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Performance Study of N+3 Turbofan Engine Model with Several Types of Fuels Using NPSS

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A fixed core engine model is developed in NPSS to study the performance of N+3 technology level turbofan engine. NPSS model is validated against the results obtained from fundamental propulsion equations coded in MATLAB for a conventional Jet-A fuel. The results from NPSS and MATLAB codes for variation in Thrust Specific Fuel Consumption (*TSFC*) with Bypass Ratio (*BPR*) are compared; good agreement is obtained. The validated NPSS is then used to study the performance of N+3 engine using liquid hydrogen and ammonia as fuels. The comparisons for the variation of *TSFC* with *BPR* are presented using alternative fuels.

Nomenclature

<i>TSFC</i>	= Thrust Specific Fuel Consumption
<i>BPR</i>	= Bypass Pressure Ratio
<i>FPR</i>	= Fan Pressure Ratio
<i>NPSS</i>	= Numerical Propulsion Simulation System
<i>NIST</i>	= National Institute of Standards and Technology
<i>LH₂</i>	= Liquid Hydrogen
<i>NH₃</i>	= Ammonia
<i>HHV</i>	= Higher Heating Value
<i>LHV</i>	= Lower Heating Value
<i>F_{net}</i>	= Net Force
<i>T_{t4}</i>	= Turbine Entrance Temperature
<i>OPR</i>	= Overall Pressure Ratio
<i>m_c</i>	= Core Mass Flow Rate

I. Introduction

Turbofan engines can achieve higher thrust level with lower fuel consumption compared to turbojet engines by utilizing some of the energy produced by the turbine to drive a fan. The fan draws large amount of air into the engine and thus yields higher thrust per amount of fuel used. One can increase the size of the fan to get lower Thrust Specific Fuel Consumption (*TSFC*) value. However, increasing the size of fan would create aerodynamic issues since the drag force would increase. Hence, modern turbofan engines are designed to achieve lower *TSFC* values while keeping the aerodynamic drag created by a larger fan as low as possible. Different simulation softwares are used in the aviation engine industry to come up with new engine designs that would have the optimum level of *TSFC*. One such simulation software is NPSS (Numerical Propulsion Simulation System).

NPSS was originally developed by NASA Glenn Research Center to be used for the creation, study and sharing of complete aerothermal-mechanical computer simulation data of propulsion systems.¹ NPSS is a component based object-oriented engine cycle analysis and simulation tool. In NPSS, the model definitions are given through input files. The simulation system has a built-in NIST compliant gas property packages to perform different thermochemistry simulations. It is a sophisticated solver with auto-setup, constraints, and discontinuity handling capabilities. The object-oriented design of NPSS facilitates user-definable elements, functions, and models. There are several published engine cycle data sets on NASA's website that can be utilized to model an engine in NPSS. One example of a NPSS modeled engine is the N+3 generation high bypass geared turbofan engine cycle

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modeled by Carter & Agarwal² using published NASA engine cycle data. The N+3 model architecture used by NASA resembles the one shown in Fig. 1.

NASA N+3 High Bypass Geared Turbo-Fan Model

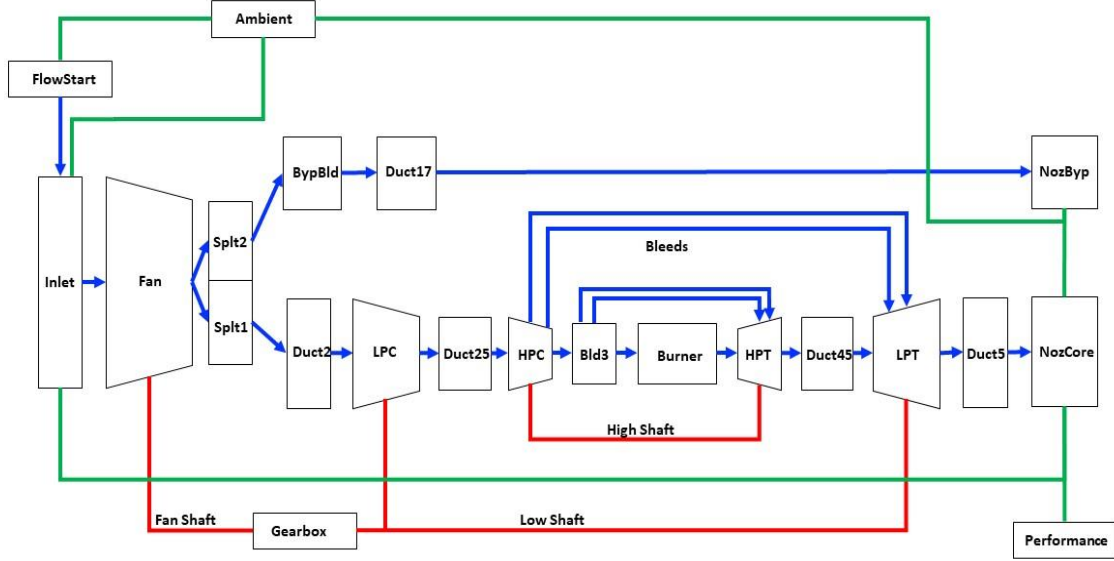


Fig. 1 Schematics of NASA N+3 Turbofan Engine Model².

As shown in Fig. 1, the major components of the N+3 turbofan engine model include a fan, a low-pressure compressor, a high-pressure compressor, a high-pressure turbine, and a low-pressure turbine. According to Ref. 3, the thrust force that is meant to push the aircraft forward is generated by the fan and the core as given in Eq. (1):

$$\begin{aligned} Thrust &= Thrust\ of\ Fan + Thrust\ of\ Core \\ T &= (\dot{m}_f v_f - \dot{m}_f v_o) + (\dot{m}_e v_e - \dot{m}_c v_o) \end{aligned} \quad (1)$$

where T is the thrust in lbf., and \dot{m}_f, \dot{m}_e , and \dot{m}_c are the mass flow rates at the fan exhaust (connected to the bypass bleed), the core exhaust, and the core inlet, respectively in lbm./s, and v_f, v_o, v_e , are the velocities at the fan exhaust (going to the bypass bleed), the fan entrance (free stream velocity), and the core exhaust, respectively in ft./s. The total mass flow of the air is the sum of the mass flow going through the bypass bleed and the mass flow going through the core as given in Eq. (2):

$$\dot{m}_o = \dot{m}_f + \dot{m}_c \quad (2)$$

where \dot{m}_o represents the total mass flow of air entering the engine in lbm./s. The thrust can be adjusted by changing the size of the core and the bleed. Moreover, the ratio between the fan exhaust mass flow rate to the compressor mass flow rate gives the bypass ratio BPR as given in Eq. (3).

$$BPR = \dot{m}_f / \dot{m}_c \quad (3)$$

Equation (1) thus can be written in terms of BPR and \dot{m}_o as given in Eq. (4):

$$T = (\dot{m}_e v_e - \dot{m}_o v_o) + BPR * \dot{m}_c * v_f \quad (4)$$

The net thrust (T_{net}) can now be expressed in terms of the change in velocity between the free stream and the jet (ΔV) as given in Eq. (5):

$$T_{net} = \dot{m} \Delta V \quad (5)$$

where

$$\begin{aligned}\Delta V &= V_{jet} - V_o \quad \text{and} \\ V_{jet} &= M_9 * \alpha.\end{aligned}$$

The \dot{m} in Eq. (5) represents the change in total mass of the aircraft (which includes fuel) overtime. M_9 is the core nozzle exit Mach number and α is the speed of sound calculated by Eq. (6):

$$\alpha = \sqrt{\gamma RT} \quad (6)$$

where γ is the adiabatic index (~ 1.4 for diatomic gas such as air), R is the universal gas constant ($\sim 53.4 \text{ ft}\cdot\text{lb}/\text{lb}\cdot^\circ\text{R}$ for air), and T is the absolute temperature of the air in which the aircraft is flying through. Moreover, the change in kinetic energy of the aircraft can be calculated by Eq. (7) as:

$$\Delta KE = \frac{1}{2} \dot{m} V_{jet}^2 - \frac{1}{2} \dot{m} V_o^2 = \frac{1}{2} \dot{m} \Delta V (2V_o + \Delta V) \quad (7)$$

For a fixed free stream velocity V_o , propulsive efficiency and V_{jet} have an inverse relationship as shown in Eq. (8):

$$\eta_{prop} = \frac{TV_o}{\Delta KE} = \frac{2V_o}{(V_o + V_{jet})} = \frac{2}{(1 + \frac{V_{jet}}{V_o})} \quad (8)$$

Moreover, the FPR , which is the ratio of the fan exit pressure to fan inlet pressure, can be calculated using the nozzle exit Mach number as given in Eq. (9). For an ideal engine, the nozzle exit pressure matches the ambient pressure ($P_\infty = P_1 = P_9$).

$$FPR = \frac{P_2}{P_1} = \frac{P_2}{P_\infty} = \left[1 + \frac{\gamma-1}{2} M_9^2 \right]^{\frac{\gamma}{\gamma-1}} \quad (9)$$

By solving Eq. (9) for M_9 , the nozzle exit Mach number can be expressed in terms of the FPR and the fan inlet pressure (P_1) as given in Eq. (10):

$$M_9 = \sqrt{\frac{2(FPR)^{\frac{\gamma-1}{\gamma}} - 1}{\gamma-1}} \quad (10)$$

If we keep the FPR constant, the exit Mach number also becomes constant leading to a constant exit jet velocity. Thus, the propulsive efficiency depends on only one factor that is the free stream velocity V_o . At constant FPR , the propulsive efficiency increases with increasing V_o . However, increasing V_o indefinitely is counterproductive since it would decrease the net thrust generated by the engine (see Eq. (5)). The Thrust Specific Fuel Consumption ($TSFC$) is better suited to study the performance of a turbofan engine in such cases as it incorporates the amount of fuel used for thrust generation. The equation for calculating $TSFC$ is given in Eq. (11):

$$TSFC = \frac{\dot{m}_{fuel}}{T_{net}} \quad (11)$$

The amount of fuel needed to generate some level of thrust differs based on the type of fuel used. Different fuels have different heating values which is a measure of the amount of energy that can be extracted during combustion. The Higher Heating Value (HHV) indicates the upper limit of the thermal energy produced during combustion while the Lower Heating Value (LHV) indicates the thermal energy produced minus the latent heat of vaporization of water since it assumes the water produced through combustion is in the vapor form⁴. The useful energy content of fuels is therefore best estimated through LHV . The higher the LHV value, higher is the thermal energy that can be extracted from the fuel.

The focus of this paper is to study the performance of the N+3 turbofan engine model by altering the BPR while keeping the core flow and the FPR constant. We also explore alternative fuel sources to conventional Jet-A, that is LH_2 and NH_3 to study their advantages and disadvantages to the performance of the engine and the environment.

II. Methodology

Previous work by Dankanich and Peters⁵ showed that the *TSFC* decreases with an increase in *BPR*. Dankanich built a simple model in MATLAB to calculate the performance of a turbine engine. The flight conditions such as free stream and altitude were set at the beginning in the code. The necessary propulsive equations were then used to calculate the performance of the different components of the engine. Two different methods were used in the performance calculations: a fixed core method with constant core mass flow rate and fuel flow rate, and the thrust convergence method with constant thrust level. The motivation for this paper is to replicate the fixed core method study of Dankanich and Peters⁵ that has already been modeled and conducted in MATLAB⁵ using NPSS. The MATLAB code was written using basic propulsion equations, therefore replicating the study of Ref. 5 in NPSS would show that the NPSS solver environment does not deviate from the fundamental propulsion equations and theories. Good comparison between MATLAB and NPSS results would give confidence in utilizing NPSS for engine modeling; it essentially provides a validation of NPSS. In addition to showing the capabilities of NPSS, this study solidifies our understanding of gas turbine performance. This work is a sequel to a previous work⁶ with an improved solver set up to match with the MATLAB code analysis method. In the new solver setup, the core mass flow and the *FPR* are kept constant in order to make one to one comparison with the work done by Dankanich and Peters⁵.

In this paper, NPSS is employed to calculate the performance of NASA N+3 geared turbofan engine using different fuels such as LH_2 and NH_3 . The NPSS engine model contains all mechanical elements shown in Fig. 1 with their appropriate connection scheme. The results considered necessary for the purpose of analyzing the engine performance were first defined in viewer files for extraction. For analysis, NPSS utilizes performance maps of the different engine elements in conjunction with input files and user defined functions. In keeping with the motivation of this paper, the engine was modeled to have constant mass flow through the core and any change in *BPR* is obtained by altering the bypass bleed without changing the core. The solver setup was constructed using dependent and independent variables. The NPSS model utilizes the predefined dependent variables to calculate for the independent variables which are used to analyze the performance of the engine. Table 1 gives a list of the independent and dependent variables used in the solver setup.

Table 1 Dependent and independent variables used for the solver setup in NPSS.

NPSS Solver Variables	
Independent Variables	Dependent Variables
Fuel Flow, W_{fuel}	Desired Gross Thrust, F_{gross}
Fan Pressure Ratio, FPR	Desired Burner Temperature, T_{04}
Bypass Ratio, BPR	Desired Overall Pressure Ratio, OPR

To make appropriate comparisons between the MATLAB code and the NPSS code, Jet-A fuel was the choice of this study. In addition, initial estimations of important design parameters were made by scaling the bypass ratio *BPR* with preliminary calculations while keeping the core flow and *FPR* fixed. The initial estimates (see Appendix A) were used to obtain the desired *BPR* values that were used in the MATLAB code. The variables that change with *BPR* were included as an input file in NPSS (see Appendix B). Fine tuning of the gross thrust from the initial estimates was needed to reach at the exact *BPR* level when running the simulation. Similar flight conditions were used in both models. The NPSS simulation was performed with the T_{04} , *FPR*, and *OPR* set to 3150.0°F, 1.3, and 55, respectively. A total of thirteen simulations with *BPR* values ranging from 1-12 and 20 were made and the *TSFC* of the engine was recorded at each *BPR* value.

Once the engine model was validated against the MATLAB code for Jet-A fuel, the same solver setup was employed to study the engine performance using liquid hydrogen (LH_2) and Ammonia (NH_3) as alternative fuels. The results from these simulations were used to analyze what effect the different energy of various fuels has on the *TSFC* of the gas turbine engine. In addition, the emissions from using Jet-A fuel and LH_2 were compared. The results are discussed in the Results and Discussion part of the paper in section III.

III. Results and Discussion

Figure 2 shows the *TSFC* vs *BPR* plot for based on results obtained from both MATLAB and NPSS codes. As can be seen from this figure, the results from the two simulations agree within $\pm 8\%$. Both simulations show a similar decreasing trend for *TSFC* as *BPR* increases consistent with the well-known relation between the two quantities. However, the *TSFC* obtained from the NPSS is below that obtained from the MATLAB at lower *BPR* values; it becomes approximately the same around *BPR* = 6, and becomes higher thereafter. NPSS employs a more

advanced thermochemical calculation model together with detailed engine map data and turbomachinery configurations, which provide more accurate results compared to the simplified calculation model used in MATLAB.

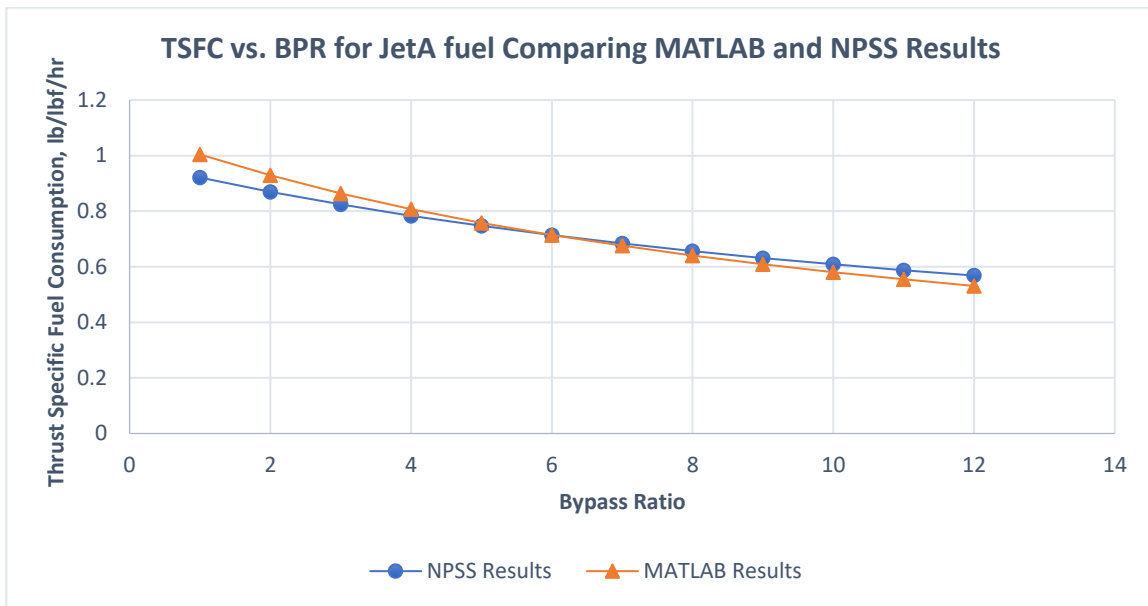


Fig. 2 *TSFC vs BPR* plot comparing the results obtained from MATLAB and NPSS using the fixed core method.

The above result in Fig. 2 demonstrates that the new NPSS solver setup can replicate the fixed core method study of Ref. 5 using MATLAB using Jet-A fuel with good accuracy. The validated NPSS is now used to study the performance of alternative fuels (LH_2 and NH_3) and the results are compared to those obtained for conventional Jet-A fuel. Figure 3 shows a plot of *TSFC vs BPR* for all three fuel types. As can be seen from this figure, LH_2 gives the lowest *TSFC* value while NH_3 gives the highest; Jet-A fuel results are somewhere in the middle. This trend is consistent with our expectation based on the heating values of the three fuels. LH_2 has the highest *LHV* (51,621 BTU/lbm) and hence can achieve the required thrust level by burning lower amount of fuel compared to Jet-A which has a *LHV* of 18550 BTU/lbm. NH_3 has the lowest *LHV* (7987 BTU/lbm) and would require much more fuel to get to the same thrust level compared to both LH_2 and Jet-A. Compared to the *TSFC* value using Jet-A as a fuel, using LH_2 decreases *TSFC* by 62.5% while using NH_3 increases *TSFC* by 130%.

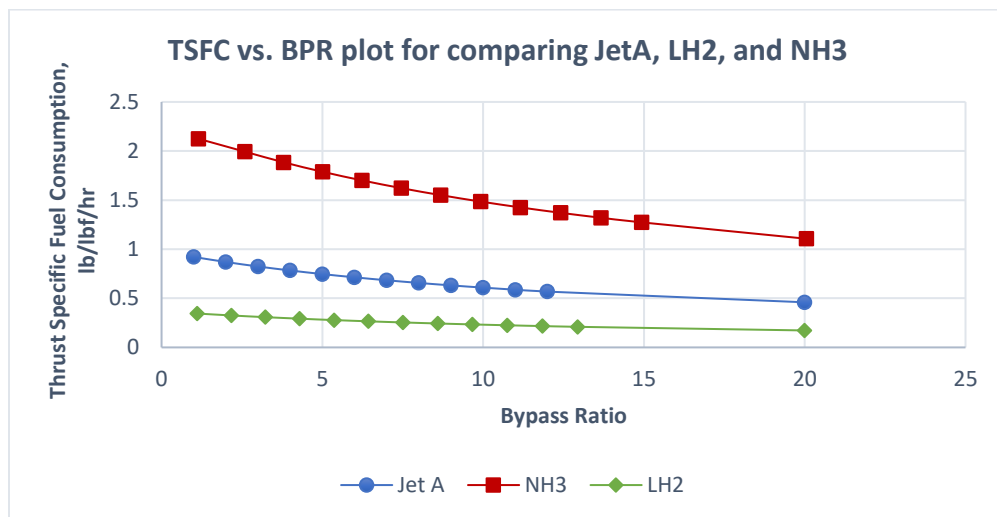


Fig. 3 *TSFC vs BPR* plot comparing the performance of three types of fuels in NPSS: conventional Jet-A, Ammonia (NH_3), and Liquid Hydrogen (LH_2).

IV. Conclusions

The main goal of this paper was to show the capability of NPSS in replicating the results of Ref. 5 based on fixed core analysis method modeled in MATLAB by building a new solver setup in NPSS. NPSS was proven to be capable of replicating the study of Ref. 5 with great accuracy. Moreover, it was demonstrated and verified that the calculations based by NPSS's built-in thermochemistry and turbomachinery data are well in line with the fundamental propulsive equations used in MATLAB code of Ref. 5. The additional capability of NPSS in performing detailed thermodynamics calculations using performance maps gives it an extra edge in sizing and studying an engine model. It is concluded that NPSS can be employed with confidence as a preferred engine performance study tool for research, development and design.

From the study of the performance of the Jet-A, LH_2 , and NH_3 fuels, it was shown that there is a direct relationship between $TSFC$ and LHV of a fuel. Higher the LHV value of a fuel, lower is its $TSFC$ value. This implies that less fuel is required (by mass) if for example conventional Jet-A fuel is replaced by LH_2 . The volume needed to store LH_2 , however would be much higher due to its low density. In addition, having LH_2 on board would require lot more energy since it needs to be kept in a cryogenic tank due to its extremely low boiling temperature. On the other hand, Jet-A can be stored with no extra energy expenditure. With the current technology level, these facts make the use of LH_2 for commercial airplanes unattractive. Nonetheless, the environmental benefits of using LH_2 , with zero emissions of CO_2 and low NO_x make it a great candidate as an alternative fuel source for zero-emission aviation. With future advancements in technology and research findings, improved LH_2 delivery and storage infrastructures can be developed to use it as a main source of fuel in aviation. Even though the low LHV of NH_3 makes it less desirable for usage as a fuel source by itself, its environmental benefit and abundance make it a worthwhile alternative fuel source to study.

V. Future Directions

Using the knowledge gained through this paper, NPSS can be utilized with confidence to obtain accurate performance predictions of new engine models and alternative fuel sources. We believe that the use of NH_3 as fuel source either mixed with LH_2 or as a carrier for hydrogen has considerable merit to investigate further their potential to achieve nearly zero emission aviation.

Acknowledgments

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C. MATLAB Code⁵

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% MEMS 500 Independent Study %%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% Andrew Dankanich %%%%%%%%%% Fall 2016 / Spring 2017 %%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% Modified for use by Abel Solomon & Richard Carter % Spring 2022 %%%%%%%%%%
clear;
close all;
clc;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% Flight Conditions and Free Stream Constants %%%%%%%%%%
M0 = 0.8; % Free Stream Mach Number 0.88 for this code and 0.8 for NPSS
gc = 32.2 ; %constant lbf to lbm
R = 287; %kJ/kg universal gas constant
g = 1.4; % Gamma for Air
alt = 35000; %Feet, This is not directly used, but coincides with T0 and P0
rec = 0.995; % Inlet Recovery
% 0.995 for NPSS and 0.96 for this code
T0 = 219; % K Free stream temperature at 35k
% this code has 233: 219 for npss
P0 = 24; % kPa Free stream pressure at 35k
% 24 for NPSS 15 for matlab
a0 = sqrt(g*R*T0); % m/s
Pt0 = P0 * (1+((g-1)/2)*M0^2)^(g/(g-1)); % lbf/ft^2
Tt0 = T0 * (1+((g-1)/2)*M0^2); % R
mft0 = sqrt(g)*M0*(1+((g-1)/2)*M0^2)^-((g+1)/(2*(g-1)));
u0 = M0*sqrt(g*R*T0); %Free Stream Velocity
den0 = P0/(R*T0); %Free Stream Density
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

bpr = [1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 20]; %Various Bypass Ratios
% FOR EVERY ITERATION CHANGE THIS TO MATCH UP TO THE BPR RANGE
% CORESPONDING TO THE FPR USED
sz = length(bpr);
n = 0;

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
for j = bpr
    BPR = j;

    mdotc = 13.23; % UPDATE FOR EVERY ITERATION USING NPSS DATA CHANGE FROM
    % lbm/s to kg/s
    % kg/s CORE AIRFLOW ONLY. This remains constant for all BPR
    % and through "guess and check" yields around 30,000lbf for the turbojet
    % configuration (BPR = 0) 162.5 kg/s for matlb 960.8 for NPSS
    mdotfan = BPR*mdotc+mdotc; % Calculate Fan mass flow
    mdot0 = mdotfan; %+ mdotc; % Total Engine Inlet Airflow

    %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% Station 2 and 3 Compressor Inlet and Exit %%%%%%%%%%
    tau_a = 8; % Thermal Limit Parameter, See definition in Burner Section
    %pic = 40;
    %Compressor Pressure Ratio: From Farohki, equation 4.74 page 161
    pic = ((sqrt(tau_a)/(1+((g-1)/2)*M0^2)))^(g/(g-1)); %
    etac = 0.9; % Compressibility Efficiency factor of the Compressor
    rec = .995; %Inlet Recovery
    Pt2 = Pt0*rec ;
    Tt2 = Tt0;
    Pt3 = Pt2*pic;
    Tt3 = Tt2*(1+((1/etac)*((pic^((g-1)/g))-1)));
    %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%% Station 13 and 19 Fan Properties %%%%%%%%%%

    pifan = 1.3; % CHANGE THIS VALUE EVERY ITERATION
    % Using a Typical Single Stage Fan value between 1.4-1.6
    Pt13 = Pt2*pifan; %
    Pt19 = Pt13*.95; %Account for a Small pressure loss across the Fan
    tau_r = Tt0/T0;
    tau_fan = pifan^((g-1)/g);
    Tt13 = Tt2*tau_fan; %
    V19_a0_fan = sqrt((2/(g-1))*((tau_r*tau_fan)-1));
    P19 = Pt19/((1+(g-1)/2)^(g/(g-1)));

```

```

M19 = (((Pt19/P19)^((g-1)/g))-1)/((g-1)/2);
T19 = Tt13/((Pt19/P19)^((g-1)/g));
a19 = sqrt(g*R*T19);
V19 = a19*M19;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
g_t = 1.33; %Ratio of specific heats for the Turbine
g_c = g; %Ratio of specific heats for the compressor is the same as air
cpt = (g_t/(g_t-1))*R; % Metric Unit value should be ~1156
cpc = (g/(g-1))*R; % Metric Unit value should be ~1004
eta_b = .999; %Burner efficiency
% 0.95 for matlab 0.999 for NPSS
pib = 0.96; % Pressure Ratio Across the burner
% 0.95 for matlab 0.96 for NPSS
hpr = 120070.45; % [kJ/kg] FOR LH2 120070.45 KJ/kg
% for JET A 18550 BTU/lb=43147 KJ/Kg
Pt4 = Pt3*pib; %
%Now we need to set the "Thermal Limit Parameter" IE Turbine Temp Limit
% tau_a = ht4 / h0 % This is the definition of the Thermal Limit Parameter
tau_a = 8; %This can be adjusted and is a driving factor in Engine
% Performance
% tau_a of 8 means Tt4 is ~1600 K if T0 is 233k
Tt4 = (cpc*T0*tau_a)/cpt; % This becomes a constant Temp Limit for all
% BPR's
f = (cpt*Tt4 - cpc*Tt3)/(hpr*10^3*eta_b - cpt*Tt4); %Need to convert hpr
% from kJ to J with 10^3. Realize that fuel to air ratio becomes constant
% as well.
mdot4 = mdotc*(1+f); % This is the core air flow and fuel flow
mdotfuel = f*mdotc;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
eta_m = .99; % Mechanical efficiency of the Turbine
eta_t = .936; % Flow efficiency of the turbine
%Energy Balance across the Turbine for Tt5.
Tt5 = Tt4 - ((cpc*(Tt3-Tt2) + BPR*cpc*(Tt13-Tt2))/((1+f)*cpt*eta_m));
Pt5 = Pt4*((Tt5/Tt4)^(g_t/(eta_t*(g_t-1))));
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Assuming an Ideal expansion through the Nozzle
Pt9 = Pt5; %Assume Ideal Nozzle
Tt9 = Tt5; %Station 9 we assume same as turbine exit
P9 = P0; % Assume ideally expanded
%%Assume the Core is Choked for Cruise Condition IE M = 1
M9 = sqrt((((Pt9/P9)^((g-1)/g))-1)*(2/(g-1)));
T9 = Tt9/(1+(g-1)/2*M9^2);
mdot9 = mdot4;
V9 = M9*sqrt(g*R*T9);
V9_a0_core = V9/a0;
% Thrust contribution from the Core ONLY
cfg = 1; % Nozzle coefficient
Fgcore = mdot9*gc*V9*cfg;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Overall Engine Thrust
%Specific Thrust
Fn_mdot = (a0/(1+BPR))*(V9_a0_core - M0+BPR*(V19_a0_fan - M0)); % N/m/s
%Net Thrust
Fn = (Fn_mdot * mdot0)*.224809; %lbf (converting from Newton to lbf)
Fn_Metric = (Fn_mdot * mdot0); %Newtons or kg(m/s^2)
% Thrust Specific Fuel Consumption
tsfc = mdotfuel / Fn_Metric; % kg/N/s
tsfc_english = ((mdotfuel*2.20462) / Fn)*3600 ; % lb/lbf/hr (converting kg
% to lbf and seconds to hour)
end

```