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MODELING PERFORMANCE CHARACTERISTICS OF A TURBOJET ENGINE

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ABSTRACT: *This work is an approach to analyze the performance of jet engine with an alternative flow control mechanism. The work analyzes the feasibility of a thrust control mechanism by regulating the flow at the engine inlet to increase the engine rpm for the same value. This approach adds value to the design process with the aim of increasing performance levels in the engine operation. The work is focused on performance charactristics of turbojet with reduced inlet pressure to the compressor engine. A real-time turbojet engine integrating aerothermodynamics of engine components, was developed, together with principles of jet propulsion and inters component volume dynamics represented in 1-D non-linear equations. Software program Simulink, a commercially available model based graphical block diagramming tool from MathWorks was used for dynamic modeling of the engine. The result shows that with increase in shaft rpm, pressure and temperature ratio values across the compressor-turbine assembly increases. Performance parameters of the engine are analyzed with the increase in compressor pressure ratio and shaft rpm.*

KEYWORDS: Turbojet, Specific Thrust, Specific Fuel Consumption, Thermal Efficiency, Propulsive Efficiency, Modeling

INTRODUCTION

Jet Engine is the gas turbine application for aircraft propulsion. Basic principle in a jet engine is to accelerate a mass of fluid in the direction opposite to motion and thereby propelling the aircraft forward by the thrust generated. The engine sucks air in at the front with a fan. A compressor raises the pressure of the air. The compressor is made up of fans with many blades and attached to a shaft. The blades compress the air. The compressed air is then sprayed with fuel and an electric spark lights the mixture. The burning gases expand and blast out through the nozzle, at the back of the engine. As the jets of gas shoot backward, the engine and the aircraft are thrust forward **[1].** Schematic diagram of a Turbojets engine is shown in figure 1.

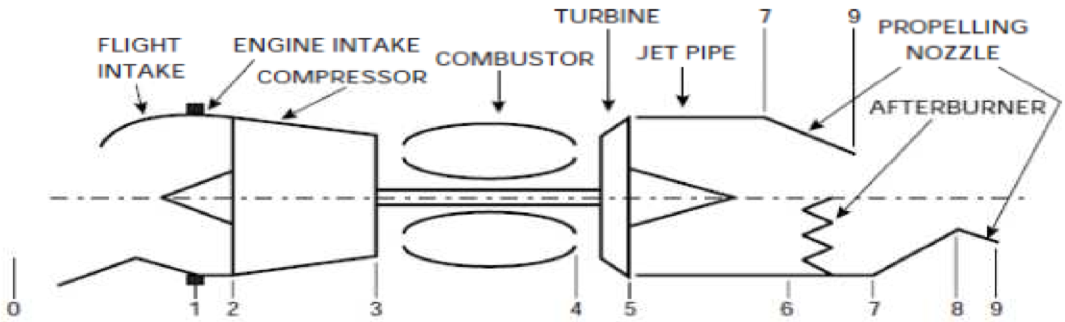

Figure 1: Turbojet Engine Configurations [2]

At point 0 ambient air passes from free stream to the flight intake leading edge at 1 and the air accelerates from free stream if the engine is static, whereas at high flight Mach number it diffuses from the free stream, ram conditions. Usually, it then diffuses at point 2 in the flight intake before passing through the engine intake to the compressor face with a small loss in total pressure. The compressor then increases both the pressure and temperature of the gas. Work input is required to achieve the pressure ratio; the associated temperature rise depends on the efficiency of the compressor. The compressor exit diffuser at point 3 passes the air to the combustor. Here, fuel is injected and burnt to raise exit gas temperature at point 4. The diffuser and combustor both impose a small total pressure loss. The hot, high pressure gas is then expanded through the turbine where work is extracted to produce shaft power; both temperature and pressure are reduced. The shaft power is that required to drive the compressor and any engine auxiliaries. On leaving the turbine at point 5, the gas is still at a pressure typically at least twice that of ambient. This results from the higher inlet temperature to the turbine. Downstream of the turbine the gas diffuses in the jet pipe at point 6. This is a short duct that transforms the flow path from annular to a full circle at entry to the propelling nozzle at point 9. The jet pipe imposes a small total pressure loss. At point 7 afterburner is introduced, an afterburner is a combustor located downstream of the turbine blades and directly upstream of the nozzle, which burns fuel from afterburner-specific fuel injectors. The propelling nozzle is a convergent duct that accelerates the flow to provide the high velocity jet to create the thrust. Engine cooling system uses the relatively cool air from the compression system that bypasses the combustor via air system flow paths to cool the turbine nozzle guide vanes and blades to ensure acceptable metal temperatures at elevated gas temperatures.

For high flight Mach number applications an afterburner is often employed, which offers produces higher thrust from the same configuration. This is also called reheat, and involves burning fuel in an additional combustor downstream of the turbine. Turbojets are quite inefficient compared to other engine types at lower Mach numbers but has dominant role for the supersonic flight modes and military applications.

MODELING OF THE COMPONENTS

The essence of modeling is to achieve the following:

• Steady State performance obtained from cycle calculations derived from component maps obtained through detailed component modeling and component tests
• Corrected parameter techniques used to reduce the number of points that need to be evaluated to estimate engine performance throughout the operating envelope
• Dynamics modeled through inertia (the rotor speeds), combustion delays, heat soak and sink modeling etc.
• Computationally intensive process since it is important to maintain mass/momentum/energy balance through each component
• Detailed thermo-dynamic cycle decks developed and parameters adjusted to match engine test results
• Simplified models generated to develop and evaluate control design

Dynamic behavior of single-shaft turbojet was first studied at NACA Lewis Laboratory in 1948.

The study showed that the transfer function from fuel flow to engine speed can be represented by a first order lag linear system with a time constant which is a function of the corrected fan speed: N(s)/WF(s) = K/(as+1) with a=f(N) **[3]**

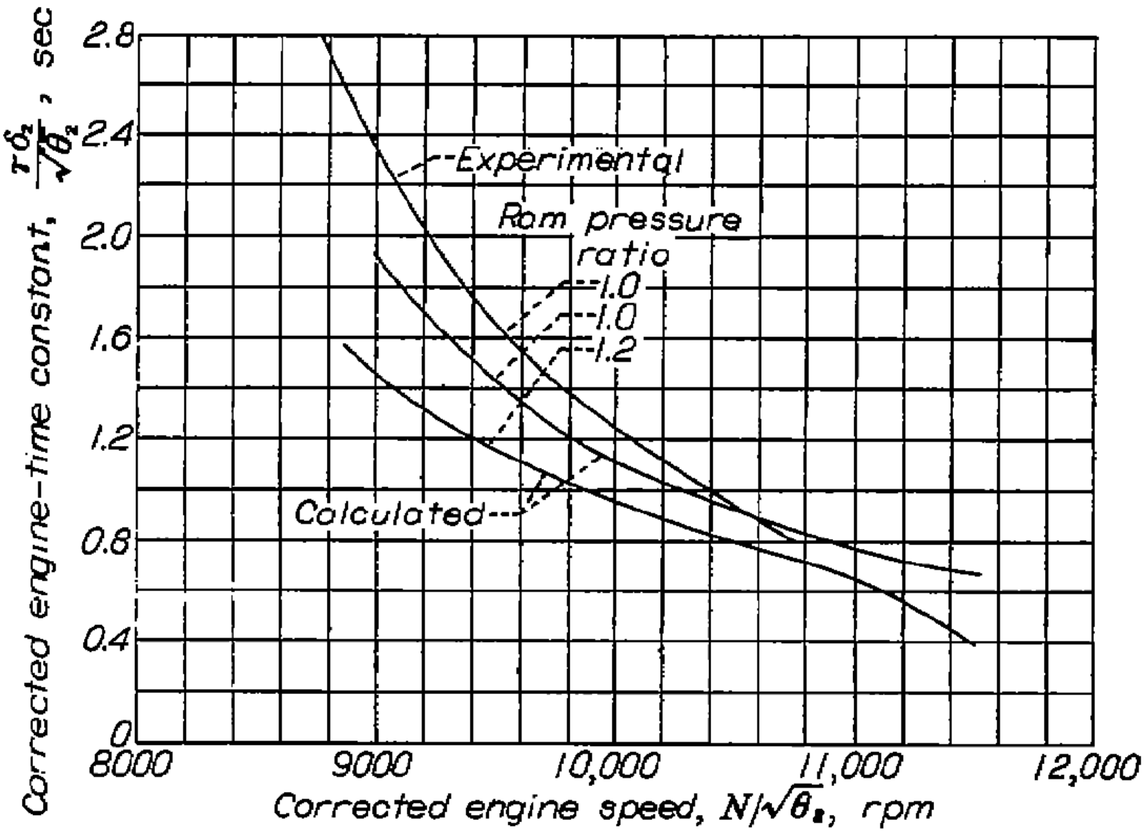

Figure 2 Plot of corrected engine time constant with corrected engine speed. [4]

A basic turbojet engine with an inlet, axial compressor, combustor, axial turbine and a converging nozzle is built in Simulink. The engine model is based on component volume dynamics. Isentropic processes are assumed for the diffuser and the converging nozzle. Every component is described by the steady state characteristics, followed by a fictitious volume where mass and energy storage takes place. Figure 6 shows the station numbering of the mathematical model for the engine.

Figure 3 shows the station numbering of the mathematical model for the engine.

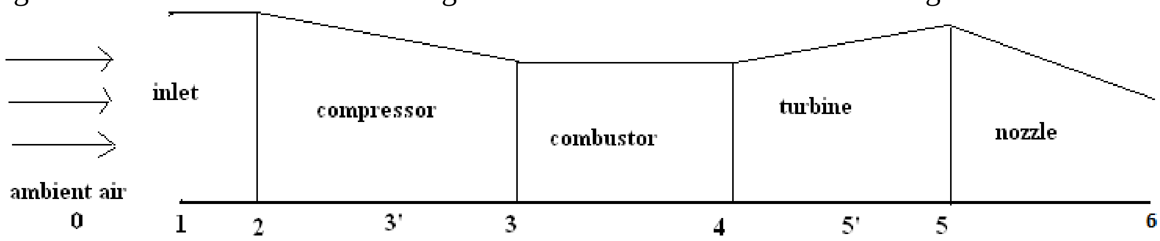

Figure 3: Station Numbering for the Engine Model

A mathematical model, based on the lumped parameter approach is used from the unsteady one dimensional conservation laws described by a set of first-order differential and algebraic equations.

Inlet

Intake diffuser is used to bring the free stream air into the engine. It does no work on the fluid but it guides the fluid flow to the compressor. However, the performance of the inlet is defines by the pressure recovery from the free stream to the engine. An isentropic process is assumed for the air flow in the inlet diffuser. Heat transfer and the friction between the air and the diffuser walls are not considered. The stagnation values of the temperature and pressure at the diffuser exit are calculated from the equations, **[5]**

$$T_2 = T_0 \quad (1)$$

$$P_2 = \eta_1 * P_0 \quad (2)$$

Pressure recovery in the diffuser is calculated using the Nigeria Air force standard **[5]**

$$\eta_1 = 1.0 \text{ if } M \leq 1 \quad (3)$$

$$\eta_1 = 1.0 - 0.075*[(M-1)^{1.35}] \text{ if } M > 1 \quad (4)$$

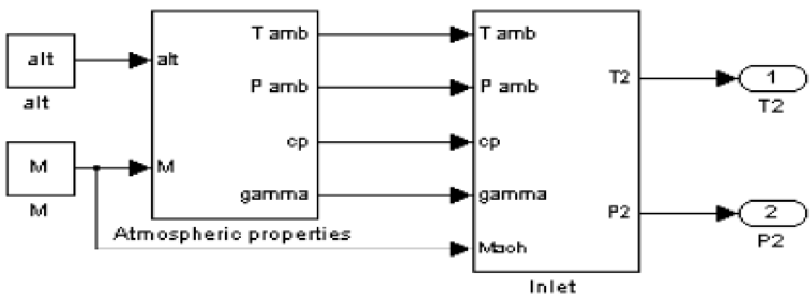

Figure 4: Atmosphere and Inlet model in Simulink

Compressor

The purpose of a compressor is to increase the total pressure of the gas stream to that required by the engine while absorbing the minimum shaft power possible. Temperature of the incoming air also increases with pressure in the compressor. The work done by the compressor on the gas is extracted from the turbine. In the reference engine model, the compressor is an axial compressor with 8 stages. But while modeling the engine in Simulink, compressor shall be modelled as a single block by stacking all the stages of the compressor into a single block. Dynamic behaviour of all the individual stages is also stacked into a single block with only the inlet and final exit conditions of the compressor.

The final temperature of the compressor is calculated from the equation from the equation. **[6]**

$$T_{3'} = T_2 \left(1 + \frac{\left(\frac{P_3}{P_2}\right)^{\frac{\gamma-1}{\gamma}} - 1}{\eta_{Is \leftrightarrow comp}} \right) \tag{5}$$

Where γ_a = 1.4 and γ_g = 1.33

Work done by the compressor to increase the pressure and temperature of the air can be calculated from the expression. **[3]**

$$w_c = \dot{m}_3 * c_p * (T_3 - T_2) \tag{6}$$

The value for the $\mathbf{C_p}$ in the above expression is calculated at the interpolated temperature value for the compressor given by **[6]**

$$T_c = \beta_c * T_3 + (1 - \beta_c) * T_2 \tag{7}$$

Compressor Volume Dynamics

In the transient process, the compressor is modeled as a mixing volume in which the mass and energy can be accumulated. The gas dynamics associated in the compressor stages are calculated by applying the continuity, energy and Ideal gas equations to the inter component volume between the compressor and the combustor.

Continuity equation: [7]

$$\frac{d}{dt}(w_3) = \dot{m}_3 - \dot{m}_3 - Bld \tag{8}$$

Energy equation: **[7]**

$$\frac{d}{dt}(T_3) = \frac{1}{w_3 c_v} \left\{ \left[\dot{m}_3 * \left(C_p T_3 - C_v T_3 \right) \right] - \left[\left(\dot{m}_3 + Bld \right) * R * T_3 \right] \right\} \tag{9}$$

Ideal Gas equation: **[7]**

$$P_3 = \frac{w_3 * R * T_3}{V_3} \quad (10)$$

These mathematical equations are used for modeling the inter component volume for the compressor in Simulink.

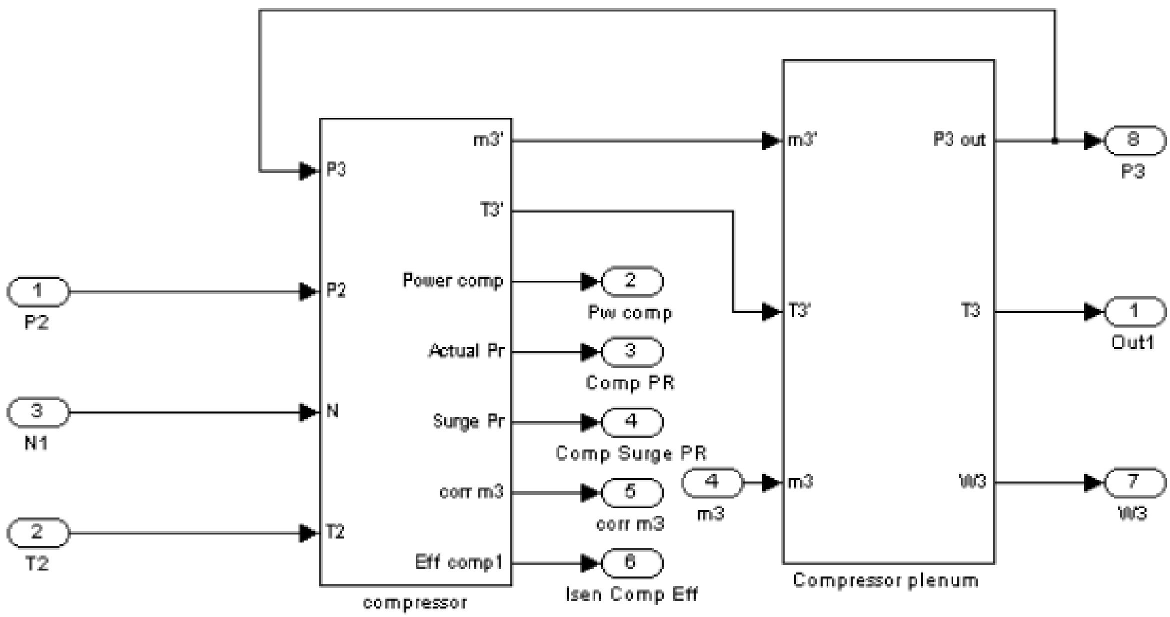

Figure 5: Simulink Model for Compressor

Combustor

Combustion process in this engine model is simplified for the performance calculations and assumed to be instantaneous and complete after the fuel injection to the combustor. However, the actual process of burning the fuel in the combustor is a complex process. After the injection of the fuel into combustor, fuel droplets are engulfed into the air stream and transported to the burning zone. During transportation, the fuel droplets are heated and evaporated due to surrounding air. Then the fuel vapours diffuse and mix with the hot air and then burn at a finite rate. All the factors like mixing of the fuel vapours to the air, combustion rate and combustion efficiency are considered for combustor modeling.

Pressure Drop in the Combustor

Unlike the theoretical combustion process where the inlet pressure and outlet pressure of the chamber are equal, there is a pressure drop in the combustion for the actual process. Because of this pressure drop, the gases can flow in the correct direction and mass flow rate through the combustion chamber is calculated. Expression for the pressure drop across the combustor is given by, **[8]**

$$\dot{m}_b = \dot{m}_3 = \sqrt{\frac{P_3 - P_4}{R_b}} \tag{11}$$

Where $\mathbf{R_b}$ is the calculated combustor pressure loss coefficient and is calculated from the steady state values of the engine at various speeds.

Combustor Volume Dynamics

The combustor is lumped into a single equivalent one dimensional volume. Model is developed in Simulink from the mathematical expressions of the continuity, energy equation and the ideal gas equation along with the algebraic expression for pressure drop in the combustor.
Continuity equation: **[9]**

$$\frac{d}{dt}\left(w_4\right) = \dot{m}_b - \dot{m}_f - \dot{m}_4 \tag{12}$$

Energy balance: **[6]**

$$\frac{d}{dt}\left(T_4\right) = \frac{1}{w_4 c_{vb}}\left\{\left[\dot{m}_3 * \left(C_{pb}T_b - C_{vb}T_4\right)\right] - \left[\dot{m}_f * \left(HVF * \eta_b - C_{vb}T_4\right)\right]\left[\dot{m}_4 * R * T_4\right]\right\} \tag{13}$$

Ideal Gas equation: **[6]**

$$P_4 = \frac{w_4 * R * T_4}{V_4} \tag{14}$$

where T_b in equation (13) is the combustor interpolation constant calculated from the expression: **[9]**

$$T_b = \beta_c * T_4 + (1 - \beta_c) * T_4 \tag{15}$$

The values for the specific heats of the combustor are calculated at the interpolated temperature of the combustor.

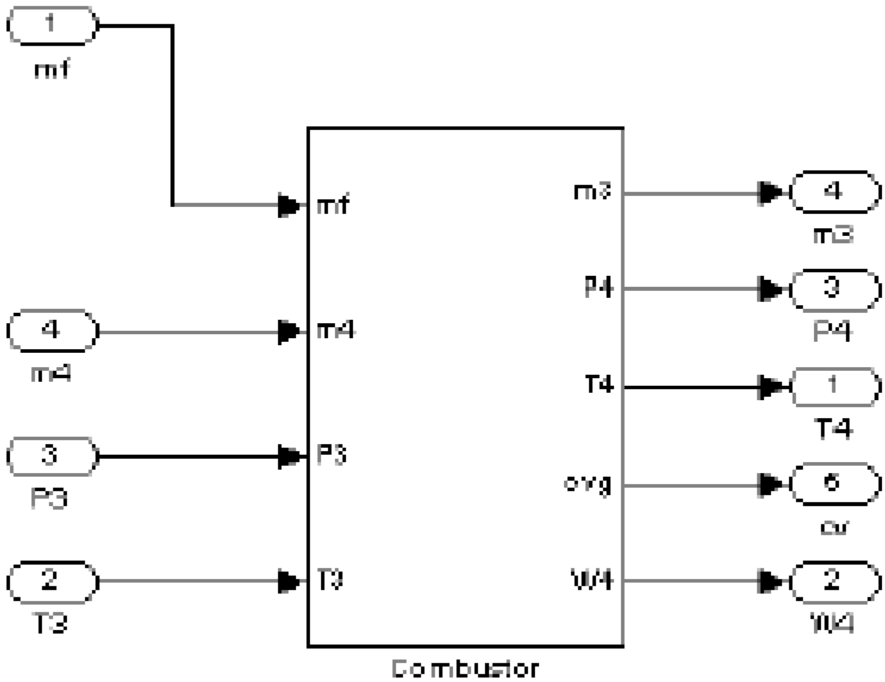

Figure 6: Simulink Model for Combustor

Turbine

Turbine is used to extract sufficient energy from the hot gases of the combustor to drive the compressor and other auxiliary power equipment. In the reference engine model, the turbine is an axial turbine with 2 stages. But while modeling the engine in Simulink, turbine is modeled as a single block by stacking the two stages of the turbine into a single block. Dynamic behavior of all the individual stages is also stacked into a single block with only the inlet and final exit conditions of the turbine.

The final temperature of the turbine is calculated from the equation. **[9]**

$$T_{5'} = T_4 * \left\{ 1 - \eta_{Is\_com} * \left[1 - \frac{1}{\left(\frac{P_3}{P_2} \right)^{\frac{\gamma-1}{\gamma}}} \right] \right\} \quad (15)$$

Work done on the turbine by the hot gases from the combustor can be calculated from the expression: **[6]**

$$w_t = \dot{m}_5 * c_{pb} * (T_5 - T_4) \quad (16)$$

Turbine Volume Dynamics

In the transient process, the turbine is modeled as mixing volume in which the mass and energy can be accumulated. The gas dynamics associated in the turbine stages are calculated by applying the continuity, energy and the Ideal gas equations to the inter component volume between the turbine and the convergent nozzle.

Continuity equation: **[9]**

$$\frac{d}{dt}\left(w_5\right) = \dot{m}_5 - \dot{m}_5 + Bld \tag{17}$$

Energy equation: **[6]**

$$\frac{d}{dt}\left(T_5\right) = \frac{1}{w_5 c_{vb}}\left\{\left[\dot{m}_5 * \left(C_{pb} T_{5'} - C_v T_5\right)\right] - \left[\dot{m}_5 * R * T_5\right]\right\} \tag{18}$$

Ideal Gas equation: **[6]**

$$P_5 = \frac{w_5 * R * T_5}{V_5} \tag{19}$$

Turbine inter component volume is built in Simulink using these mathematical equations.

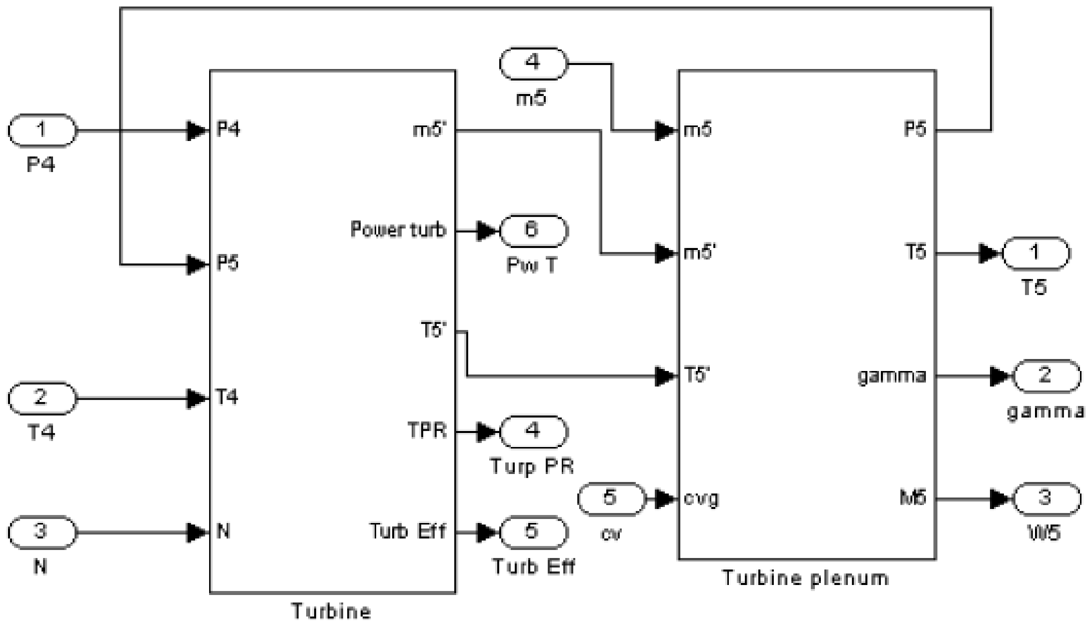

Figure 7: Simulink Model for Turbine

Nozzle

A convergent nozzle is considered in modeling the engine. The partially expanded gas coming from the turbine at a relatively high pressure is accelerated to a high velocity in the nozzle. Finally, the gases expand to the ambient pressure and provide the thrust to propel the aircraft. Gas flow in the nozzle is considered to be quasi steady state and so the dynamics in the nozzle are not considered.

Modeling of the convergent nozzle for the engine is based on the following mathematical equations. The mass flow rate through the nozzle depends on two factors: nozzle back-pressure P_{back} and the nozzle exit critical pressure $\mathbf{P_{cr}}$.

For a specific inlet pressure to the nozzle, there exists a critical back-pressure defined by the expression. **[10]**

$$P_{cr} = \left[\frac{2}{\gamma_b + 1}\right]^{\frac{\gamma - 1}{\gamma}} P_{inlet} \tag{20}$$

Depending on the value of the critical pressure the flow through the nozzle is defined below

Case:1: If the back pressure is greater than the critical pressure value, the flow is subsonic at the exit and the exit pressure is equal to the back pressure. The flow rate, exit velocity and the Thrust produced is calculated with the expressions below

If P_b > Per,

Exit pressure from the nozzle, [10]

$$\mathbf{P_{exite} = P_0} \tag{21}$$

Mass flow rate in the nozzle, [11]

$$\dot{m}_5 = \frac{P_5}{\sqrt{RT_5}} A_{nz} \left[\frac{P_e}{P_5}\right]^{\frac{1}{\gamma_b}} \sqrt{\frac{2\gamma_b}{\gamma_b - 1}\left[1 - \left(\frac{P_e}{P_5}\right)\right]^{\frac{\gamma_b - 1}{\gamma_b}}} \tag{22}$$

Thrust produced, [8]

$$Thrust = c_v \dot{m}_5 \sqrt{2 c_p T_5 \left[1 - \left(\frac{P_e}{P_5}\right)^{\frac{\gamma_b - 1}{\gamma_b}}\right]} \tag{23}$$

Jet velocity at the exit of the nozzle, **[8]**

$$V_e = \sqrt{\frac{2\gamma_b}{\gamma_b - 1} R T_5 \left[1 - \left(\frac{P_e}{P_5}\right)^{\frac{\gamma_b - 1}{\gamma_b}}\right]} \tag{24}$$

Case:2: If the back pressure is less than the critical pressure, the flow is sonic or choked at the exit and the exit pressure is equal to the critical pressure. Flow rate, exit velocity, and the Thrust produced is calculated from the expressions below

If $P_b < P_{cr}$

Exit pressure from the nozzle, [11]

$$P_{exite} = P_0 \tag{25}$$

Mass flow rate in the nozzle, [11]

$$\dot{m}_5 = \frac{P_5}{\sqrt{RT_5}} A_{nz} \sqrt{\gamma_b \left(\frac{\gamma_b}{\gamma_b + 1} \right)^{\frac{\gamma_b}{\gamma_b - 1}}} \tag{26}$$

Thrust produced, [8]

$$Thrust = c_v \dot{m}_5 \sqrt{2 c_p T_5 \left[1 - \left(\frac{P_e}{P_5} \right)^{\frac{\gamma_b - 1}{\gamma_b}} \right]} + A_{nz} \left[P_{cr} - P_e \right] \tag{27}$$

Jet velocity at the exit of the nozzle, **[11]**

$$V_e = \sqrt{\frac{2\gamma_b}{\gamma_b - 1} R T_5 \left[1 - \left(\frac{P_e}{P_5} \right)^{\frac{\gamma_b - 1}{\gamma_b}} \right]} \tag{28}$$

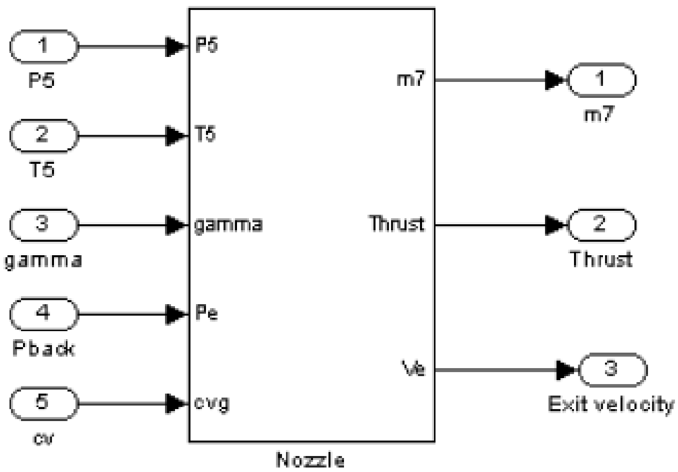

Figure 8: Simulink Model for Nozzle

Rotor Dynamics

The steady state performance of a turbojet engine matches the operating points of the compressor with that of turbine. A mismatch between these components produces unbalanced torque that is integrated with the dynamic relations to find the new steady state match. Therefore, the change of the rotor speed is a function of the energy differential of the work extracted by the turbine and the work done by the compressor given by,

$$\frac{d}{dt}\{N\}=\left[\frac{30}{\pi}\right]^2\frac{1}{IN}\left[PW_{turb}-PW_{comp}\right] \tag{29}$$

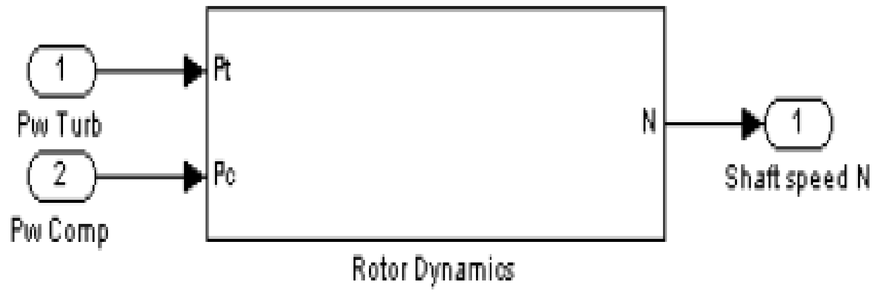

Figure 9: Rotor dynamics model in Simulink

Fuel Controller

The transient response in the engine is due to the response from the fuel controller. The control is set according to the output demand for each case. The control is set according to the output demand for each case.

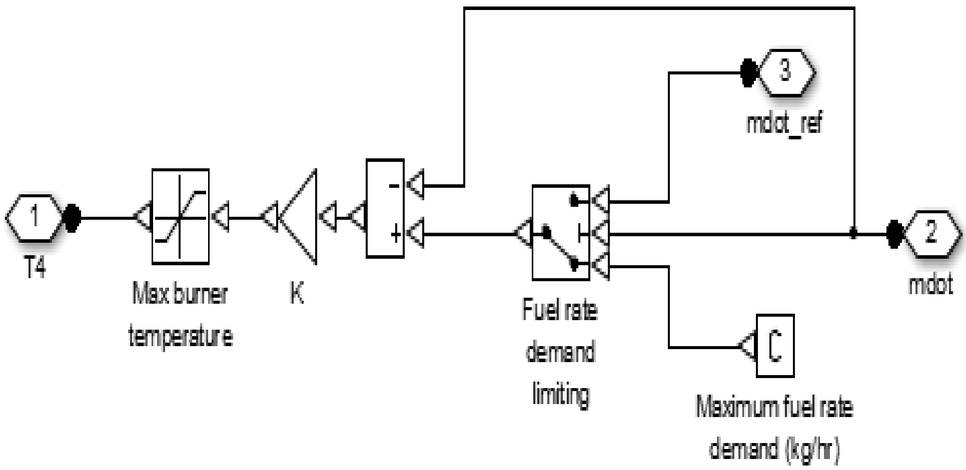

Figure 10: Simulink model for fuel controller

OBSERVATION

It is observed that the thermal efficiency and fuel flow rate do not have a significant change in lower thrust range of operations. But the engine has the advantage of operating at a higher rpm. This particular application can be applied to any aircraft engine during the time of descent and landing. There will be cases where the aircraft need to gain maximum power from lower thrust range to a maximum thrust range in the minimum time. For an example, consider the case where the aircraft is in its approach to landing. Engine at this point operates in the lower thrust range. At the times like a runway overshoot or a go-around landing case, the engine needs to regain its maximum thrust from the lower range in the minimum amount of time. It is at this particular case, the concept of flow control with reduced inlet pressure has a real-time application. Starting at the time of descent the flow control mechanism to reduce compressor inlet pressure is employed and the efficiency of the engine doesn't change much with this application. But the engine is operated at a higher rpm than normal operation with pressure drop across the inlet. If the engine needs to regain the maximum power for the cases as discussed, it could be achieved faster than the normal operation just by deploying the flow control mechanism.

CONCLUSION

With the reduction in the inlet pressure, the engine works at a higher rpm to produce the same thrust. This results with increase in the work output but decrease in the net work. Therefore, even with the increase in the compressor pressure ratio and turbine inlet temperature, the engine thermal efficiency goes down in a small proportion. With the higher exit jet velocities, propulsive efficiency of the engine also goes down in a very small proportion. SFC, Thermal and propulsive efficiencies are the major performance parameters for commercial engines. But for military engines, situations demand higher Specific Thrust.

Specific thrust for engine increases with reduced compressor inlet pressure. This is particularly important for the cases of shorter runways and higher climb rate.

Thrust specific fuel consumption value increases with reduced inlet pressure. With increase in the specific fuel consumption value for an engine, the range value for the engine decreases. Thermal and Propulsive efficiencies are also decreasing with reduced inlet pressure. However, if the flow control mechanism is employed for certain time of flight such as landing or takeoff, performance decrease in these actors will not affect the engine operation. It is advantageous to employ the reduced compressor inlet pressure approach if the situation demands engine power to efficiency

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