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

Abel Solomon
Washington University in St. Louis

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

Abel M. Solomon[1], Ramesh K. Agarwal[2]
Washington University in Saint Louis, Saint Louis, Missouri, 63130, USA

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

$TSFC$	=	*Thrust Specific Fuel Consumption*
BPR	=	*Bypass Pressure Ratio*
FPR	=	*Fan Pressure Ratio*
NPSS	=	*Numerical Propulsion Simulation System*
NIST	=	*National Institute of Standards and Technology*
LH_2	=	*Liquid Hydrogen*
NH_3	=	*Ammonia*
HHV	=	*Higher Heating Value*
LHV	=	*Lower Heating Value*
F_{net}	=	*Net Force*
T_{t4}	=	*Turbine Entrance Temperature*
OPR	=	*Overall Pressure Ratio*
$\dot{m}_c$	=	*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

[1] PhD candidate, Mechanical Engineering and Materials Science Department
[2] William Palm Professor of Engineering, Mechanical Engineering and Materials Science Department

modeled by Carter & Agarwal[2] using published NASA engine cycle data. The N+3 model architecture used by NASA resembles the one shown in Fig. 1.

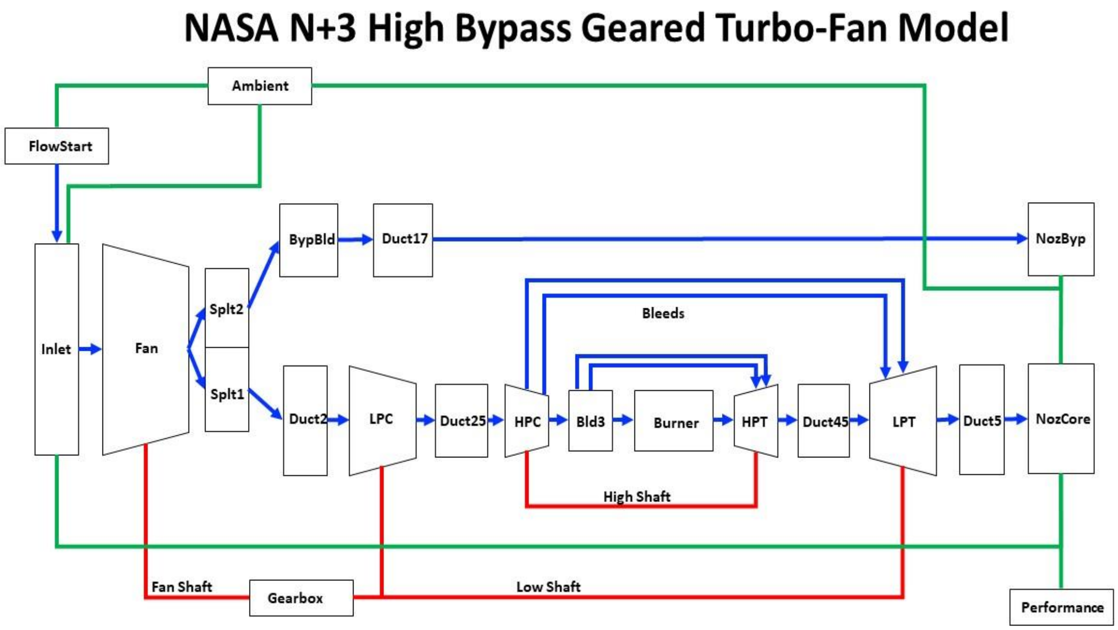

Fig. 1 Schematics of NASA N+3 Turbofan Engine Model[2].

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):

$$Thrust = Thrust\ of\ Fan + Thrust\ of\ Core \qquad (1)$$
$$T = (\dot{m}_f v_f - \dot{m}_f v_o) + (\dot{m}_e v_e - \dot{m}_c v_o)$$

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):

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

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).

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

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

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

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):

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

where $$\Delta V = V_{jet} - V_o \quad \text{and}$$
$$V_{jet} = M_9 * \alpha.$$

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):

$$\alpha = \sqrt{\gamma R T} \tag{6}$$

where γ is the adiabatic index (~1.4 for diatomic gas such as air), R is the universal gas constant(~53.4 ft·lbf/lb·°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:

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

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

$$\eta_{prop} = \frac{TV_o}{\Delta KE} = \frac{2V_o}{(V_o + V_{jet})} = \frac{2}{(1 + \frac{V_{jet}}{V_0})} \tag{8}$$

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 ($P_\infty = P_1 = P_9$).

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

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):

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

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):

$$TSFC = \frac{\dot{m}_{fuel}}{T_{net}} \tag{11}$$

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 form[4]. 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 Peters[5] 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 Peters[5] that has already been modeled and conducted in MATLAB[5] using NPSS. The MATALB 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 work[6] 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 Peters[5].

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_{t4}, *FPR*, and *OPR* set to 3150.0^{o}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.

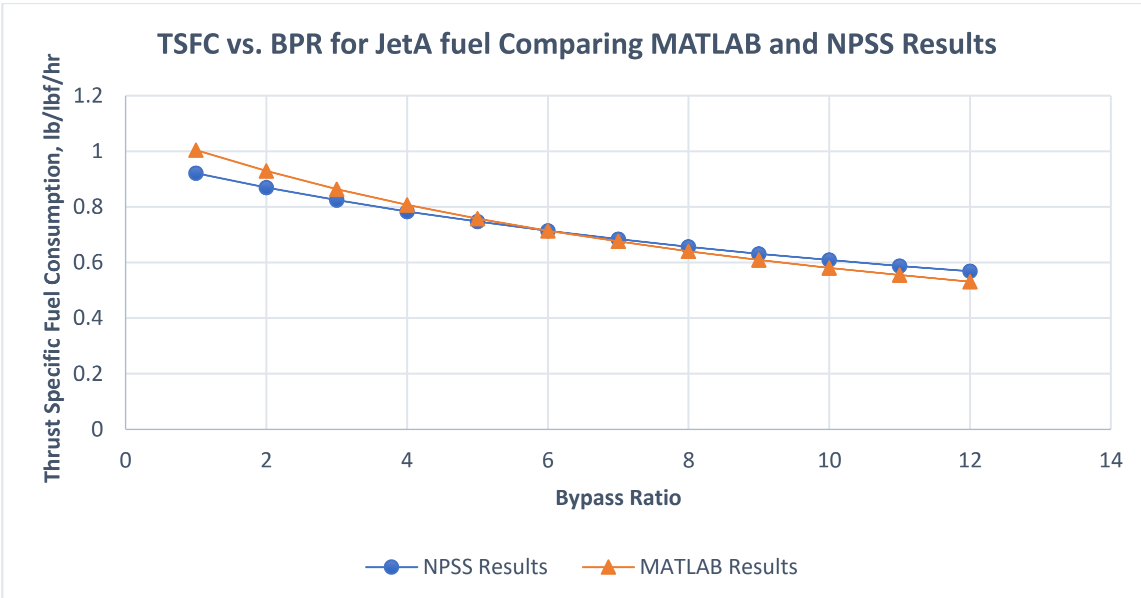

Fig. 2 *TSFC* vs *BPR* plot comparing the results obtained from MATLAB and NPSS using the fixed core method.

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%.

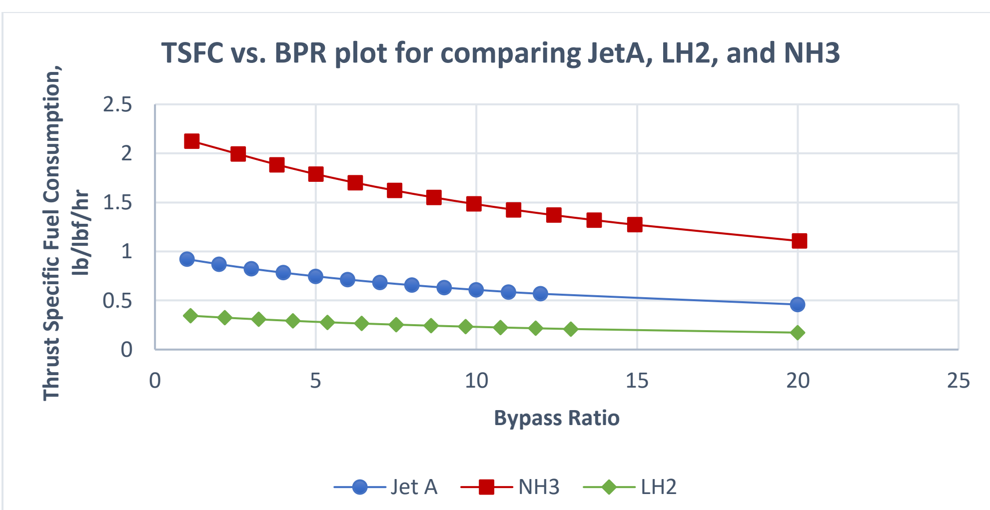

Fig. 3 *TSFC* vs *BPR* plot comparing the performance of three types of fuels in NPSS: conventional Jet-A, Ammonia (*NH_3*), and Liquid Hydrogen (*LH_2*).

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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Appendix

A. Initial Estimates

Variable Name	Description	Change or Constant	Scaler	Current Setting	Actual	Check													
InletStart.W_in	Mass flow into the engine	Change		820.921	820.921		612.633	379.249	350.076	320.903	291.73	262.557	233.384	204.211	175.038	145.865	116.692	87.519	58.346
InEng.Fl_O.Aphy	Physical Area	Change	1.00553617	7174.4	7174.4	7174.4	5356.377751	3319.20684	3064.56048	2809.91411	2555.26775	2300.62139	2045.97502	1791.32866	1536.68229	1282.03593	1027.38957	772.743203	518.09684
InEng.Afs	Free Stream Area	Change	0.89794952	6209.5	6406.78	6406.78	4783.275792	2964.07058	2736.66993	2509.26928	2281.86863	2054.46798	1827.06733	1599.66668	1372.26603	1144.86538	917.464726	690.064075	462.663424
ratio_Aphy_Afs																			
Fan.Fl_O.Aphy	Physical Area	Change	1.00385429	7162.4	7162.4	7162.4	5347.418601	3313.65509	3059.43465	2805.21421	2550.99377	2296.77334	2042.5529	1788.33246	1534.11202	1279.89158	1025.67114	771.450703	517.230264
ratio																			
SplitFan.BPRdes	Design BPR	Change		23.742	27.1394	27.1394	20	12	11	10	9	8	7	6	5	4	3	2	1
SplitFan.Fl_O2.Aphy	Bypass Physical Area	Change		6872.94	6872.9	6872.9	5064.887212	3038.93233	2785.68797	2532.44361	2279.19925	2025.95488	1772.71052	1519.46616	1266.2218	1012.97744	759.733082	506.488721	253.244361
SplitFan.Fl_O1.Aphy	is the same as Duct2 Aphy	Constant		292.4	262	262	262	262	262	262	262	262	262	262	262	262	262	262	262
Total Internal Area		Scaling Term		7165.34	7134.9	7134.9	5326.887212	3300.93233	3047.68797	2794.44361	2541.19925	2287.95488	2034.71052	1781.46616	1528.2218	1274.97744	1021.73308	768.488721	515.244361
Duct17.Fl_I.MNdes	Mach	Constant		0.45	0.45		0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45
Duct17.Fl_O.Aphy	Physical Area	Change	1.00513611	6977.61	6908.2	6908.2	5090.901052	3054.54063	2799.99558	2545.45053	2290.90547	2036.36042	1781.81537	1527.27032	1272.72526	1018.18021	763.635158	509.090105	254.545053
NozSec.CdTh_in	Nozzle Throat Discharge Coeff	Constant		1	1		1	1	1	1	1	1	1	1	1	1	1	1	1
NozSec.Cfg_in	Nozzle Exit Gross Thrust Coeff	Constant		0.9975	0.9975	0.99750388	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975	0.9975
NozSec.Ath	Nozzle Throat Area	Change	0.67507893	4746.8	4639.75		3419.198656	2051.51919	1880.55926	1709.59933	1538.6394	1367.67946	1196.71953	1025.7596	854.799664	683.839731	512.879798	341.919866	170.959933
NozSec.Vactual	Nozzle Exit Velocity	Constant		975.7	980.8		980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8	980.8
NozSec.Fg	Gross Thrust	Change		24436.7	25256.1		18222.52032	10933.5122	10022.3862	9111.26016	8200.13415	7289.00813	6377.88211	5466.7561	4555.63008	3644.50406	2733.37805	1822.25203	911.126016
NozSec.FgIdeal	Ideal Gross Thrust	Change		[illegible]	25319.3	25319.3985	18268.11973	10960.8718	10047.4659	9134.05987	8220.65388	7307.24789	6393.84191	5480.43592	4567.02993	3653.62395	2740.21796	1826.81197	913.405987
NozSec.Fl_Th.MN	Throat Mach	Constant		1	1		1	1	1	1	1	1	1	1	1	1	1	1	1
NozSec.Fl_O.MN	MN at exit	Constant		1	1		1	1	1	1	1	1	1	1	1	1	1	1	1
NozSec.PR	Pressure Ratio	Constant		[illegible]	2.033		1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914	1.914
Perf.myFg	Total Gross Thrust	Change			25993.6		18960.02032	11671.0122	10759.8862	9848.76016	8937.63415	8026.50813	7115.38211	6204.2561	5293.13008	4382.00406	3470.87805	2559.75203	1648.62602
OPR	55																		
SplitFan.Fl_O1.W	29.173																		
T4	3150																		
Bypass Fnet				772508.5236	776546.438														
Core Fnet				40147.8826	23160.4447														
NozPri.Vactual				1376.2	793.9														
NozPri.Fg				1476.3	737.5														
					25993.6														
Gross Thrust				22256.97124	25464.9134		18960.02032	11671.0122	10759.8862	9848.76016	8937.63415	8026.50813	7115.38211	6204.2561	5293.13008	4382.00406	3470.87805	2559.75203	1648.62602

B. Example Input File

The input file consists of of the changing variables from the initial estimate table. Shown here is .inp file for BPR 10. Notice the commented out (//F_{gross}) value is the one obtained from the initial estimate (see table above). The actual gross thrust value that was able to give us the desired BPR level of 10 was 12030.42 lbf and it was achieved through fine tuning.

```
InletStart.W_in        =    320.903;

InEng.Fl_O.Aphy        =    2809.914113;
InEng.Afs              =    2509.269283;

Fan.Fl_O.Aphy          =    2805.214213;

SplitFan.BPRdes        =    10;
SplitFan.Fl_O2.Aphy    =    2532.443606;

Duct17.Fl_O.Aphy       =    2545.450526;

NozSec.Ath             =    1709.599328;

NozSec.Fg              =    9111.260162;
NozSec.FgIdeal         =    9134.059867;

//Fgross               =    9848.760162;

Fgross                 =    12030.42;
```

C. MATLAB Code[5]

```
%%%%%%%%%%%% 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 ITTERATION 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 ITTERATION 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 ITTERATION
% 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;
%%%%%%%% Station 4 Burner Exit/Turbine Inlet %%%%%%%%%%%%%%%%%%%%%%%%%%%%
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;
%%%%%%%% Station 5 Turbine Exit %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
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))));
%%%%%%%% Station 9 Core Exit %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% 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 lbm and seconds to hour)
end
```