THERMAL ANALYSIS OF A GAS TURBINE CYCLE FOR A TURBOJET ENGINE

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ABSTRACT

This paper deals with the application of a gas turbine engine for a single spool turbojet engine. The design and operation theories of the components used in the jet engines are complicated. The complexity of thermodynamic analysis makes it impossible to mathematically solve the optimization equations involve in turbojet engine cycle. It was expected that these tools would help in predicting the performance of individual components such as compressors, combustion chamber, turbine, nozzle etc. The present work based on detailed parametric thermodynamic analysis of the single spool turbojet engine with transpiration air cooling techniques for turbine blade cooling. The analysis has been carried out by selecting models for different components of engines. Software in “C++” has been developed, which is capable of predicting engine dependent parameters for turbojet engine at varying independent parameters. In conventional configurations, the high temperature and high pressure gas expand through the turbines which provide just enough power to drive the compressor and other engine auxiliaries. Further expansion of the gases takes place in the nozzle to produce the high speed jet. Transpiration cooling technique has been taken for the turbine blade in the present analysis. Amount of the air bled from the high pressure compressor as a fraction of the mainstream flow has been calculated. Cooling of turbine blades (first stator only) also examined at various altitudes and Mach numbers for the various configurations of turbojet and turbofan engines.

Keywords: Gas Turbine, Turbojet Engine, Specific Thrust, Turbine Blade Cooling.
1. INTRODUCTION

Engines are installed on aircraft for propulsion. Exhaust gases from aircraft engines exit at the rear of the engine nozzle, causing the aircraft to move forward and air to pass over the aircraft wings. This air motion creates lift. Most modern aircraft use gas turbine engines to produce the required thrust force. Such engines are relatively light and compact and have a high power to weight ratios. Aircraft gas turbines operate in an open cycle based on the Brayton cycle. Actual jet cycles are not ideal due to thermodynamic losses. The gases in the jet cycle are expanded in the gas turbine to the pressure that allows generation of the power needed by the compressor, generator, hydraulic pump, and other work-consuming devices. Other gas turbine-based aerospace engines exist beyond the turbojet, including turbofan and turboshaft engines, but only turbojets are considered here. Production of jet aircraft began in 1945, and the technology has advanced significantly since then. Efforts to increase power and reduce noise and fuel consumption have led to improvements in turbojet and turbofan engines [1]. Nearly 16,800 jet aircraft were operating globally in 2007, and this number is expected to increase to 35,300 by 2024 [2]. Due to decreasing world oil reserves, increasing oil prices and increasing environmental concerns, efforts have increased to improve system efficiency. Exergy analysis is a practical and useful tool for such activities, with many engineers and scientists suggesting that exergy analysis is a highly effective method for evaluating and enhancing thermodynamic performance, and superior to energy analysis [3].

The state of present technologies in technical and also non-technical practice implies creation of growing complexity of systems. Turbojet engine as a complex system is multidimensional highly parametric system with complex dynamics and non-linearities. Its particular property is operation in a broad spectrum of changes in environment (e.g., temperatures from -60 to +40 °C). If we want to secure optimal function of such system, it is necessary to develop models of a turbojet engine for the prediction of the results that would be closed to real system. Furthermore we need to design a control system that will secure operation converging to optimal one in all eventual states of environment and also inner states of the system represented by its parameters. Except implementation of classic algorithms of control, it is possible to design such systems of control and models by use of progressive methods of artificial intelligence [4]. Measurement programs designed to ascertain the vulnerability and survivability of gas turbine engines in airborne weapon systems are currently producing operational results which apply to a broad range of combat scenarios. Battlefield situations have been postulated in triservice studies [5, 6]. However, the reported work does not consider the response of the engine to ingestion of dust-laden air. Survivability studies for airborne command aircraft must consider potential engine degradation caused by the ingestion of atmospheric dust in the operational theater. The reality of this engine degradation mechanism has been dramatically emphasized by recent incidents in which gas-turbine-powered air transports have attempted to traverse volcanic ash clouds. Motivated by the disastrous nature of these encounters on engine performance, the measurement program described in this paper was initiated. Gas turbine engines are routinely tested for the effect of ingestion of solid particles according to the procedures of Military Specification MIL-E-5007D, the commonly known Arizona road-dust test. Within the last several years the material used in the road-dust test was changed from an earthlike material to a crushed quartz material. Encounters with volcanic clouds and clouds of earthlike soils are felt to be much different than the MIL-E-5007D test in material chemical composition, particle size distribution, and cloud concentration [7].

2. MATERIALS AND METHODS

2.1 Modeling of the Components

Early generation jet engines were pure turbojets, designed initially to use a centrifugal compressor, and very shortly afterwards began to use axial compressors for a smaller diameter to the
overall engine housing. They were used because they were able to achieve very high altitudes and speeds, much higher than propeller engines, because of a better compression ratio and their high exhaust speed. However, they were not very fuel efficient.

![Fig. 1: Schematic of simple turbo jet engine](image)

Fig. 1 represents the single-spool (or single-shaft) turbojet engine. In which air enters in an intake before being compressed to a higher pressure by a compressor. The compressed air passes into a combustor, where it is mixed with a fuel and ignited. The hot combustion gases then enter a windmill-like turbine, where power is extracted to drive the compressor. Although the expansion process in the turbine reduces the gas pressure (and temperature) somewhat, the remaining energy and pressure is employed to provide a high-velocity jet by passing the gas through a propelling nozzle. This process produces a net thrust opposite in direction to that of the jet flow.

### 2.1.1 Atmospheric Model

As per the ISA, pressure altitude is not set by the elevation of the point above sea level. Standard day temperature falls at the rate of approximately $6^\circ$C per 1000 m [8] until a pressure altitude of 11000 m which is referred to the tropopause (the region below this is the troposphere, and that above it the stratosphere) and is given as

1. If Altitude (m) $\leq$ 11000
   
   \[ T_a = T_{sl} - 0.00649 \times \text{Altitude} \]  
   \[ p_a = p_{sl} \left( \frac{T_a}{T_{sl}} \right)^{5.256} \]  

2. If 11000 < Altitude (m) $\leq$ 25000

   \[ T_a = 216.66 \]  
   \[ p_a = 0.2265 \times e^{1.73 - 0.000157 \times \text{Altitude}} \]  

3. If Altitude (m) > 25000

   \[ T_a = 141.89 + 0.002991 \times \text{Altitude} \]  
   \[ p_a = 0.02488 \left( \frac{T_a}{216.66} \right)^{5.256} \]
3.1.2 Gas Model

The thermodynamic properties of air and products of combustion are calculated by considering variation of specific heat and with no dissociation. Specific heats against temperature variation have been published in many references such as [9, 10, 11].

2.1.2 Diffuser Model

At inlet of the diffuser the condition of the air will be given by,

\[
\frac{T_{0,i}}{T_i} = \left[1 + \frac{\gamma_{\text{air}} - 1}{2} \cdot M^2\right] (7)
\]

\[
\frac{P_{0,i}}{P_i} = \left(\frac{T_{0,i}}{T_i}\right)^{\frac{\gamma_{\text{air}} - 1}{2} \cdot M^2} \text{ for } M \leq 1 (8)
\]

\[
\frac{P_{0,i}}{P_i} = \left(\frac{T_{0,i}}{T_i}\right)^{\frac{\gamma_{\text{air}} - 1}{2} \cdot M^2} \times \left(\frac{P_{0,e}}{P_{0,i}}\right)_{\text{shock}} \text{ for } 1 < M \leq 5 (9)
\]

where \(\left(\frac{P_{0,e}}{P_{0,i}}\right)_{\text{shock}} = 1 - 0.075(M - 1)^{1.35}\)

At exit of the diffuser the condition of the air will be given by,

\[
T_{0,e} = T_{0,i} (11)
\]

\[
\frac{P_{0,e}}{P_{0,i}} = \left[1 + \eta_d \cdot \left(\frac{\gamma_{\text{air}} - 1}{2}\right) \cdot M^2\right]^{\frac{\gamma_{\text{air}} - 1}{2} \cdot M^2} (12)
\]

2.1.3 Compressor Model

The purpose of a compressor is to increase the total pressure of the gas stream that required by the cycle by absorbing the minimum possible shaft power.

Mass balance for compressor:

\[
\dot{m}_{\text{air},i} = \dot{m}_{\text{air},e} + \sum \dot{m}_{\text{cl}} (13)
\]

Energy balance for compressor in terms of enthalpy and mass:

\[
W_c = \dot{m}_{\text{air},e} \cdot h_{\text{air},e} + \sum \dot{m}_{\text{cl}} \cdot h_{\text{cl}} - \dot{m}_{\text{air},i} \cdot h_{\text{air},i} (14)
\]

where \(\dot{m}_{\text{cl}}\) is the mass of the cooling air which is extracted from the compressor stages. It will be calculated after turbine cooling flow calculation.

2.1.4 Combustion Chamber Model

The purpose of the combustion system of an aircraft gas turbine engine is to increase the thermal energy of a flowing gas stream by combustion. Inflowing air is diffused at the entrance of the burner.

Mass balance for the combustion chamber component will be given by

\[
\dot{m}_{\text{gas}} = \dot{m}_{\text{air},i} + \dot{m}_{f,b} (15)
\]
Energy balance for the combustion chamber will be as follows

\[ \eta_b \dot{m}_{fb} \text{LCV}_f = \dot{m}_{\text{gas}, e} h_{\text{gas}, e} - \dot{m}_{\text{air}, i} h_{\text{air}, i} \]  
(16)

The pressure at combustor exit is given by

\[ p_{b,e} = p_{b,i} - \Delta p_b \]  
(17)

where \( \Delta p_b \) is the pressure loss in the combustion chamber taken as 2% of \( p_{b,i} \), for simplicity.

The fuel to air ratio (FAR) is calculated as,

\[ \frac{\dot{m}_{fb}}{\dot{m}_{\text{air}}} = \frac{c_{\text{gas}, T_{0,e}} - c_{\text{pair}, T_{0,i}}}{\eta_b \text{LCV}_f - c_{\text{gas}, T_{0,e}}} \]  
(18)

\( \text{LCV}_f \) is the lower calorific value of the fuel.

2.1.5 Gas Turbine Model

Turbine produced power to drive engine compressor due to expansion of gas stream. A general model that represents the gas turbine with turbine blade cooling has been developed. The model is intended for use in cycle analysis applications.

Work compatibility of turbine and compressor is given as follows:

\[ W_c + W_{\text{aux}} = \eta_m W_t \]  
(19)

\( W_{\text{aux}} \), auxiliary power, required for powering accessories is tapped out from the high pressure turbine. It is assumed to be 10% of the compressor work. Since

\[ W_t = \frac{W_c}{\eta_m} \times 1.10 \]  
(20)

Here \( W_t \) is the work developed by high pressure turbine, \( W_c \) is the work required by the high pressure compressor and \( \eta_m \) is the mechanical efficiency which is used to account for the losses due to windage, bearing friction, and seal drag and is defined as

\[ \eta_m = \frac{\text{mechanical power output}}{\text{mechanical power input}} \]

It is to be noticed that the auxiliary work taken should be balanced only once with high pressure turbine work.

Mass and energy balances yield the turbine work, as given below.

Mass balance for the gas turbine component will be given by

\[ \dot{m}_{\text{gas}, i} = \dot{m}_{\text{gas}, e} + \sum \dot{m}_{cl} \]  
(21)

where
\( \dot{m}_{\text{gas,i}} = \dot{m}_{\text{air,i}} + \dot{m}_{\text{f,b}} \) \hspace{1cm} (22)

and \( \sum \dot{m}_{\text{cl}} \) represents the sum of air coolant flow rates to the stage (stator and rotor).

Energy balance in terms of enthalpy is given as

\[
\dot{m}_{\text{gas,i}} \cdot h_{\text{gas,i}} + \dot{m}_{\text{cl,i}} \cdot h_{\text{cl,i}} = \dot{m}_{\text{gas,e}} \cdot h_{\text{gas,e}} + W_t
\]

(23)

2.1.6 Cooling Model

The cooling of turbine nozzle guide vanes or blades and the rotor is required to limit turbine blade temperature within the metallurgical limits of the turbine blade material. The cooling air is drawn off from the high-pressure compressor. It is used to cool the high-pressure and low-pressure turbine nozzle guide vanes or blades as well as their respective rotors. Due to this air-cooling, there will be mixing and heat transfer losses in the expansion path and as a result the turbine exhaust temperature will be affected. The turbine blade cooling model is based on the work of Horlock et al. [12] and Sanjay et al. [13]. Each stage of turbine is divided into two cooled elements- stator row and rotor row.

Transpiration cooling is considered in the present work. Fig. 2 shows the simple model for transpiration cooling.

For an internally cooled turbine configuration, the ratio of coolant to main gas flow rates \( \left( \frac{\dot{m}_{\text{cl}}}{\dot{m}_g} \right) \) required to cool the gas to \( T_b \) at the blade surface is proportional to the difference of enthalpy, which drives the heat transfer from gas to the blades to the ability of the coolant to absorb heat, which is also termed as cooling factor \( R_c \).

Thus

\[
\frac{\dot{m}_{\text{cl}}}{\dot{m}_g} = \frac{\text{Heat transfer to blades}}{\text{Ability of coolant to absorb heat}} \propto \frac{h_{\text{cl,i}} - h_{\text{b,i}}}{h_{\text{g,i}} - h_{\text{cl,i}}}
\]

(24)

A concept of isothermal effectiveness for transpiration air-cooling \( (\eta_{\text{iso}})_{\text{trans}} \) is introduced and the resulting cooling factor is given by

\[
(R_c)_{\text{trans}} = \frac{(T_{\text{g,i}} - T_{\text{b,i}}) \cdot c_{p,g} \cdot (1 - (\eta_{\text{iso}})_{\text{trans}})}{\varepsilon (T_{\text{b,i}} - T_{\text{cl,i}}) \cdot c_{p,cl}}
\]

(25)

The value of \( (\eta_{\text{iso}})_{\text{trans}} = 0.5 \)

For air transpiration cooling, the value of \( c_{p,cl} \) will be taken for that air. Thus the cooling requirement for transpiration cooling is given as follows:

\[
\frac{\dot{m}_{\text{cl}}}{\dot{m}_g} = S_{\text{in}} \cdot \left[ \frac{S_g}{\tan \alpha} \cdot F_{\text{sa}} \right] \cdot (R_c)_{\text{trans}} \cdot 0.0156 \cdot (R_c)_{\text{trans}}
\]

(26)

The total pressure losses in mixing of coolant and mainstream are expressed as

\[
\frac{\Delta p}{p} = 0.07 \frac{\dot{m}_{\text{cl,i}}}{\dot{m}_{\text{g,i}}}
\]

(27)
Transpiration air-cooling model for a single row of gas turbine

Mass of coolant required per kg of gas flow is given by following equations

\[
\frac{m_{cl}}{m_g} = 0.0156 \left[ R_c \right]_{trans} \tag{28}
\]

where

\[
(R_c)_{trans} = \frac{(T_{g,i} - T_{bl}) \cdot \eta_{iso} \cdot (\gamma_{cl} - 1)}{(T_{bl} - T_{cl,i}) \cdot \eta_{iso}} \tag{29}
\]

The total pressure loss in mixing of coolant and mainstream are taken as 2% in this work.

2.1.7 Nozzle Model

The jet nozzle suffers from the aerodynamic losses mainly due to skin friction, which is modeled by introducing the concept of nozzle efficiency [14]. The nozzle may be choked or unchoked and may be decided using the critical pressure ratio.

**Choked Nozzle and Unchoked Nozzle:** If \(\frac{P_{oi}}{P_a} > \frac{P_{oi}}{P_c}\), then nozzle is choked. In this condition the calculation is based on the following equations.

The jet velocity is given by

\[
C_{n,e} = \sqrt{\left(\frac{\gamma_g}{\gamma_g - 1} \cdot R_g \cdot T_c \right)} \tag{30}
\]

The mass flow rate is expressed by

\[
m_{g,e} = \rho_{n,e} \cdot A_{n,e} \cdot C_{n,e} \tag{31}
\]

If \(\frac{P_{oi}}{P_a} < \frac{P_{oi}}{P_c}\) the nozzle is unchoked and the jet velocity is given by

\[
C_{n,e} = \sqrt{\left[2c_{pg} \cdot (T_{0,i} - T_{0,e})\right]} \tag{32}
\]

2.1.8 Governing Equations for performance prediction and software developments

Following equations are used to calculate specific thrust and thrust specific fuel consumption of a turbojet engine. The cycle performance and optimization are achieved through software
developed in “C++” language and then graphs plotted in the menu driven software “origin 61”. Under different nozzle conditions it is expressed as given below:

If nozzle is choked, the net specific thrust comprise by the two components, namely

Momentum thrust given by

\[ F_m = \dot{m}_e (C_{n,e} - C_a) \]  (33)

Pressure thrust given by

\[ F_p = A_{n,e} (p_{0,e} - p_a) \]  (34)

Therefore the net thrust \( F_s \) is given by the following equation

\[ F_s = F_m + F_p \]  (35)

If nozzle is unchoked, net specific thrust is directly expressed as

\[ F_s = \dot{m}_e C_{n,e} \]  (36)

Thrust Specific Fuel Consumption is expressed as

\[ TSFC = \frac{\dot{m}_e}{F_s} \]  (37)

3. RESULTS AND DISCUSSIONS

<table>
<thead>
<tr>
<th>S. No.</th>
<th>Symbols</th>
<th>Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>TA (Atmospheric Temperature)</td>
<td>288.15 K</td>
</tr>
<tr>
<td>2</td>
<td>PA (Atmospheric Pressure)</td>
<td>1.01325 bar</td>
</tr>
<tr>
<td>3</td>
<td>R (Gas constant for air)</td>
<td>287 kJ/kgK</td>
</tr>
<tr>
<td>4</td>
<td>ETAD (Diffuser Efficiency)</td>
<td>0.95</td>
</tr>
<tr>
<td>5</td>
<td>ETAF (Fan Efficiency)</td>
<td>0.87</td>
</tr>
<tr>
<td>6</td>
<td>ETAC (Compressor Efficiency)</td>
<td>0.87</td>
</tr>
<tr>
<td>7</td>
<td>ETAP (Turbine Efficiency)</td>
<td>0.87</td>
</tr>
<tr>
<td>8</td>
<td>RG (Gas constant for gas)</td>
<td>0.296 J/kgK</td>
</tr>
<tr>
<td>9</td>
<td>ETAM (Mechanical Efficiency)</td>
<td>0.99</td>
</tr>
<tr>
<td>10</td>
<td>TRS (Maximum number of stages)</td>
<td>25</td>
</tr>
<tr>
<td>11</td>
<td>DOR (Degree of reaction)</td>
<td>0.5</td>
</tr>
<tr>
<td>12</td>
<td>C (cooling factor)</td>
<td>0.03</td>
</tr>
<tr>
<td>13</td>
<td>ETACL (Cooling Efficiency)</td>
<td>0.8</td>
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<tr>
<td>14</td>
<td>TBL (Blade Temperature)</td>
<td>1122 K</td>
</tr>
<tr>
<td>15</td>
<td>CCL (Cooling constant)</td>
<td>2</td>
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<tr>
<td>16</td>
<td>MA (Mass of air rate flow)</td>
<td>1 kg/s</td>
</tr>
<tr>
<td>17</td>
<td>ETAB (Blade efficiency)</td>
<td>0.98</td>
</tr>
<tr>
<td>18</td>
<td>ETAJ (Nozzle Efficiency)</td>
<td>0.97</td>
</tr>
<tr>
<td>19</td>
<td>CV (Calorific Value of Fuel)</td>
<td>43124 kJ/kg</td>
</tr>
<tr>
<td>20</td>
<td>( \Delta p ) (Pressure drop in Combustion chamber)</td>
<td>2%</td>
</tr>
<tr>
<td>21</td>
<td>M (Mach Number)</td>
<td>0.9 - 1.5</td>
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<tr>
<td>22</td>
<td>OPR (Overall Compression Pressure Ratio)</td>
<td>10 – 30</td>
</tr>
<tr>
<td>23</td>
<td>TIT (Turbine Inlet Temperature)</td>
<td>1000 – 1500K</td>
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</table>
In the following sections the results of the present work have been discussed in the form of graphs plotted in the menu driven software “Origin 51”. All the results are shown for various parameters of Altitudes, Turbine inlet temperature and overall pressure ratio. The variations of Specific thrust, Thrust specific fuel consumption and mass of fuel required in the combustion chamber have been plotted with respect to the turbine inlet temperatures.

**Figure 3:** Variations of Specific thrust with TIT

Figure 3 represents the variation of Specific thrust with turbine inlet temperature for a given Mach number and Altitude for various overall pressure ratios. On increasing the overall pressure ratio of the compressor, the specific thrust increases for a given value of TIT.

**Figure 4:** Variations of Mass of fuel required in combustion chamber with TIT
Figure 5: Variations of Thrust specific fuel consumption with TIT

Figure 4 represents the variations of mass of fuel required in the combustion chamber with TIT for a given low Mach number and low altitude. On increasing the pressure ratio the fuel required in the combustion chamber decreases due to increase in temperature of inlet air before entering into the combustion chamber.

Figure 5 shows the variations of Thrust specific fuel consumption with TIT for a given Mach number and Altitude but different pressure ratio. The thrust specific fuel consumption decreases on increasing the TIT in the low range say 1000 to 1200 K. After that range of temperature, the thrust specific fuel consumption increases on increasing the TIT for a given pressure ratio.

Figure 6: Variations of Mass of coolant required with TIT

Figure 6 represents the variations of Mass of coolant required per kg of air flow with TIT. It is clearly shown from the figure that the turbine blades are safe up to temperature 1123 K. When the temperature reached above that value, the coolant required to cool the turbine blade and increases the life.

Figure 7 represents the variations of Specific thrust with TIT for various Mach number. At higher Mach number or for sonic velocity, the specific thrust is higher as compares to the subsonic velocity but in the higher turbine inlet temperature range.
Figure 7: Variations of Specific thrust with TIT

Figure 8: Variations of Mass of fuel required in combustion chamber with TIT

Figure 8 represents the variations of mass of fuel required in the combustion chamber with TIT for different Mach numbers and low altitude. On increasing the mach number the fuel required in the combustion chamber decreases.

Figure 9: Variations of Thrust Specific Fuel Consumption with TIT
Figure 9 shows the variations of Thrust specific fuel consumption with TIT for different Mach number and a given Altitude and pressure ratio. The thrust specific fuel consumption decreases on increasing the Mach number.

4. CONCLUSION

A parametric thermodynamic study has been carried out for a single spool turbojet engine presenting a summary of point design performance estimating using simplified component technology assumptions. The present work may be concluded as follows:

1) The results obtained for single spool turbojet engine shows high specific thrust produces at higher Mach number for a given Turbine Inlet Temperature.

2) Specific thrust is higher for a given Mach number and turbine inlet temperature on increasing the pressure ratio of the compressor.

3) Mass of fuel required in the combustion chamber decreases on increasing the pressure ratio for a given value of TIT.

4) At any altitude the TSFC first decreases at lower TIT and then increases with increase in TIT and Mach number, whereas reverse is true with increase in altitude at any Mach number.

5) Mass of cooling air required will depend on the TIT. The cooling required after a certain temperature of gases entering into the turbine blade. The cooling is necessary when the temperature of the gases reached above metallurgical limit of the turbine blade, therefore it is important to use cooling air above 1123 K to increases the life of the turbine blade.

5. REFERENCES


AUTHOR’S DETAIL

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