruise and relatively slow during take-off and landing. The lift coefficient is given as;

$$C_L = L / \frac{1}{2} \rho AV^2 \quad (4)$$

Where L is the lift force that acts perpendicular to the direction of travel. The figure below shows how the lift of the aircraft varies with angle of attack at low speeds such as take-off and it can be seen that the lift rises almost proportionally to the incidence until around the peak, beyond which it falls rapidly. The rapid fall in the lift is referred to as stall and in simple terms occurs when the boundary layer separates from the upper surface of the wing.

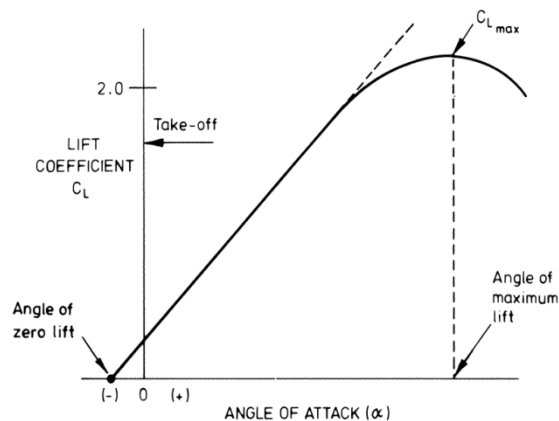


Fig 3.1: A Typical curve of Lift vs Incidence

It is an objective of a civil aircraft to lift as much as it can with the least possible drag. Reducing the drag for the same lift allows the aircraft to use less fuel and to travel further. For steady level flight at small incidence, as for cruise, two statements can be made on the basis of simple mechanics:

$$\text{Lift} = \text{Weight} \quad \text{and} \quad \text{Drag} = \text{Thrust of the engines}$$

To estimate the range we use Breguet's Range equation;

$$\frac{dw}{dt} = -g(\text{sfc} \times \text{net thrust}) = -g(\text{sfc} \times \text{drag}) = -g \frac{\text{sfc} \times w}{1/d} \quad (5)$$

Where L/D ratio of lift to drag and we use $W=L$. Rearranging

$$\frac{dw}{w} = -\frac{g \text{ sfc} dt}{L/D} \quad (6)$$

This equation can then be rewritten in terms of distance travelled s as

$$\frac{dw}{w} = -\frac{g \text{ sfc} ds}{V L/D} \quad (7)$$

Keeping VL/D and sfc constant the above equation can be then integrated to give *Breguet's range formula*

$$s = -\frac{VL}{g \text{ sfc}} \times \ln\left(\frac{w_{\text{end}}}{w_{\text{start}}}\right) \quad (8)$$

A flow of fuel m_f passes down the pylon, but its velocity is low and it conveys negligible momentum. A mass flow of air m_{air} enters the engine; m_{air} is typically two orders of magnitude greater than m_f . It is assumed for simplicity here that the jet is uniform as it crosses the control surface with velocity V_j . The thrust is calculated by considering the flux of momentum across the control surface around the engine – since the pressure is assumed uniform over the control surface the pressure creates no net force. Considering the flow which crosses the control surface and passes through the engine, we can write down two fluxes are;

$$\text{Flux of momentum entering the engine} = m_{air} \times V. \quad (9)$$

$$\text{Flux of momentum leaving the engine} = (m_{air}+m_f) \times V_j. \quad (10)$$

The extra mass flow in the jet is included here for completeness, but this represents a small effect for high bypass ratio engine.

The net thrust F_N which is available in flight is given by the difference between the two momentum fluxes, that is;

$$F_N = (m_{air}+m_f) \times V_j - m_{air} \times V \quad (11)$$

If the engine were being tested on a stationary test bed or before the aircraft started to move the thrust produced is known as the **gross thrust**, which, since $V=0$, is given by;

$$F_G = (m_{air}+m_f) \times V_j. \quad (12)$$

The difference between gross and net thrust is $m_{air}V$, often referred to as the **ram drag** or the **inlet-momentum drag**, so;

$$F_N = F_G - m_{air} \times V_j. \quad (13)$$

Here we have ignored the drag on the outside of the engine nacelle drag, which leads to a ring of reduced relative velocity around the jet and consequent reduction in useful thrust. Unfortunately, nacelle drag is becoming a more significant factor as the engine becomes larger for the same thrust, i.e. as the bypass ratios go up. The Kinetic Energy of the engine can be given as;

$$\Delta KE = \frac{1}{2} [(m_{air} + m_f)V_j^2 - m_{air}V^2] \quad (14)$$

The power actually associated with propelling the aircraft is given by;

$$\begin{aligned} \text{Power to the aircraft} &= \text{flight speed} \times \text{net thrust} = V \times F_N \\ &= V[(m_{air}+m_f) \times V_j - m_{air} \times V]. \end{aligned} \quad (15)$$

The propulsive efficiency compares the power supplied to the aircraft with the rate of increase in kinetic energy of the air through the engine. Propulsive efficiency η_p is straightforward to define by;

$$\eta_p = \frac{\text{Power to aircraft}}{\text{Power to jet}} \quad (16)$$

$$\eta_p = \frac{V[(m_{air}+m_f) \times V_j - m_{air} \times V]}{\frac{1}{2}[(m_{air} + m_f)V_j^2 - m_{air}V^2]} \quad (17)$$

Since, as already noted, the mass flow rate of fuel is very much less than that of air it is possible to write η_p with sufficient accuracy for our present purposes as;

$$\eta_p = \frac{2V}{V+V_j} \tag{18}$$

The above equation is known as the *Froude* equation for propulsive efficiency.

The overall efficiency is given by;

$$\eta_o = \frac{\text{Useful work}}{\text{Thermal energy from fuel}} = \eta_p \times \eta_{th} \tag{19}$$

From the first law of thermodynamics when applied to this steady flow process can be written as;

$$Q_{net} - W_{net} = m_{air} \times \Delta h \tag{20}$$

Where Δh is the enthalpy difference between the inlet air and the exhaust based on stagnation condition;

$$Q_{net} - W_{net} = m_{air} \times C_p \times (T_5 - T_2) \tag{21}$$

This can be written in terms of lower calorific value of the fuel;

$$mf \times lcv = m_{air} \times C_p \times (T_4 - T_3) \tag{22}$$

These isentropic processes are those which ideal compressors and turbines would perform. As can be seen, the actual compression process involves a greater temperature rise than

That of the isentropic compressor for the same pressure rise;

$$T_3 - T_2 > T_{3is} - T_2 \tag{23}$$

And in the same way

$$T_4 - T_5 < T_4 - T_{5is} \tag{24}$$

For compressors and turbines it is normal to define efficiencies which relates actual work per unit mass flow to that of an ideal (i.e. loss free) machine with equivalent pressure change;

$$\eta_{comp} = \frac{\text{Ideal work}}{\text{Actual work}} \text{ And } \eta_{turb} = \frac{\text{Actual work}}{\text{Ideal work}} \tag{25}$$

Treating fluid as a perfect gas, for which $h = c_p T$.

$$\eta_{comp} = \frac{T_{3is} - T_2}{T_3 - T_2} \text{ and } \eta_{turb} = \frac{T_4 - T_5}{T_4 - T_{5is}} \tag{26}$$

Nowadays the isentropic efficiencies in a high quality aircraft engine for use on a civil aircraft are likely to be around 90% for compressor and turbine and this round number will normally be used in this project.

$$\frac{p}{T^\gamma} = \text{constant} \tag{27}$$

This means in the present cases,

$$\frac{T_{3is}}{T_2} = \left(\frac{p_3}{p_2}\right)^{(\gamma-1)/\gamma} \text{ and } \frac{T_4}{T_{5is}} = \left(\frac{p_4}{p_5}\right)^{(\gamma-1)/\gamma} \tag{28}$$

The power which must be supplied to the compressor is given by;

$$W_c = m_{air} \times C_p \times (T_3 - T_2) \tag{29}$$

$$W_t = m_{air} \times C_p \times (T_4 - T_{5is}) \tag{30}$$

The Thermal Efficiency of the cycle can be given as;

$$\eta_{cycle} = \frac{W_{net}}{m_{air} \times C_p \times (T_4 - T_3)} \tag{31}$$

But the thermal efficiency can be given by;

$$\eta_{th} = W_{net}/mf \times LCV \tag{32}$$

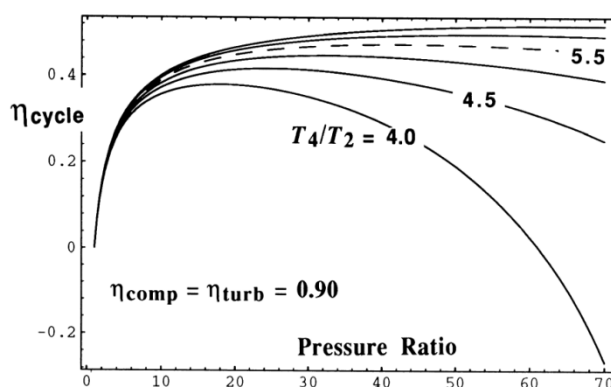


Fig 3.3: The Comparison between Efficiency of cycle and that of Pressure Ratio.

The blade cooling can be given by;

$$\varepsilon = (T_g - T_m)/(T_g - T_c) \tag{33}$$

Turbine inlet temperature is important because increasing its value makes the pressure ratio across the core turbine smaller in relation to the pressure rise of the core compressor and thereby increases the power available from the LP turbine. Increasing this temperature also increases the thermal efficiency, provided that the pressure ratio increases by an appropriate amount.

Table 3.1:Representing the Compressor and Turbine inlet temperatures

	T_2	T_4	T_4/T_2
Take-Off (Standard day, sea level)	288.15 K	1700 K	5.90
Top of Climb(31000ft, M=0.85)	259.5 K	1575 K	6.07
Start Of Cruise(31000ft, M=0.85)	259.5 K	1450 K	5.59

The steady flow energy equations for the flow of gas with no heat transfer and no external work transfer can be written as;

$$h_1 + \frac{v_1^2}{2} = h_2 + \frac{v_2^2}{2} \tag{34}$$

Where h is the enthalpy, which can be rewritten in a way more convenient for us;

$$C_p \times T + \frac{v^2}{2} = C_p \times T_o \tag{35}$$

Where T_o is the *stagnation* temperature, being that temperature which the gas would attain if brought to rest without work and heat transfer, not necessarily in an ideal or loss-free manner. The specific heat can be rewritten as $C_p = \gamma R / (\gamma - 1)$ and on rearranging and inserting this into the above equation the stagnation temperature may be written as;

$$\frac{T_o}{T} = 1 + (\gamma - 1)/2 \times \left(\frac{v^2}{\gamma RT}\right) \tag{36}$$

Again this equation between stagnation temperature T_o and static temperature T does not imply ideal or loss-free acceleration or deceleration.

At this point we suppose that the acceleration or deceleration of the gas between the static state p and T and that at stagnation p_o and T_o is reversible and adiabatic, for which we know that;

$$\frac{p}{\rho^\gamma} = \text{constant and } \frac{p}{T^{\frac{\gamma}{\gamma-1}}} = \text{constant} \tag{37}$$

It then immediately follows that;

$$\frac{p_o}{p} = \left(\frac{T_o}{T}\right)^{\gamma/(\gamma-1)} \tag{38}$$

The bypass ratio is defined as the mass flow of the air passing outside the core divided by the mass flow through the core;

$$bpr = mb/mc \tag{39}$$

The choice of bypass ratio has a major effect on the efficiency, because for a given core the bypass ratio determines the jet velocity. The bypass ratio also greatly affects the appearance, size and weight of the engine: the pure turbojet has a small diameter relative to its length, whereas high bypass ratio engines have their overall diameter comparable to their length. Previous engines (the RB211, CF6 and JT9D generation) had bypass ratios of around 5 and we may assume that a bypass ratio at least as great as this will be employed in a new engine for the New Large Aircraft.

To achieve this it would probably be necessary to install a gear box between the LP turbine and the fan, to allow the turbine to run faster; this requires considerable development and will very probably be heavy. For a civil transport engine a major requirement is low fuel consumption, equivalent to high overall efficiency. The overall efficiency is the product of propulsive and thermal efficiency;

$$\eta_0 = \eta_p \times \eta_{th} \tag{40}$$

The combination of constraints on η_{cy} and η_p give rise to the high bypass engine for subsonic propulsion. The constraints give rise to;

$$sf_{installed} = \{1.04 + 0.01(bpr - 1)\}sf_{bare} \tag{41}$$

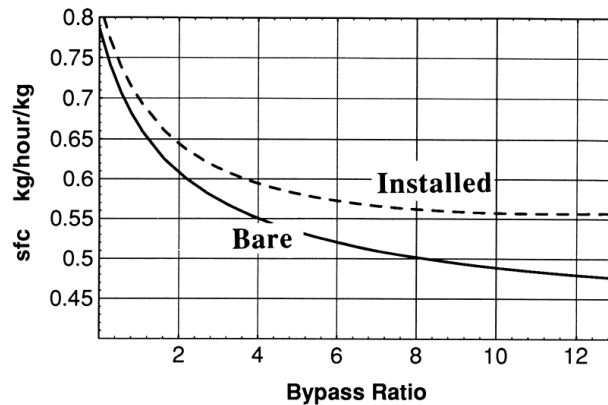


Fig 3.4: Prediction variation of sfc with bypass ratio for bare and installed engine

IV. PROBLEMS WITH SOLUTION

To find the engine thrust needed for steady flight at start of cruise, when the weight of the aircraft and wing area for the NLA at altitude 31000ft and Mach number of 0.85. Given;

Initial Cruise altitude: 31000ft Cruise Mach number: 0.85 Ambient Temperature= 226.7K

Ambient pressure= 28.7×10^3 Pa Speed of sound at cruise altitude $a = 301.827$ m/s

Cruise speed of the aircraft $V = 256.5$ m/s Aircraft mass at start of cruise = 635.6×10^3 Kg

Aircraft L/D ratio= 20

And thus;

$L/D = 20$

Lift/ Drag = 20

Replacing Lift with Weight and Drag with Thrust,

Weight/Thrust = 20

$W/T = 20$

$635.6 \times 1000 \times 9.81/T = 20 \quad T = 635.6 \times 1000 \times 9.81/20$

$T = 311.655$ kN

Thrust per engine (Assuming the large aircraft to have 4 engines) = 77.914 kN/engine

Calculating the thermal efficiency and the net work per kg of air flowing in the engine. Assume isentropic efficiencies of compressor and turbine is 90%. For the following conditions;

$T_2 = 259.5$ K, $T_4 = 1450$ K, PR = 40.

Given: $T_2 = 259.5$ K

$T_4 = 1450$ K

PR = 40

$$T_{3is} = \left(PR \frac{\gamma-1}{\gamma} \right) \times T_2 \quad (41)$$

$T_{3is} = 716.63$ K on substitution of values in the above equation

∴

The work done by the compressor is,

$$W_c = C_p \times (T_{3is} - T_2) / \eta_{comp}$$

$$W_c = 510.47 \text{ Kj/kg}$$

$$T_{5is} = T_4 / \left(PR \frac{\gamma-1}{\gamma} \right) \quad (42)$$

$T_{5is} = 525.05$ K on substituting the values in the above equation

∴

The work done by the turbine is;

$$W_t = C_p \times (T_4 - T_{5is}) \times \eta_{turb} \quad (43)$$

$$W_t = 836.61 \text{ Kj/kg}$$

∴

$$W_{net} = W_t - W_c = 326.14 \text{ kJ/kg}$$

$$\eta_{th} = \frac{W_{net}}{C_p} \times (T_4 - T_3) = 0.4754$$

Finding Temperature entering the core compressor and hence the temperature at exit from the core compressor and the temperature leave the H.P turbine and the pressure at the outlet from the turbine. The temperature and pressure at cruise is 259.5K and 46.0 KPa. Assume a pressure ratio of 1.6 for flow through the fan which enters the core and a pressure ratio of 25 in the core compressor itself. The temperature of the entry of H.P turbine is 1450K. Assume efficiency of compressor and turbine 90%.

$$\frac{T_{fan}}{T_{air}} = Pr^{(\gamma-1)/\gamma} = 1.143$$

$$T_{fan} = 296.79K$$

$$(T_{fanis} - T_{air}) \times 100 = (T_{fan} - T_{air}) \times 90$$

$$T_{fan} = 300.9K$$

$$\Delta T = 41.4K$$

Now to find the Core temperatures:

$$\frac{T_{textcoreis}}{T_{textcore}} = Pr2^{(\gamma-1)/\gamma}$$

$$T_{textcoreis} = 754.80K$$

$$(T_{textcoreis} - T_{textcore}) \times 100 = (T_{textcore} - T_{textcore}) \times 90$$

$$T_{textcore} = 805.2K$$

$$\Delta T_{textcorecomperssor} = \Delta T_{textturbine}$$

$$T_{textturb} = 945.6K$$

$$(T_{textturb} - T_{textturbis}) \times 100 = (T_{textturb} - T_{textturbis}) \times 90$$

$$T_{textturbis} = 889.66K$$

$$P_{fan} = 1.6 \times 46 = 73.6 \quad P_{comp} = 73.6 \times 25 = 1840$$

$$\frac{T_{5is}^{\gamma/(\gamma-1)}}{T_4} = \frac{P}{1840} = 332.82 \text{ kPa}$$

If the airplane cruises at Mach number of 0.85 at an altitude of 31000ft find the stagnation temperature and pressure of the flow perceived by the plane.

We know that;

$$\frac{T_0}{T} = 1 + \frac{\gamma-1}{2} \times M^2 = 1.1445 \tag{44}$$

$$T_0 = 1.1445 \times 226.73 = 259.5K$$

$$\frac{p_0}{p_a} = \left(\frac{T_0}{T}\right)^{\frac{\gamma}{\gamma-1}} = 1.603$$

$$p_0 = 1.603 \times 28.7 = 46.0 \text{ kPa}$$

Using the results above to provide the inlet conditions into the LP turbine at a flight Mach number of 0.85 at 31000 ft. Assume equal velocity for the core and bypass jets. Assume that the inlet decelerates the flow isentropically and the nozzles expand the flow isentropically to ambient pressure. Take the isentropic efficiencies for the fan and for the LP turbine to be each 0.90. The following is calculated: The jet velocity, propulsive efficiency, gross thrust F_G and net thrust F_N and the Overall Efficiency.

For $bpr = 10$

We know that

$$(bpr + 1) \times \Delta T_{fan} = \Delta T_{lpturb}$$

$$7 \times \Delta T_{fan} = \Delta T_{lpturb} = 376.2K$$

∴

$$T_{05} = 945.7 - 376.2 = 569.5K$$

$$T_{05is} = 945.7 - \frac{\Delta T_{lpturb}}{0.90} = 527.7K$$

Temp ratio = 7.705

$$P_{05} = 43.2 \text{ kPa}$$

$$P_a = 28.7 \text{ kPa}$$

Pressure ratio = 0.889

$$V_j^2 = 2 \times 1004.5 \times 569.5 \times (1 - 0.889)$$

$$V_j = 355.3 \text{ m/s}$$

$$F_G = \frac{V_j}{m_{air} + m_f} = 3.91 \text{ kN kg}^{-1} \text{ s}$$

$$\eta_p = \frac{2V}{V + V_j} = 0.839$$

$$\eta_0 = 0.428 \quad F_N = 1.087 \text{ kN kg}^{-1} \text{ s} \quad sfc = 0.492 \frac{\text{kg}}{\text{h/kg}}$$

From the above results the following graphs are obtained:

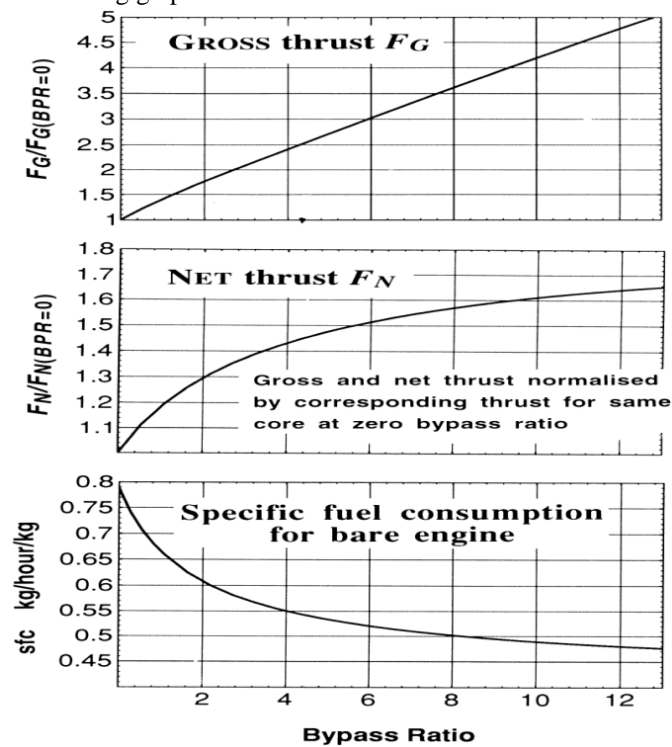


Fig 4.1: Prediction variations in Thrust and SFC with bypass ratio for constant core

V. RESULTS AND CONCLUSION

From the above we can say that the parameters are interdependent on each other and one must be careful in calculating the parameters involved in designing an engine for a new large aircraft. As one can see that there is a lot of work involved when one sits to design an engine, in order to simply the above process we developed a program using HTML and PHP 5 programming language. The result program looks like this:

E.E.P.C.U

(Easy Engine Performance Calculation Utility)

Please Input Initial Parameters and Hit Submit to Calculate performance Metrics.	
Initial Cruise Altitude	<input type="text" value="10000"/>
Cruise Mach Number	<input type="text" value="85"/>
Aircraft Mass at Start Of Cruise (Tonnes)	<input type="text" value="635.6"/>
Aircraft L/D Ratio at Cruise	<input type="text" value="20"/>
Number of Engines	<input type="radio"/> One <input checked="" type="radio"/> Two <input type="radio"/> Three <input type="radio"/> Four
Engine Pressure Ratio at Cruise	<input type="text" value="15"/>
Pressure Ratio of the core compressor	<input type="text" value="15"/>
Efficiency of Compressor	<input type="text" value="90"/>
Efficiency of Turbine	<input type="text" value="90"/>
<input type="button" value="Submit Form"/>	

Fig 5.1: The main page of the program with all the input variables

Hence one can just input all the variables required and then submit the form. The result when one submits the form looks like this:

Calculated Metrics based on Input Values.			
Basic Parameters of Engine.			
Initial Cruise Altitude	31000 ft (9448.8 m)	Initial Cruise Mach	.85
Ambient Temperature	226.77 K	Ambient Pressure	0.287 Pa
Air Density	0.442 kg/m ³	Speed of Sound @ 31000 ft	301.855 m/s
Aircraft Speed	256.576 m/s	Aircraft Weight	6233106.74 Kgs
Target Aircraft Thrust with 4 Engines		311.655 kN (77.914 kN / Engine)	
Air Entering Engine.			
Stagnation Temperature ($T_{02}=T_{01}$)	259.538 K	Stagnation Pressure ($P_{02}=P_{01}$)	46.087 kPa
Turbine inlet stagnation Temperature at Cruise (T_{04})	1450 K	Engine pressure ratio at Cruise (flow through core (P_{03}/P_{02}))	30
Gas Turbine Thermal Efficiency of the aircraft		0.471	
Air Entering the core engine			
Entering Core Compressor Stagnation Temperature (T_{023})	300.984 K	Entering Core Stagnation pressure (P_{023})	73.74 kPa
Leaving Core Compressor Stagnation Temperature (T_{03})	890.113 K	Leaving Core Compressor Stagnation Pressure (P_{03})	2580.895 kPa
Leaving HP Turbine Stagnation Temperature (T_{045})	860.871 K	Leaving HP Turbine Stagnation Pressure (P_{045})	2540.345kPa

Fig 5.2: The output page of the program

The input page(fig 5.1) of the program is written in HTML, whereas the output page(fig 5.2) is written in PHP5 programming language.

The site on which the program is been uploaded is www.konidala.in/engine. The project as said is intended to help one learn about the engine designing and all reduce efforts. We have succeeded in making one understand the basics behind the theory of designing. The main objective as discussed in the introduction was to fill in the design sheet that is essential for designing an engine and thus:

NEW LARGE CIVIL AIRCRAFT Design sheet for the engine

This table gives quantities which are likely to be needed repeatedly. Numbers in parenthesis indicate the exercise in which the data is evaluated.

- Initial Cruise altitude = 31000 ft, Cruise Mach number = 0.85
 Ambient temperature = 226.7 K, ambient pressure = 28.7 kPa (Standard Atmosphere)
 Speed of sound at cruise altitude $a =$ m/s; cruise speed of aircraft $V =$ m/s (1.2)
 Aircraft mass at start of cruise = $635.6 \cdot 10^3$ kg, aircraft lift/drag ratio at cruise = 20.
 Thrust per engine for level flight at 31000ft (assume 4 engines) $F_N =$ kN. (2.3)
- Entering engine: Stagn. temp. and press. $T_{02}=T_{01} =$ K, $p_{02}=p_{01} =$ kPa (6.2)
 Turbine inlet stagnation temperature at cruise, $T_{04} = 1450$ K
 Engine pressure ratio at cruise for flow through core, $p_{03}/p_{02} = 40$
 Gas turbine cycle efficiency $\eta_{cy} =$ (4.3e)
- Entering core (after fan): stagn. temp. and press. $T_{023} =$ K, $p_{023} =$ kPa (5.1)
 Leaving core compressor: stagn. temp. and press. $T_{03} =$ K, $p_{03} =$ kPa
 Leaving HP turbine: stagn. temp. and press. $T_{045} =$ K, $p_{045} =$ kPa (5.1)
 Leaving LP turbine: stagn. temp. and press. $T_{05} =$ K, $p_{05} =$ kPa (7.1)
 Velocity of core jet (taken equal to vel. of bypass jet) $V_j =$ m/s (7.1)
 Bypass ratio $bpr =$ (7.1)
 Bypass leaving fan: stagn. temp. and press, $T_{013} =$ K, $p_{013} =$ kPa (7.1)
 sfc of bare engine $=$ kg/hkg (7.1)
 sfc of installed engine $=$ kg/hkg (7.2)

Fig 5.3: Design sheet

Calculated Metrics based on Input Values.			
Basic Parameters of Engine.			
Initial Cruise Altitude	31000 ft (9448.8 m)	Initial Cruise Mach	0.85
Ambient Temperature	226.77 K	Ambient Pressure	0.287 Pa
Air Density	0.442 kg/m ³	Speed of Sound @ 31000 ft	301.855 m/s
Aircraft Speed	256.576 m/s	Aircraft Weight	6233106.74 Kgs
Target Aircraft Thrust with 4 Engines	311.655 kN (77.914 kN / Engine)		
Air Entering Engine.			
Stagnation Temperature ($T_{02}=T_{01}$)	259.538 K	Stagnation Pressure ($P_{02}=P_{01}$)	46.087 kPa
Turbine inlet stagnation Temperature at Cruise (T_{04})	1450 K	Engine pressure ratio at Cruise (flow through core (P_{03}/P_{02}))	30
Gas Turbine Thermal Efficiency of the aircraft	0.471		
Air Entering the core engine			
Entering Core Compressor Stagnation Temperature (T_{023})	300.984 K	Entering Core Stagnation pressure (P_{023})	73.74 kPa
Leaving Core Compressor Stagnation Temperature (T_{03})	890.113 K	Leaving Core Compressor Stagnation Pressure (P_{03})	2580.895 kPa
Leaving HP Turbine Stagnation Temperature (T_{045})	860.871 K	Leaving HP Turbine Stagnation Pressure (P_{045})	2540.345kPa

Fig 5.4: The completed design sheet.

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