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MAE 112

24 November 2021

Design of a Turbofan Engine

Abstract

A turbojet engine is a type of engine that contains a compressor to increase the pressure of air that comes through the inlet. After being compressed, the air travels through a burner, where fuel is burned with the air to greatly increase the (stagnation) enthalpy of the air. The air travels out of the burner and through a turbine, where energy can be extracted to power the compressor. Finally, air leaves through the nozzle, faster than it came in through the inlet, and this difference in momentum is what creates thrust for the plane. This is similar in concept to a ramjet, except a ramjet relies on the fast forward motion of the plane to compress the air, while a turbojet has a compressor and thus can provide adequate compression even at lower speeds.

Turbofan engines further build on the basic functioning of a turbojet engine, introducing a fan that provides slight compression for the incoming air, as well as a bypass stream of air that does not travel into the main engine such that its only source of compression comes from the fan and not the compressor within the engine. Because there is a greater mass flow rate of air, in order to provide the same amount of thrust, lower exit velocities are required. As velocity is proportional to the square of kinetic energy, this means that there is less kinetic energy lost in the exhaust, directly improving the propulsion efficiency of the engine, which is defined as the ratio of thrust power to the change in kinetic energy between the exhaust and inlet air streams.

Introduction

In this project, we aimed to design a turbofan engine that met specific design requirements. Notably, these requirements included a minimum range for the aircraft, minimum thrust generated by the engine, a cruise Mach number, maximum inlet diameter, and a maximum temperature at the exit of the burner/inlet of the turbine.

The design process involved many different tradeoffs between various performance parameters. For example, increasing the burner stagnation temperature or decreasing the compressor pressure ratio creates greater specific thrust which is a good thing, but also increases the thrust specific fuel consumption, which is a negative thing. Similarly, increasing bypass ratio generally increases the propulsion efficiency and overall efficiency of the engine, but because the plane is traveling at supersonic speeds, the increase in engine size due to increased bypass ratio creates significant drag that reduces thrust and possibly even efficiency, so a lower bypass ratio may be necessary. The "ideal" engine is thus one that meets all minimum requirements while optimizing the most important performance parameters.

Design Method

The first step was creating programs to determine the following performance parameters: thrust specific fuel consumption, specific thrust, fuel-air ratio, as well as the exhaust velocity of the cold and hot streams, as a function of the max stagnation temperature in the burner T_{04}, the compressor pressure ratio π_c, the bypass ratio β, and the fan pressure ratio π_f. The program containing the function for calculating TSFC is located in Appendix (1) at the end. The code for the functions for specific thrust, fuel-air ratio, and the exhaust exit velocities are essentially the same as the function for TSFC, with the only difference being which variable within the code becomes the output.

To begin, the conditions after the diffuser (state 2) were determined. As the flow through the nozzle was adiabatic with no work done, the stagnation temperature T_{02} remained constant, which could be determined from the flight Mach number and ambient temperature with the following equation:

$$\frac{T}{T_0} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-1} \tag{1}$$

The isentropic stagnation temperature T_{02s} could then be determined using the definition of adiabatic efficiency for the diffuser:

$$\eta_c = \frac{T_{02s} - Ta}{T_{02} - Ta} \tag{2}$$

With the isentropic stagnation temperature, the stagnation pressure P_{02} could also be determined, using the following relation for an isentropic process:

$$\frac{P_{02}}{P_a} = \left(\frac{T_{02s}}{T_a}\right)^{\frac{\gamma}{\gamma - 1}} \tag{3}$$

After the diffuser comes the fan, and the conditions after the fan (state 3) were next calculated. The stagnation pressure could be calculated simply using the following definition of the fan pressure ratio π_f:

$$\pi_f = \frac{P_{08}}{P_{02}} \tag{4}$$

Using equation (3), the isentropic stagnation temperature T_{08s} could be determined, as T_{02} as well as $\frac{P_{08}}{P_{02}}$ were both already known. From here, the actual stagnation temperature T_{08} could be determined using equation (2) and the adiabatic efficiency for the fan. In addition, the following equation could be used to calculate the velocity of the flow as it left the fan nozzle, as this was the velocity of the cold air/bypass stream (T_8 can be calculated with equation (3) as the the value of the other three variables are known):

$$\frac{1}{2}u_{ef} = \eta_{fn}(c_p T_{08} - c_p T_8) \tag{5}$$

Next, the conditions after the compressor (state 3) were determined. The stagnation pressure could be found in a similar way to equation (4), except using the compressor pressure ratio instead of the fan pressure ratio. The stagnation temperature was found using the equation for a polytropic process:

$$\frac{T_{03}}{T_{08}} = \pi_c^{\frac{\gamma-1}{\gamma^*\eta_c}} \text{ where } \pi_c = \frac{P_{03}}{P_{08}} \tag{6}$$

Afterwards, the burner conditions (state 4) were found. The temperature T_{04} was already known, as it was one of the input variables to the function. The pressure P_{04} could be easily determined using P_{03} and the burner pressure ratio, similarly to how P_{03} and P_{08} were calculated.

In addition, the required fuel-air ratio could be calculated in this step. Using conservation of energy, the enthalpy at state 4 was necessarily equal to the enthalpy at state 3 plus the energy added through burning fuel in the burner, multiplied by the burner efficiency. The equation for this conservation of energy could be written as follows:

$$(\dot{m}_a + \dot{m}_f)c_p T_{04} = \eta_b \dot{m}_f Q_R + \dot{m}_a c_p T_{03} \tag{7}$$

$\dot{m}_a$ could be divided out of the equation ($\dot{m}_f/\dot{m}_a = f$), leaving f, the fuel-air ratio, as the only unknown, which could be solved for.

Next was after the turbine (state 5). Similar to the burner calculation, conservation of energy was used in this step. The energy extracted from the turbine was used to power the fan and the compressor, which means that the enthalpy change across the turbine equals the enthalpy change across the compressor + the enthalpy change across the fan (mass flow rate across each of these components is different- hot air stream + fuel goes through the turbine, only the hot air stream goes through the compressor, and both the hot and cold air streams go through the fan), leading to the following equation:

$$(1 + f)c_p(T_{04} - T_{05}) = c_p(T_{03} - T_{08}) + (1 + \beta)c_p(T_{08} - T_{02}) \tag{8}$$

The pressure P_{05} after the turbine could be calculated using equation (6), as both the temperature before and after the turbine as well as the pressure before the turbine were known.

Finally, the nozzle exit conditions were determined. Using equation (5), the hot stream exit velocity was calculated, assuming perfect expansion such that static pressure at the exit was equal to the ambient pressure.

From the above, the fuel-air ratio f as well as the hot stream velocity u_e and the cold stream velocity u_e had already been calculated, leaving the thrust specific fuel consumption TSFC and specific thrust ST as the only things left to calculate.

The bare specific thrust $\frac{T_{bare}}{\dot{m}_a}$ could first be calculated by determining the rate of change of momentum of the air stream. The air came in with specific momentum $(1 + \beta)u$ and left the engine with $(1 + f)u_e + \beta u_{ef}$, meaning that: $\frac{T_{bare}}{\dot{m}_a} = (1 + f)u_e + \beta u_{ef} - (1 + \beta)u$. From this, the hot stream specific thrust $\frac{T}{\dot{m}_a}$ could be found by dividing the bare specific thrust by $(1.04 + 0.01\beta^{1.2})$, an approximation of the drag caused by increased engine size due to the bypass flow.

The overall specific thrust ST was calculated by dividing the hot stream specific thrust by $(1 + \beta)$, resulting in the thrust per total mass flow rate of air. The thrust specific fuel consumption could be calculated by dividing the fuel-air ratio by the hot stream specific thrust, resulting in the mass flow rate of fuel per unit thrust.

Next, four nested loops were created, cycling through max stagnation temperatures from 1400 to 1800 in intervals of 50 (9 values total), compressor pressure ratio from 16 to 40 in intervals of 2 (13 values total), bypass ratio from 0 to 10 in intervals of 0.5 (21 values total), and fan pressure ratio from 1 to 2 in intervals of 0.2 (6 values total). The minimum TSFC was found at a bypass ratio of 1.5 and a fan pressure ratio of 2, so values of $\beta = 1.5$ and $\pi_f = 2$ were used. Using these values of β and π_f, a carpet plot in the (TSFC,ST) plane could be created, with lines of constant T_{04} and π_c.

The minimum specific thrust was calculated and plotted, using the definition $ST = \frac{T}{\dot{m}_a(1+\beta)}$. The minimum thrust was given, and $\dot{m}_a$ could be calculated using $\dot{m}_a = \rho u A$, where $\rho = \frac{P}{RT}$, u is the flight speed, and A is the cross sectional area of the inlet, determined by choosing a diameter within the design specifications.

The maximum TSFC was also calculated and plotted using the range equation $s = \frac{L}{D}\frac{T}{\dot{m}_f}\frac{u}{g}ln(\frac{m_1}{m_2})$, with estimations of $\frac{L}{D}$ and $\frac{m_1}{m_2}$ based on similar aircraft.

Based off the minimum ST and maximum TSFC values plotted on the carpet plot, appropriate values of T_{04} and π_c were chosen.

Finally, contour plots of performance characteristics were created in the T_{04} and π_c plane. These included TSFC and ST, which could be calculated using the program created as described above, as well as the thermal, propulsion, and overall efficiencies, which could be calculated using the exit velocities that are outputs of the same program.

The thermal efficiency is defined as the ratio of the change in kinetic energy of the air divided by the energy provided by the burner:

$$\eta_t = \frac{\frac{1}{2}(1+f)u_e^2 + \frac{1}{2}\beta u_{ef}^2 - \frac{1}{2}(1+\beta)u^2}{fQ_R}$$

The propulsion efficiency is defined as the ratio of the thrust power divided by the change in kinetic energy of the air:

$$\eta_p = \frac{ST*u}{\frac{1}{2}(1+f)u_e^2 + \frac{1}{2}\beta u_{ef}^2 - \frac{1}{2}(1+\beta)u^2}$$

Lastly, the overall efficiency η_o is simply defined as the product of the thermal efficiency and propulsion efficiency:

$$\eta_o = \eta_t \eta_p$$

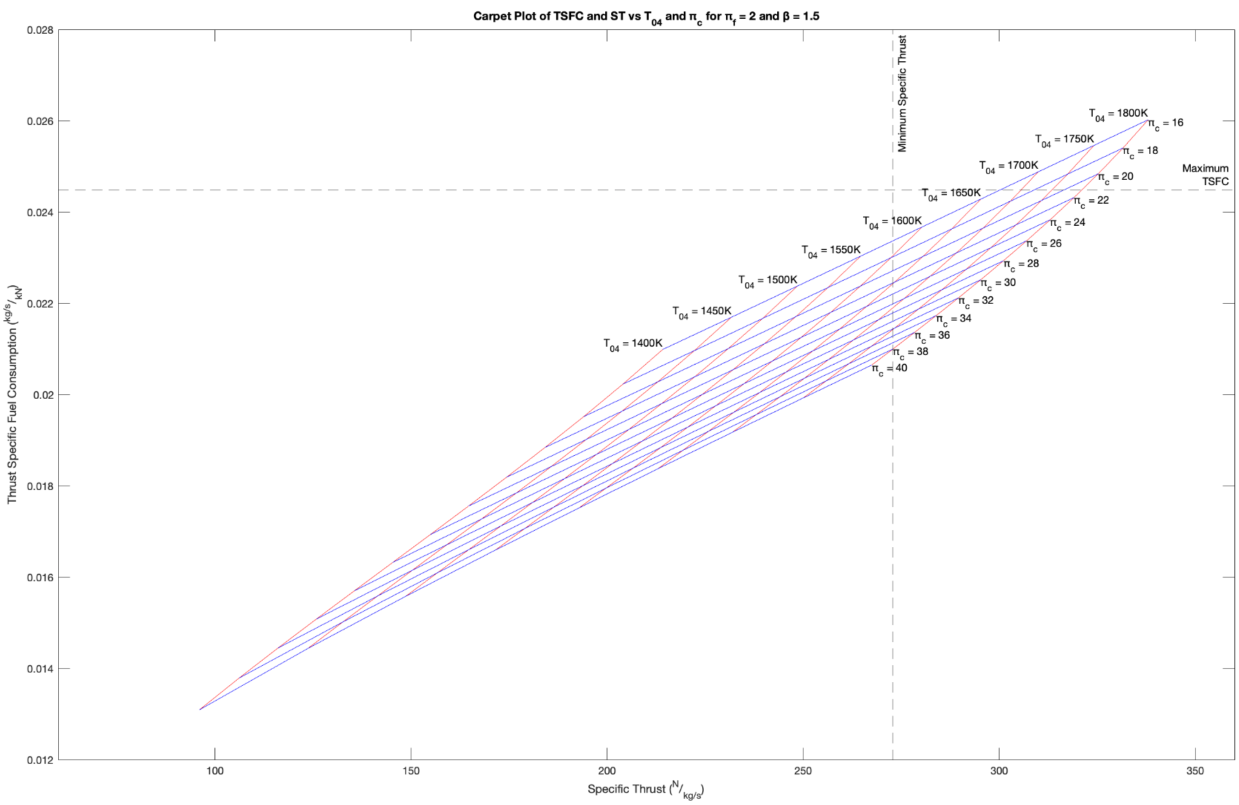
Results/Analysis

The following table is a list of normalized reciprocal TSFC values created from the four nested loops as described on page 6 above: https://docs.google.com/spreadsheets/d/1LjGau3wb4p4r8xD0aPsQUSD1WbFSBy3wk9Hf-GTvrOA/edit?usp=sharing.

The reciprocal of the TSFC values were taken (usually lower TSFC means better performance, so higher reciprocal value means better performance), and then each value was divided by the maximum value in the data set. The maximum value occurred at (1,13,4,6), which corresponds to a max stagnation temperature of 1400 K, compressor pressure ratio of 40, bypass ratio of 1.5, and fan pressure ratio of 2. From this, a bypass ratio value of 1.5 and a fan pressure ratio of 2 were chosen.

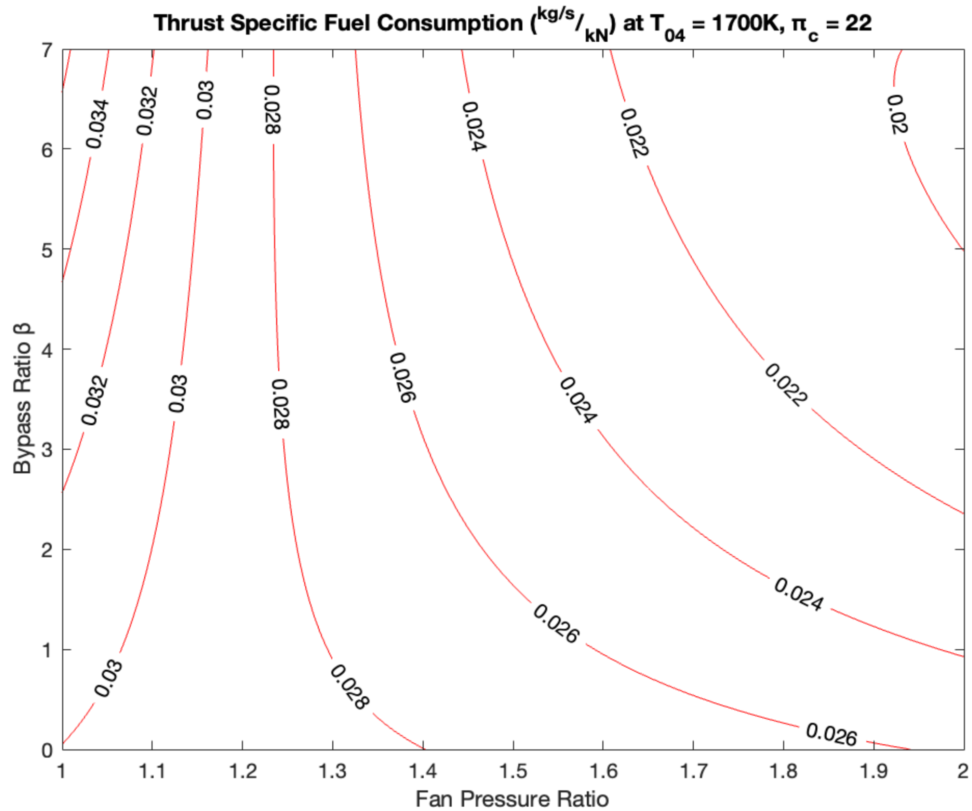
Next, the carpet plot was created, using the program described above to calculate TSFC and ST for each set of inputs. The max stagnation temperature T_{04} again ranged from 1400 to 1800 in intervals of 50 and the compressor pressure ratio π_c ranged from 16 to 40 in intervals of 2. Using the process described above, the minimum specific thrust was calculated to be 272 $\frac{N}{kg/s}$. The maximum TSFC was calculated also using the process described above. The $\frac{L}{D}$ ratio was estimated to be 7.5, which is approximately the value for a Concorde traveling at Mach 2, while the fuel was estimated to take up 40% of the total take off weight, which is the approximate fuel fraction for long-haul flights, resulting in $\frac{M_1}{M_2} = 1.67$. An inlet diameter of 1.6

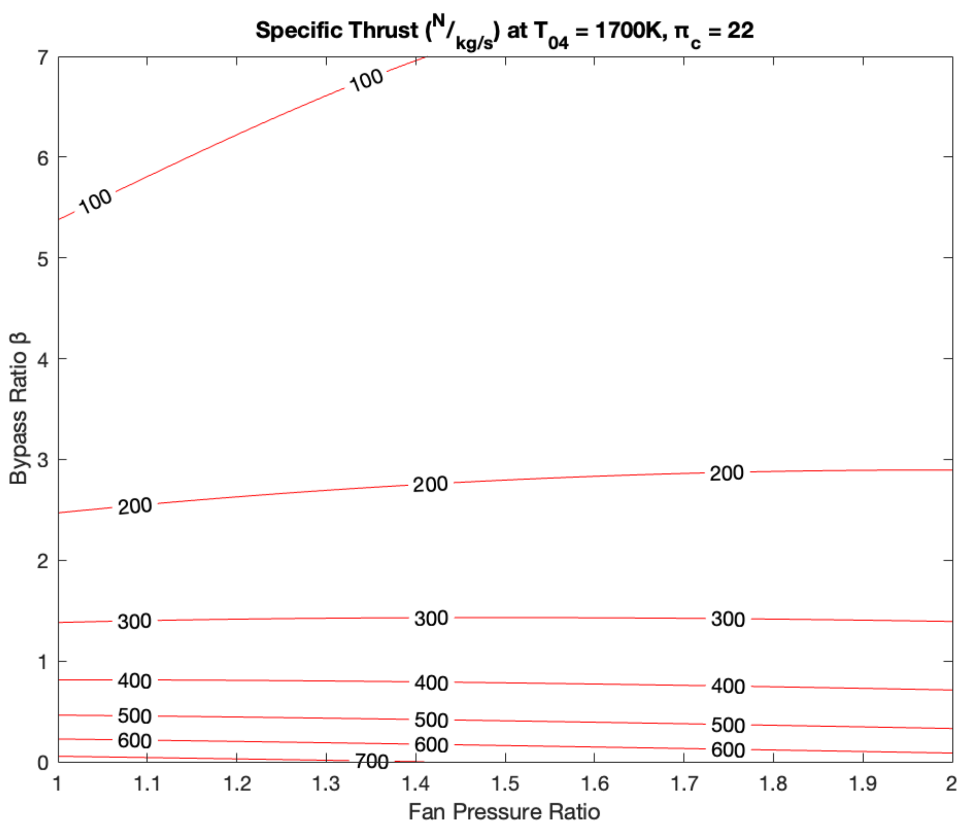
meters was also chosen, which is within the 2 meter limitation, which means a cross sectional area of 2 square meters. This leads to a maximum TSFC of 0.0245 $\frac{kg/s}{kN}$. The maximum TSFC and minimum ST lines were plotted with the carpet plot, resulting in the following graph:



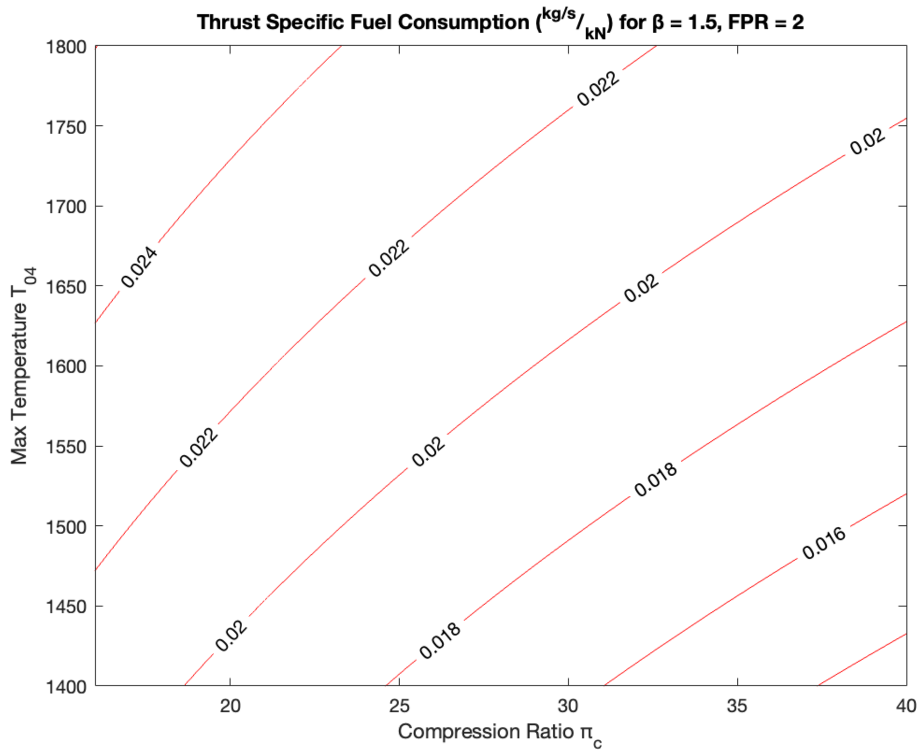
From the graph, a maximum stagnation temperature of 1700 K and a compressor pressure ratio of 22 were chosen, resulting in a TSFC below the maximum and an ST below the minimum.

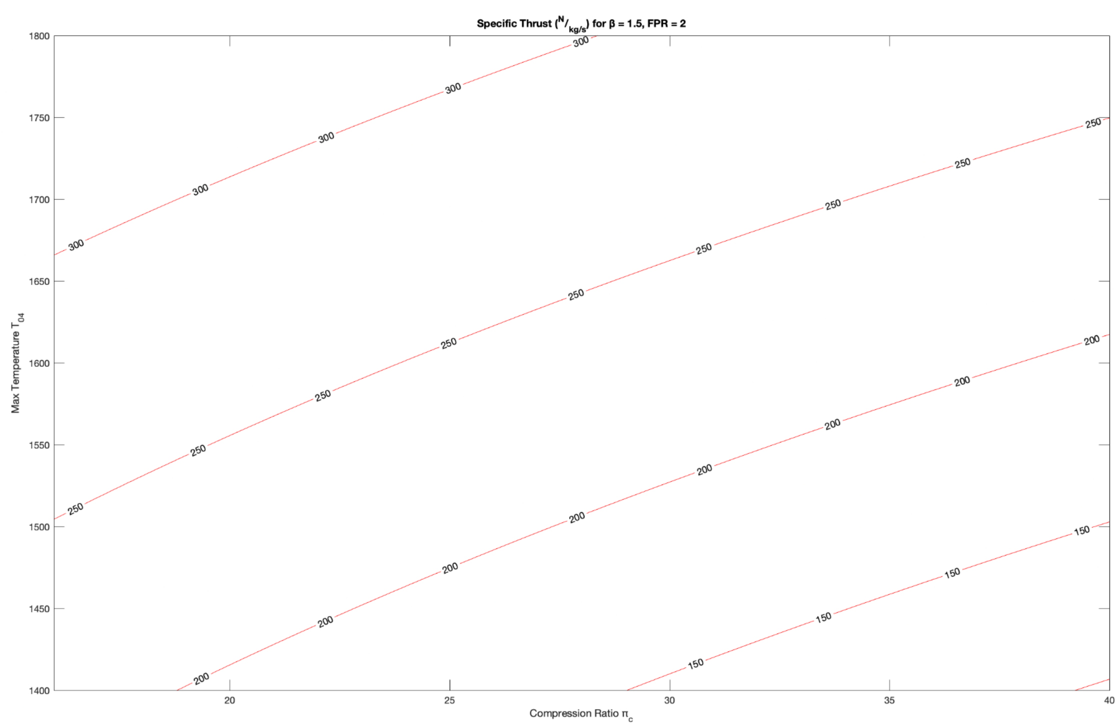
Using these stagnation temperature and pressure ratio values, contour plots for TSFC and ST in the (β, π_f) plane were created as follows, confirming the choice of β and π_f of minimizing TSFC while staying within the ST requirements:


Thrust Specific Fuel Consumption (kg/s/kN) at T04 = 1700K, πc = 22
Bypass Ratio β
Fan Pressure Ratio
0.034
0.032
0.03
0.028
0.026
0.024
0.022
0.02
1
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2
0
1
2
3
4
5
6
7


Specific Thrust (N/kg/s) at T04 = 1700K, πc = 22
Bypass Ratio β
Fan Pressure Ratio
100
200
300
400
500
600
700
1
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2
0
1
2
3
4
5
6
7

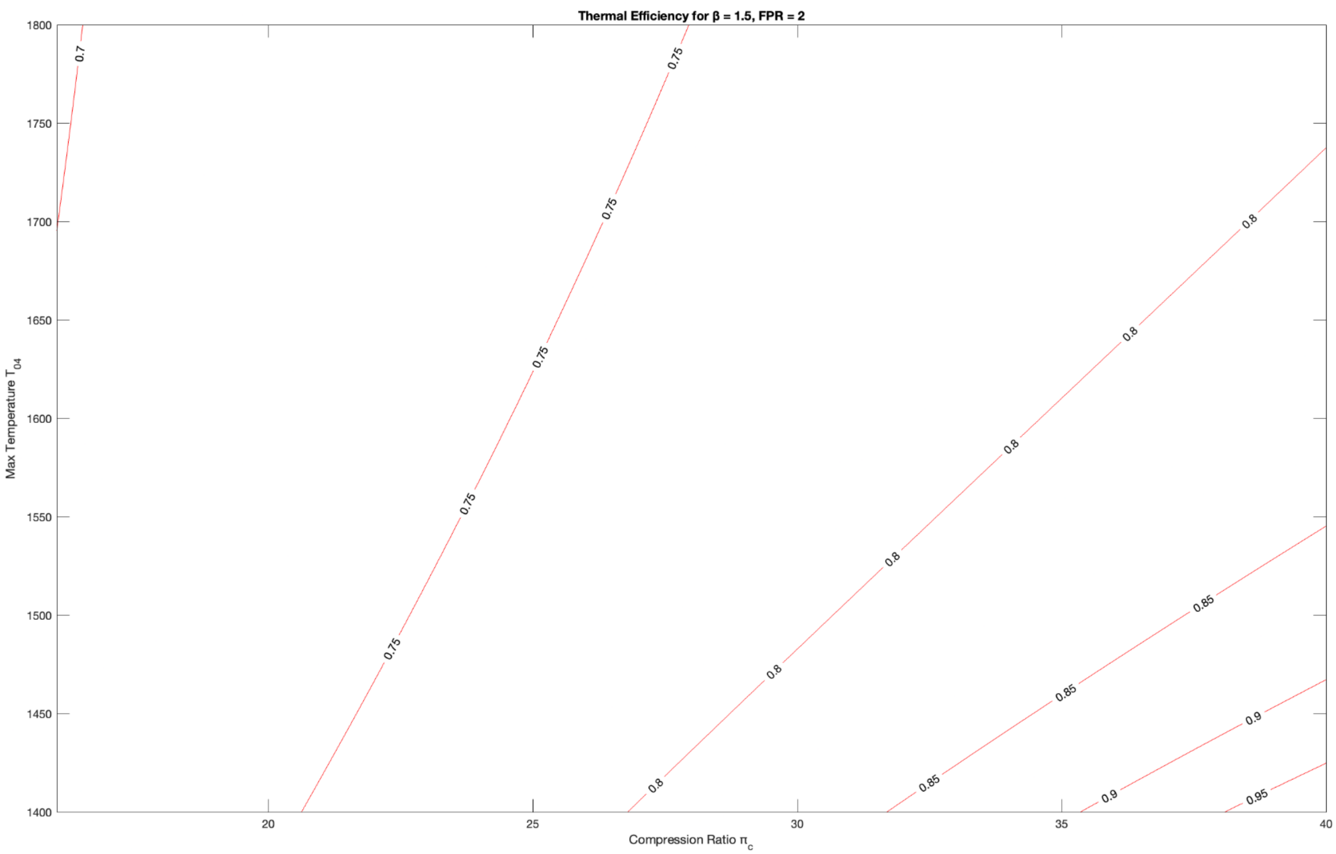
Afterwards, contour plots for the performance parameters TSFC, ST, η_t, η_p, and η_o in the (π_c, T_{04}) plane were also created.

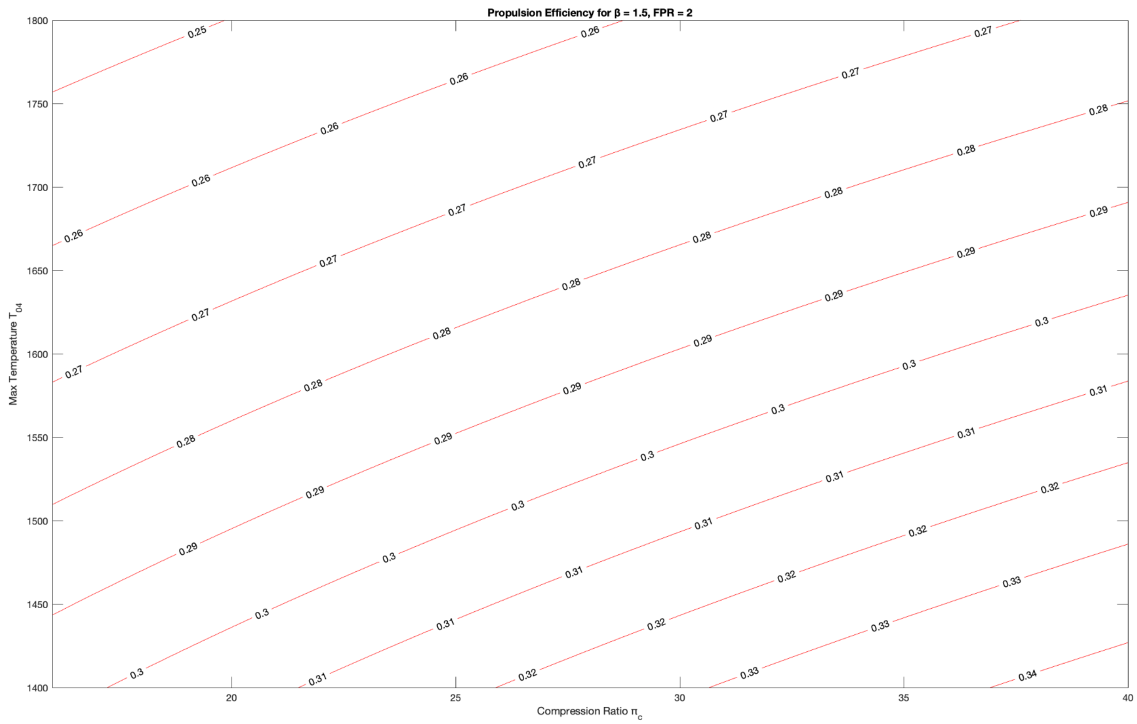




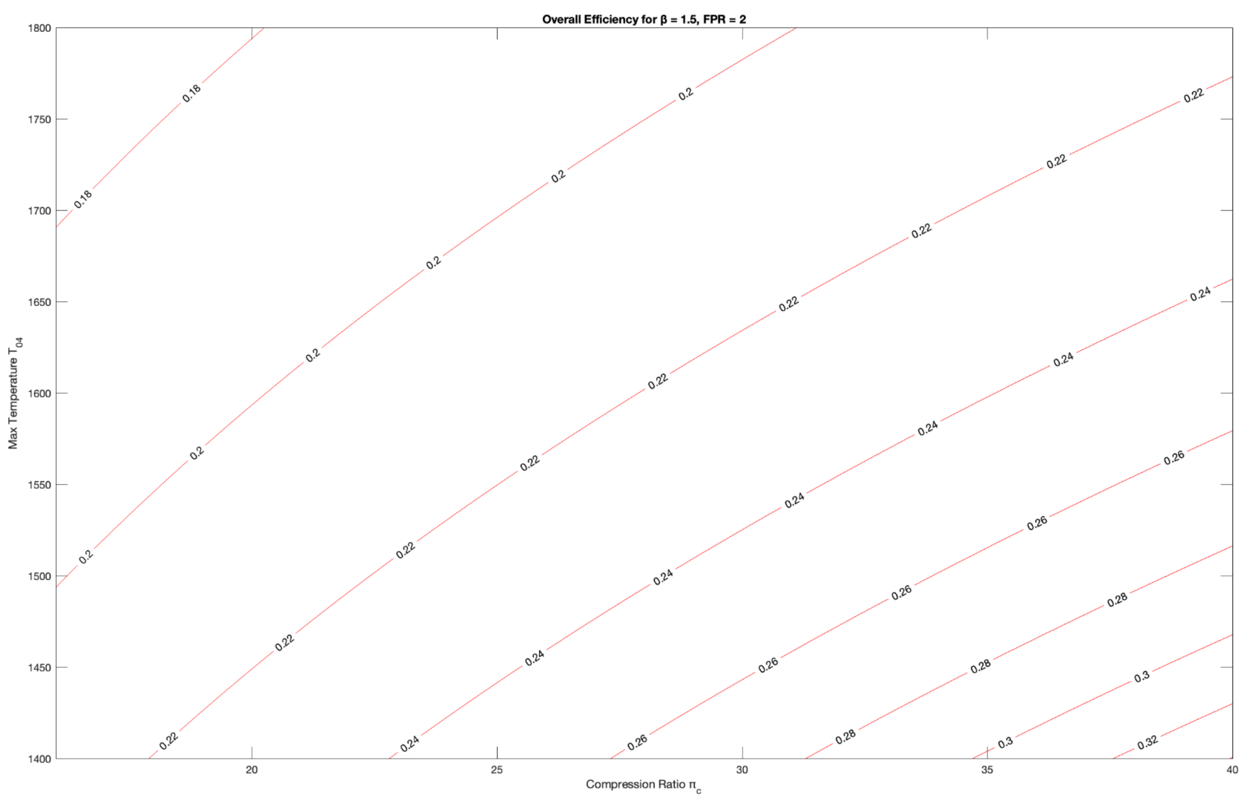
From the graphs above, it can be seen that increasing the compression ratio causes decrease in both TSFC and ST, while increasing the max stagnation temperature increases both ST and TSFC. Therefore, changing either value results in a tradeoff, as maximum ST while minimum TSFC (which is what is desired) is not possible.

The contour plots of the efficiencies are as follows:





Propulsion Efficiency for β = 1.5, FPR = 2
Max Temperature T_{04}
Compression Ratio π_c
1800
1750
1700
1650
1600
1550
1500
1450
1400
20
25
30
35
40
0.25
0.26
0.27
0.28
0.29
0.3
0.31
0.32
0.33
0.34



Overall Efficiency for β = 1.5, FPR = 2
Max Temperature T_{04}
Compression Ratio π_c
1800
1750
1700
1650
1600
1550
1500
1450
1400
20
25
30
35
40
0.18
0.2
0.22
0.24
0.26
0.28
0.3
0.32

From these plots, it can be seen that overall efficiency increases as temperature decreases, and it also increases as the pressure ratio increases. Thus, it appears that minimizing the maximum temperature while maximizing the pressure ratio is ideal for the engine. However, the problem with this is that both decreasing the max temperature and increasing pressure ratio cause a decrease in specific thrust. The specific thrust of this engine is only about 20% higher than the minimum value, so changing any parameters to increase the efficiency may lead to a specific thrust that is below an acceptable value.

The design aimed to maximize the fuel efficiency by minimizing the thrust specific fuel consumption- that is, needing as little fuel as possible to achieve the desired thrust. This comes as a tradeoff with specific thrust, so for the same amount of air flow, there is less thrust, but also less fuel needed relative to the amount of air for the same amount of thrust. For a given specific thrust, this would mean lower actual thrust which means a longer travel time, but also more fuel efficient travel. Thus, costs associated with taking more time for trips are (in my opinion) offset by reduced emissions caused by less fuel consumption, which is better overall for the environment. However, if faster travel times are necessary, the maximum temperature of the burner can be increased by about 100 K to increase the specific thrust while remaining within the temperature bounds.

Summary

The design began with selecting the bypass ratio and fan pressure ratios that provided the minimum thrust specific fuel consumption for the maximum temperature and compressor pressure ratio values. From these bypass ratio and fan pressure ratio values, a carpet plot of TSFC and ST values were created, and the maximum TSFC and minimum ST were determined and graphed alongside the carpet plot. This was used to select the maximum burner temperature and compressor pressure ratio that provided as low of a thrust specific fuel consumption while still being a comfortable margin above than the minimum specific thrust. The design objective was comfortably able to meet the stated requirements, including exceeding the minimum cruise and take-off thrust, exceeding the required range, being within the diameter size maximum, and having a maximum turbine temperature comfortably below the limit, while flying at a cruise Mach number of 1.7 at an altitude of 60,000 feet.

For the final design, values of $T_{04} = 1700K$, $\pi_c = 22$, $\beta = 1.5$, and $\pi_f = 2$ were chosen, as well as an inlet diameter of 1.6 meters. With these design parameters, the thrust specific fuel consumption is 0.0231 $\frac{kg/s}{kN}$ and the specific thrust is 289.04 $\frac{N}{kg/s}$. In addition, the overall efficiency is approximately 20% and the fuel-air ratio is 0.167. With the given inlet diameter, the mass flow rate through the core engine is approximately 117 kg/s, resulting in a cruise thrust of 85 kN. The range is approximately 12,000 km under the given flight conditions, still assuming an L/D ratio of 7.5 and a fuel fraction of 40%.

Appendix 1: Performance Parameters Calculation (TSFC, ST, fuel-air ratio, exhaust velocities)

NOTE: program below is only for TSFC, but all four programs have the same code with different output variable

```
function thrustspecificfuelconsumption = TSFC(To_max,pic,B,prf)

%all givens
nd = 0.95;
yd = 1.4;
nc = 0.9;
yc = 1.37;
nb = 0.97;
yb = 1.35;
pib = 0.95;
nt = 0.92;
yt = 1.33;
nn = 0.98;
yn = 1.36;

nf = 0.92;
yf = 1.4;
nfn = 0.99;
yfn = 1.4;


QR = 45*10^6;
R = 287;
Ta = 216.65;
mach = 1.7;
Pa = 7231.355;

%conditions of ambient (state a)
Toa = Ta*(1+(yd-1)/2*mach^2);
Poa = Pa*(1+(yd-1)/2*mach^2)^(yd/(yd-1));
u = mach*sqrt(R*yd*Ta);

%conditions after diffuser, before fan (state 2)
To2 = Toa;
To2s = (To2-Ta)*nd+Ta;
Po2 = Pa*(To2s/Ta)^(yd/(yd-1));

%conditions after fan, before fan nozzle (state 8)
Po8 = Po2*prf;
```

```
To8s = To2*prf^((yf-1)/yf);
To8 = (To8s-To2)/nf+To2;
cpf = yf/(yf-1)*R;

%conditions after fan nozzle, before compressor
cpfn = yfn/(yfn-1)*R;
uef = sqrt(2*nfn*cpfn*To8*(1-(Pa/Po8)^((yfn-1)/yfn)));

%conditions after compressor, before burner (state 3)
Po3 = pic*Po8;
To3 = To8*(pic^((yc-1)/(yc*nc)));
cpc = yc/(yc-1)*R;

%conditions after burner, before turbine (state 4)
To4 = To_max;
cpb = yb/(yb-1)*R;
Po4 = pib*Po3;
f = (To4-To3)/(nb*QR/cpb-To4);

%conditions after turbine, before nozzle (state 5)
cpt = yt/(yt-1)*R;
To5 = To4 - (cpc*(To3-To8)+(1+B)*cpf*(To8-To2))/((1+f)*cpt);
   %(1+f)*cpt*(To4-To5) = cpc*(To3-To8) + (1+B)*cpf*(To8-To2) solved for To5
Po5 = Po4*(To5/To4)^(yt*nt/(yt-1));

%conditions after nozzle (state 6)
P6 = Pa;
Po6 = Po5;
To6s = To5;
T6s = To6s*(P6/Po6)^((yn-1)/yn);
cpn = yn/(yn-1)*R;
ue = sqrt(2*nn*cpn*(To6s-T6s));


st_bare = (1+f)*ue+B*uef-(1+B)*u;
sthrust = st_bare/(1.04+0.01*B^1.2);

thrustspecificfuelconsumption = f/sthrust;
%specificthrust = sthrust/(1+B);
%fuelairratio = f;
%vcold = uef;
%vhot = ue;
end
```

Appendix 2: TSFC and ST for All Possible Design Parameter Values

```
sThrust = zeros(9,13,21,6);
sFC = 999999.*ones(9,13,21,6);

for t_max = 1400:50:1800 %9 loops
  for pi_c = 16:2:40 %13 loops
    for beta = 0:0.5:10 %21 loops
      for pi_f = 1:0.2:2 %6 loops
        if isreal(ST(t_max,pi_c,beta,pi_f)) && ST(t_max,pi_c,beta,pi_f) > 0
          sThrust((t_max-1400)/50+1,(pi_c-16)/2+1,beta/0.5+1,round((pi_f-1)/0.2+1)) =
ST(t_max,pi_c,beta,pi_f).*(1+beta);
        end

        if isreal(TSFC(t_max,pi_c,beta,pi_f)) && TSFC(t_max,pi_c,beta,pi_f) > 0
          sFC((t_max-1400)/50+1,(pi_c-16)/2+1,beta/0.5+1,round((pi_f-1)/0.2+1)) =
TSFC(t_max,pi_c,beta,pi_f);
        end
      end
    end

  end
end

invTSFC = 1./sFC;
normInvTSFC = invTSFC./(max(invTSFC,[],'all')); %normalized reciprocal of TSFC data, 1.00 is highest
value
```

Appendix 3: Code for Generating All Contour and Carpet Plots

```
%fpr contour from 1 to 2
%bypass ratio contour from 0 to 5
%to4 = 1700, pic = 22
testrangeST = zeros(701,101); %rows are constant bypass ratio, columns constant FPR
testrangeTSFC = zeros(701,101);
for var1 = 1:701
    for var2 = 1:101
        testrangeST(var1,var2) = ST(1700,22,(var1-1)/100,1+(var2-1)/100);
        testrangeTSFC(var1,var2) = 1000.*TSFC(1700,22,(var1-1)/100,1+(var2-1)/100);
    end
end

[C,h] = contour(1:0.01:2,0:0.01:7,testrangeST,'red');
clabel(C,h);
xlabel("Fan Pressure Ratio");
ylabel("Bypass Ratio Î²");
title("Specific Thrust (^{N}/_{kg/s}) at T_0_4 = 1700K, Ï€_c = 22");

figure();
[C,h] = contour(1:0.01:2,0:0.01:7,testrangeTSFC,'red');
clabel(C,h);
xlabel("Fan Pressure Ratio");
ylabel("Bypass Ratio Î²");
title("Thrust Specific Fuel Consumption (^{kg/s}/_{kN}) at T_0_4 = 1700K, Ï€_c = 22");

Ta = 216.65; %kelvin
Pa = 7231.355; %pascals
M = 1.7;
uIn = M*sqrt(Ta*287*1.4); %inlet velocity
B = 1.5; %bypass ratio
Qr = 45*10^6; %J/kg

%To4 from 1400 to 1800
%compression ratio from 16 to 40
%fpr = 2, bypass ratio = 1.5 carpet plot
testrangeST = zeros(9,13); %rows are constant To4, columns constant pi_c
testrangeTSFC = zeros(9,13);
for var1 = 0:8
    for var2 = 0:12
        testrangeST(var1+1,var2+1) = ST(1400+var1*50,16+var2*2,1.5,2);
        testrangeTSFC(var1+1,var2+1) = 1000.*TSFC(1400+var1*50,16+var2*2,1.5,2);
    end
end
```

```
figure();
plot(testrangeST(1,:),testrangeTSFC(1,:),'red');
text(testrangeST(1,1)-15,testrangeTSFC(1,1)+.0001, 'T_0_4 = 1400K');
hold on
for temploop = 2:9
    plot(testrangeST(temploop,:),testrangeTSFC(temploop,:),'red');
    text(testrangeST(temploop,1)-15,testrangeTSFC(temploop,1)+.0001, append('T_0_4 =
',num2str(1400+(temploop-1)*50),'K'));
end
for presloop = 1:13
    plot(testrangeST(:,presloop),testrangeTSFC(:,presloop),'blue');
    text(testrangeST(9,presloop),testrangeTSFC(9,presloop)-.0001, append('Ï€_c =
',num2str(16+(presloop-1)*2)));
end
xlim([60 360]);
ylim([.012 .028]);
xlabel("Specific Thrust (^{N}/_{kg/s})");
ylabel("Thrust Specific Fuel Consumption (^{kg/s}/_{kN})");
title("Carpet Plot of TSFC and ST vs T_0_4 and Ï€_c for Ï€_f = 2 and Î² = 1.5");

density = Pa/(287*Ta);
diameter = 1.6;
area = pi*(diameter/2)^2;
STmin = 80000/((density*uIn*area)*(1+B));
xline(STmin,"--k",'Minimum Specific Thrust');

LD = 7.5;
fuelFraction = 0.4;
TSFCmax = (log(1/(1-fuelFraction))*LD*uIn/9.81)/8000;
yline(TSFCmax,"--k", {'Maximum','TSFC'});
hold off

%contours in pi_c T04 plane
testrangeST = zeros(401,2401); %rows are constant temp, columns constant compression ratio
testrangeTSFC = zeros(401,2401); %rows are constant temp, columns constant compression ratio
testrangeThermEff = zeros(401,2401); %rows are constant temp, columns constant compression ratio
testrangePropEff = zeros(401,2401); %rows are constant temp, columns constant compression ratio
testrangeOverallEff = zeros(401,2401); %rows are constant temp, columns constant compression ratio

for var1 = 1:401
    for var2 = 1:2401
        testrangeST(var1,var2) = ST(1400+var1-1,16+(var2-1)/100,1.5,2);
        testrangeTSFC(var1,var2) = 1000.*TSFC(1400+var1-1,16+(var2-1)/100,1.5,2);
            [uCold,uHot] = exitVel(1400+var1-1,16+(var2-1)/100,1.5,2);
            f = fuelair(1400+var1-1,16+(var2-1)/100,1.5,2);
            deltaKE = 0.5*((1+f)*uHot^2+B*uCold^2-(1+B)*uIn^2);
```

```
        energyConsump = f*Qr;
      testrangeThermEff(var1,var2) = deltaKE/energyConsump;
        thrust_ma = testrangeST(var1,var2);
        thrustPower = thrust_ma*uIn;
      testrangePropEff(var1,var2) = thrustPower/deltaKE;
      testrangeOverallEff(var1,var2) = thrustPower/energyConsump;
   end
end

figure();
[C,h] = contour(16:0.01:40,1400:1:1800,testrangeST,'red');
clabel(C,h);
xlabel("Compression Ratio Ï€_c");
ylabel("Max Temperature T_0_4");
title("Specific Thrust (^{N}/_{kg/s}) for Î² = 1.5, FPR = 2");

figure();
[C,h] = contour(16:0.01:40,1400:1:1800,testrangeTSFC,'red');
clabel(C,h);
xlabel("Compression Ratio Ï€_c");
ylabel("Max Temperature T_0_4");
title("Thrust Specific Fuel Consumption (^{kg/s}/_{kN}) for Î² = 1.5, FPR = 2");

figure();
[C,h] = contour(16:0.01:40,1400:1:1800,testrangeThermEff,'red');
clabel(C,h);
xlabel("Compression Ratio Ï€_c");
ylabel("Max Temperature T_0_4");
title("Thermal Efficiency for Î² = 1.5, FPR = 2");

figure();
[C,h] = contour(16:0.01:40,1400:1:1800,testrangePropEff,'red');
clabel(C,h);
xlabel("Compression Ratio Ï€_c");
ylabel("Max Temperature T_0_4");
title("Propulsion Efficiency for Î² = 1.5, FPR = 2");

figure();
[C,h] = contour(16:0.01:40,1400:1:1800,testrangeOverallEff,'red');
clabel(C,h);
xlabel("Compression Ratio Ï€_c");
ylabel("Max Temperature T_0_4");
title("Overall Efficiency for Î² = 1.5, FPR = 2");
```