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Spreadsheet Modelling of Turbojet Performance*

C. N. REFFOLD
Royal Military College of Science, Cranfield University, Shrivenham, Swindon, UK

This paper develops a simple spreadsheet model of the turbojet engine for use by students of gas turbine performance and design. Spreadsheet modelling is a useful educational tool since it allows the student to monitor gas properties throughout the cycle without laborious calculation. The model allows the choice of key design point parameters and computes the cycle performance. A simple scheme is then presented to model performance at off design point, i.e. at widely varying temperatures, pressures, rotational speeds and Mach numbers.

1. The paper describes software applications useful in the following engineering disciplines:
   Thermodynamics, gas turbine theory, aircraft propulsion, mechanical engineering.
2. The paper is suitable for teaching/classwork/self-study for engineering students at the following level:
   Final-year undergraduate and postgraduate.
3. What aspects of your contribution are new?
   Off design analysis is of a turbojet engine has not been presented on a spreadsheet program before. The paper also presents a more simple iterative method of off design solution than has been published before.
4. How is the material as presented to be incorporated in engineering teaching?
   The material can be presented in a computer 'workshop' as a reinforcement of theory lectures. The material could also be used for student self-study. It could accompany experimental work in running a gas turbine engine.
5. Have the concepts presented been tested in the classroom or in project work?
   The design point work has been tested and was successful with undergraduates in the classroom, working on their own and in an individual project.
6. What conclusions have been drawn from the experience?
   Students have found the method agreeable since they do not have to do large numbers of tedious calculations in order to produce results.
7. Other comments on the benefits of your approach for engineering education:
   The use of spreadsheets for modelling performance is very useful for many different types of system and problems. For example, the author has used spreadsheets for applying the factorial method of analysis of diesel engine performance. This reduced the student calculation time from 9 h to 30 min. This enables the student to understand the problem without duplication of effort. The student stays focused on the higher-level learning objectives and is not 'turned off' by tedium.

NOMENCLATURE

A	area
C	velocity
CR	compressor pressure ratio
C_p	specific heat at constant pressure
C_v	specific heat at constant volume
D	diameter
Δ	Delta, increase or decrease (in property)
DP	design point
F	thrust
γ	gamma, ratio of specific heats, C_p/C_v
h	specific enthalpy
η	eta, efficiency
k	a constant
LCV	lower calorific value (specific energy)
M	flight Mach number
$\dot{m}$	mass flow rate (air, gas, fuel)
OPL	overall pressure loss (combustor)
p	pressure
ρ	rho, density
R	specific gas constant
RPM	rotational speed in rev./min
$SDMF$	semi-dimensional mass flow
SFC	specific fuel consumption
T	temperature
TET	turbine entry temperature
TTR	turbine temperature ratio
$\dot{w}$	power (work rate)

Subscripts

1–5	station number (see Fig. 1)

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a	adiabatic (efficiency), air (pressure, temperature, velocity)
c	compressor, critical
comb	combustor
t	turbine
poly	polytropic (efficiency)
r	required (thrust)
s	static (temperature and pressure), specific (thrust power)

INTRODUCTION

THE TURBOJET engine is a relatively simple power plant which is used for a variety of tasks varying from aircraft propulsion to gas pumping. Modelling the performance of the turbojet is an important part of the industrial design process but is also very valuable for education and training. To this end, the model is simplified by omitting such considerations as mechanical power extraction, cooling and utility bleed air, friction in the exhaust duct and the convergent nozzle.

The use of computers greatly improves the speed of calculations and requires alternative techniques to replace the use of, for example, temperature rise charts and characteristic charts. The use of spreadsheets allows all major parameters to be monitored as design decisions are made and key design parameters are altered. This is particularly useful in an educational context.

This paper presents a spreadsheet that is primarily aimed towards students of aeronautical engineering and can be used for classwork and self-study. It would be particularly valuable if the spreadsheet were given to the student after some experimental work running a gas turbine engine. Work of this kind was done by the author with undergraduate students at Coventry University, although design point (DP) modelling only was covered. The students responded with enthusiasm for the 'new' way of learning and showed a high order of learning outcomes.

The spreadsheet is suitable for final-year undergraduate level and postgraduate propulsion specialist students, and would be of use to students of thermodynamics and turbomachinery in general. It would also be a good example to postgraduate students in engineering design of the complexity of even the most abstract performance modelling with a system as complex as a jet engine.

It is important for students to understand that the design process for a jet engine is a complex team effort. In the first instance, the cycle will be evaluated at DP. When a satisfactory DP performance has been found, the cycle will be modelled at off DP conditions. The final design is one that gives the best overall performance at DP and over a wide range of off DP conditions. The methods presented in this paper allow rapid simulation of the turbojet's performance to develop understanding of how performance will vary over that range of conditions.

Design point spreadsheet modelling, for turbofans, has previously been presented by Weston [1]. This paper goes further by offering a simple scheme for determining off DP performance for the turbojet cycle.

The DP calculations are based upon one-dimensional methods put forward by Oates [2], Harman [3] and Cohen *et al.* [4]. The off DP calculations and methods are based upon the work of Oates and of Mattingly *et al.* [5] as well as the methods taught on the gas turbine performance course at Cranfield University. However, the new off DP method used in this paper uses a different semi-dimensional mass flow parameter to determine compressor pressure ratio. This avoids the use of 'nested' iterative calculations which simplifies the solution process.

ENGINE DESCRIPTION

The engine's key components are shown in Figure 1; stations before and after key components are numbered as shown. The station number is subscripted to denote pressures and temperatures at that station, e.g. T_3 is the stagnation, or total, temperature at turbine entry. The static pressure at nozzle exit is p_{s5}. Note that we work mainly with stagnation or total conditions except at the inlet and outlet.

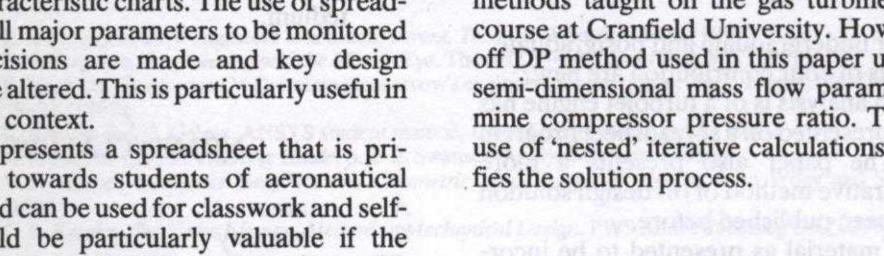

Fig. 1. Schematic diagram of turbojet engine.

DESIGN POINT ANALYSIS

Design point specification

First, a DP flight altitude and Mach number (M) are specified. In a real design specification the DP and much more detailed information is specified; reference [5] gives detail on the US Department of Defense 'Request for Proposal' process, those of other nations are broadly similar. By reference to the International Standard Atmosphere (ISA) we can determine air pressure (p_a) and temperature (T_a) at the DP.

Next we must record the following important target efficiencies and pressure loss parameters:

- Mechanical efficiency for the main shaft and combustion efficiency for the combustion chamber (combustor).
- Polytropic efficiencies for the compressor and the turbine.
- Maximum pressure ratio for the intake.
- Maximum allowable overall pressure loss (OPL) for the combustor. This is normally expressed as percentage of entry pressure.

The next stage is to find suitable values for air and fuel data. C_p for air, and consequently γ (the ratio of specific heats), varies as a function of temperature. Computational methods for their calculation are available [6] but are unnecessary for this simple model. The following values are used:

- The specific gas constant, R (SGC R in the spreadsheet) for dry, unvitiated air is 287 J/kgK.
- Typical values of C_p are 1005 J/kgK for the cold and 1150 J/kgK for the hot sections
- γ can be found for each value of C_p since $\gamma = C_p/C_v = C_p/(C_p - R)$.
- The fuel heating value is normally taken as the lower calorific value (LCV) in gas turbine modelling. A typical value for an aviation kerosene type fuel would be 42.8 MJ/kg.

The designer must make the key design choices of the engine before modelling the engine performance; these are compressor pressure ratio (*CR*) and turbine entry gas temperature (*TET*).

Dynamic recovery

Preliminary calculations start with recovery of the air pressure and temperature to total conditions. As the airflow stagnates from flight true airspeed (TAS) to relative speed zero, the temperature and pressure increase as functions of airspeed (and Mach Number, *M*).

The *TAS* (C_a) will need to be evaluated for thrust calculation and can be found by:

$$C_a = M\sqrt{\gamma R T_a} \tag{1}$$

Houghton and Brock [7] showed that the following equations can be derived from the law of conservation of energy:

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

$$\frac{p_0}{p_a} = \left[\frac{T_0}{T_a}\right]^{\gamma/(\gamma-1)} \tag{3}$$

Intake

The performance of the intake system can be modelled by using the US DoD military specification MIL-E-5008 [5]:

If $M > 1$ then

$$\frac{p_1}{p_0} = \left(\frac{p_1}{p_0}\right)_{\max}\{1 - 0.075(M-1)^{1.35}\} \tag{4}$$

else

$$\frac{p_1}{p_0} = \left(\frac{p_1}{p_0}\right)_{\max} \tag{5}$$

Assuming that the intake (diffuser) process is isentropic, then $T_1 = T_0$.

Compressor

$CR(p_2/p_1)$ is already decided for the compressor. Now, to determine the temperature rise using the polytropic efficiency:

$$\frac{T_2}{T_1} = \left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma\eta_{\text{poly}}} \tag{6}$$

The power consumed by the compressor is found by calculating the enthalpy rise:

$$\dot{W}_c = \dot{m}C_p\Delta T \tag{7}$$

At the initial stage we can compute specific compressor power by omitting the mass flow term.

The adiabatic efficiency for the compressor is found by the following equation:

$$\eta_{ca} = \frac{(CR^{(\gamma-1)/\gamma} - 1)}{\left(\frac{T_2}{T_1} - 1\right)} \tag{8}$$

Combustor

Combustor exit pressure, p_3, is found by subtracting the design specification combustor overall pressure loss (*OPL*) from the compressor exit pressure p_2. *OPL* is usually expressed as a percentage of compressor exit pressure:

$$p_3 = p_2\frac{(100 - OPL)}{100} \tag{9}$$

T_3 is the *TET*, chosen by the designer. The air/fuel ratio (*AFR*) required to produce this temperature rise can be found by a variety of methods. Harman [3] contributes a useful correlation:

If ΔT is more than 10K and less than 400K, then:

$$\frac{\dot{m}_{\mathrm{f}}}{\dot{m}_{\mathrm{a}}}=\frac{990(\Delta T-10)\left(\frac{T_2}{3250}+1\right)}{\mathrm{LCV}\eta_{\mathrm{comb}}} \tag{10}$$

else if ΔT is greater than or equal to 400K but less than 900K:

$$\frac{\dot{m}_{\mathrm{f}}}{\dot{m}_{\mathrm{a}}}=\frac{1100(\Delta T-50)\left(\frac{T_2}{3250}+1\right)}{LCV\eta_{\mathrm{comb}}} \tag{11}$$

Turbine

The temperature drop across the turbine is a function of the enthalpy extracted to drive the compressor. We have previously calculated the specific compressor power. However, we must take into account the mechanical efficiency of the main shaft. So, the specific power required from the turbine is given by:

$$\dot{w}_{\mathrm{st}}=\frac{\dot{w}_{\mathrm{sc}}}{\eta_{\mathrm{shaft}}} \tag{12}$$

which equates to the specific enthalpy drop in the turbine:

$$\Delta h_{\mathrm{t}}=C_{\mathrm{pt}}(T_3-T_4) \tag{13}$$

thus:

$$T_4=T_3-\frac{\dot{w}_{\mathrm{sc}}}{\eta_{\mathrm{shaft}}C_{\mathrm{pt}}} \tag{14}$$

The pressure drop in a real cycle will be greater than the isentropic case:

$$\frac{p_4}{p_3}=\left[\frac{T_4}{T_3}\right]^{(\gamma/(\gamma-1)\eta_{\mathrm{poly}})} \tag{15}$$

The adiabatic efficiency for the turbine is found by dividing the isenotropic temperature drop by the actual temperature drop:

$$\eta_{\mathrm{ta}}=\frac{1-\frac{T_4}{T_3}}{1-\left(\frac{p_4}{p_3}\right)^{(\gamma-1)/\gamma}} \tag{16}$$

Note that the manually calculated result may vary by up to 10% at this stage, depending on the number of significant figures used in the working. The spreadsheet should be very accurate as most such software automatically detects maths co-processors and uses double-precision arithmetic.

Jet pipe and nozzle

If we ignore frictional losses in the jet pipe, then total temperature and pressure at station 5 will be equal to those at station 4.

In order to calculate the static pressure at exit, p_{s5}, we first calculate p_{crit}, the pressure of the expanded flow at the choked condition ($M=1$). If that pressure is less than the static air pressure, then the nozzle will expand the flow only to static air pressure ($M<1$).

The critical pressure would be:

$$p_{\mathrm{c}}=p_5\left[1-\left(\frac{\gamma-1}{\gamma+1}\right)\right]^{\gamma/(\gamma-1)} \tag{17}$$

and if this is greater than the static air pressure, p_{a}, then exit flow is choked and that will be the exit static pressure, p_{s5}.

Should the critical pressure be less than p_{a}, then the nozzle will only allow expansion to p_{a} and there will, therefore, be no component of pressure thrust.

The total temperature at nozzle exit, T_5, is the same as at turbine exit. However, if the flow is choked, the static temperature can be found thus:

$$T_{\mathrm{s5}}=T_5\left(\frac{2}{\gamma+1}\right) \tag{18}$$

If the flow is not choked, the exit static temperature is found by:

$$T_{\mathrm{s5}}=T_5\left(\frac{p_{\mathrm{a}}}{p_5}\right)^{(\gamma-1)/\gamma} \tag{19}$$

The equation of state enables us to find the density at the exit:

$$\rho_{\mathrm{s5}}=\frac{p_{\mathrm{s5}}}{(RT_{\mathrm{s5}})} \tag{20}$$

We can find exit velocity for choked flow by:

$$C_5=\sqrt{\gamma RT_{\mathrm{s5}}} \tag{21}$$

If the flow is not choked, then:

$$C_5=\sqrt{2C_{\mathrm{p}}(T_5-T_{\mathrm{s5}})} \tag{22}$$

PERFORMANCE CALCULATIONS

The pressure component of thrust is $A(p_{\mathrm{s5}}-p_{\mathrm{a}})$ where A is the nozzle exit area. To find specific thrust due to pressure we must divide by mass flow rate.

Thus, since $A=\dot{m}/\rho C$, the pressure component of specific thrust is given by $(p_{\mathrm{s5}}-p_{\mathrm{a}})/(\rho_{\mathrm{s5}}C_5)$.

The momentum component of thrust is $\dot{m}(C_5-C_{\mathrm{a}})$ and we divide by $\dot{m}$ to give specific thrust due to momentum. Thus, total F_{s} is given by:

$$F_{\mathrm{s}}=\frac{(p_{\mathrm{s5}}-p_{\mathrm{a}})}{(\rho_{\mathrm{s5}}C_5)}+(C_5-C_{\mathrm{a}}) \tag{23}$$

and

$$SFC = \frac{\dot{m}_f}{F} = \frac{FAR\,\dot{m}_{air}}{F_s \dot{m}_{air}} = \frac{FAR}{F_s} \tag{24}$$

Note that we have now calculated the performance of the engine without using an air mass flow rate figure or exit nozzle area. We can now set the mass flow rate in order to achieve the required thrust F_r.

$$\dot{m}_{air} = \frac{F_r}{F_s} \tag{25}$$

Once we have arrived at a suitable mass flow rate we can calculate exit area using the continuity equation. Here we use the total mass flow, including the fuel added:

$$\dot{m}_{total} = \dot{m}_{air}(1 + FAR) \tag{26}$$

giving

$$A = \frac{\dot{m}_{total}}{(\rho_5 C_5)} \tag{27}$$

This nozzle area will be fixed at DP and off DP. The nozzle diameter is given by:

$$D = \sqrt{4A/\pi} \tag{28}$$

OFF DESIGN POINT ANALYSIS

Assumptions

Some workers have, in the past, used the DP semi-dimensional mass flow (SDMF), $\dot{m}\sqrt{T}/p$, for turbines and nozzles as data in order to determine *CR*, *TET* and turbine temperature ratio (*TTR*) in off DP calculation. Unfortunately this requires a 'nested' iterative process where a guess is made at *CR* followed by iteration of *TET* and *TTR* to match compressor power; this is then followed by a refined guess at *CR* to repeat the iterative cycle. Thus, the solution process can be time consuming and complex.

The analysis of performance at off DP conditions can be simplified if we make some reasonable assumptions. Inspection of characteristic charts and contemporary gas turbine performance shows that the following assumptions are reasonable:

- Compressor and turbine will be choked at all steady-state running conditions in the single spool turbojet. Thus, turbine temperature and pressure ratios will remain constant.
- Compressor and turbine adiabatic efficiencies remain constant. In fact, at reduced mass flows and rotational speeds, the efficiencies will vary slightly. Generally, in steady-state operating conditions below DP the efficiencies will improve, with the advantage that off DP performance estimates will be slightly pessimistic.
- The ratio of DP and off DP semi-dimensional mass flows (*SDMF*), $\dot{m}\sqrt{T}/p$, is constant for and proportional to the square of the ratio of DP and off DP semi-dimensional shaft speed, $N/\sqrt{T}$.

This last assumption is a broad simplification since the relationship between mass flow and shaft speed is dependent upon the DP compression ratio. It is, however, sufficiently accurate to produce meaningful results without recourse to a compressor characteristic map.

In addition to these assumptions, it can be shown that compressor *SDMF*, $\dot{m}\sqrt{\Delta T}/p_2$, will remain constant, i.e. the off DP value will equal the DP value.

The consequence of these points is that we can simply adjust *CR*, calculating temperature rise, until *SDMF* matches the DP datum. Then, we can adjust *TET*, calculating temperature drop across the turbine to match compressor power consumption, until the *TTR* matches the DP datum. This simplified approach makes the solution process very much quicker.

Off design point specification

Off DP temperature, pressure and Mach number are specified. These enable the calculation of p_1, T_1 and $TAS(C_a)$ using equations (1)–(5). Off DP mass flow is found, using p_1 and T_1, by the following method:

$$\dot{m}\sqrt{T}/p = kN^2/T$$

thus

$$\left(\frac{\dot{m}T^{3/2}}{pN^2}\right)_{\text{off DP}} = k = \left(\frac{\dot{m}T^{3/2}}{pN^2}\right)_{\text{DP}} \tag{29}$$

$$\dot{m}_{\text{off DP}} = \dot{m}_{\text{DP}}\left(\frac{T_{\text{DP}}}{T_{\text{off DP}}}\right)^{3/2}\left(\frac{p_{\text{off DP}}}{p_{\text{DP}}}\right)\left(\frac{N_{\text{off DP}}}{N_{\text{DP}}}\right)^2 \tag{30}$$

Cycle calculations

The DP compressor *SDMF*, $\dot{m}\sqrt{\Delta T}/p_2$, and *TTR* should be calculated. These must remain the same at off DP; changes in estimates of *CR* will be made in order to balance the *SDMFs*. *TET* estimates will be made to equalize the *TTRs*.

Once the mass flow is known, the temperature and pressure ratios for the cycle can be found; the simplest method is by iteration in a spreadsheet. With estimated *CR* and *TET* put into the program, the temperature and pressure changes for compressor and turbine can be calculated. Obviously, p_1 is multiplied by *CR* to yield p_2. Temperature rise for the compressor is calculated using the adiabatic efficiency (equation 8). Thus:

$$T_2 = T_1 + \frac{T_1}{\eta_{ca}}\left[\left(\frac{p_2}{p_1}\right)^{\gamma-1/\gamma} - 1\right] \tag{31}$$

TET (T_3) is chosen and p_3 is found by equation (9). It is unnecessary to calculate *FAR* until a final figure for *TET* is arrived at. Temperature drop across the turbine is found by reference to the compressor specific power:

$$T_4 = T_3 - \frac{C_{pc}(T_2 - T_1)}{C_{pt}\eta_{shaft}} \qquad (32)$$

Pressure drop for the turbine is calculated thus:

$$p_4 = p_3 \left[1 - \frac{1 - T_4/T_3}{\eta_{ta}} \right]^{\gamma/\gamma-1} \qquad (33)$$

With T_2, p_2, T_4 and p_4 known, the *SDMF* is evaluated and a new *CR* estimated until the off and DP values are closely similar.

When the *CR* is finalized, the *TET* is found by iteration until off and DP *TTR* are matched within acceptable limits.

RESULTS

The performance of the engine is calculated in a similar manner to the DP case, except that the nozzle area is fixed. Equations (10) or (11) are used to determine *FAR*. Equations (17)–(24) are evaluated using the final values for the properties at exit, to yield *Fs* and *SFC*.

Off DP thrust is found by adding the pressure thrust to the momentum thrust:

$$F = A(p_{s5} - p_a) + \dot{m}_{air}(1 + FAR)/(C_5 - C_a) \qquad (34)$$

USING THE SPREADSHEET

Layout

The original spreadsheet was developed for the Lotus 1-2-3 program (release 2.01). This program produces data files with a .WK1 filename extension which are readily imported by most other spreadsheet programs. However, other formats may change layout, colours and symbology, so that some of the following instructions are not valid.

The spreadsheet is developed in three screens (see Appendix for spreadsheet format). The DP data input screen starts at cell A1. The DP results screen is directly below the input screen and is found by actuating the Page Down key. The off DP screen is directly below again and is accessed in the same way

DP input

On screen, the cells containing the values to be entered are unprotected and are presented in contrasting colour. The user should enter suitable values, and observe the interim calculations.

When complete, the user should hit the Page Down key to move to the next section.

DP results

The temperatures and pressures at each station are shown in the Results section along with other key data. The user is required to enter an air mass flow rate in order to obtain a final thrust figure. When this is done, the spreadsheet will also compute nozzle area and diameter. Note that thrust is given in kN and in thousands of pounds since many aeronautical gas turbine users commonly use that unit for comparative purposes.

Off DP

The spreadsheet user must first define the off DP Mach, pressure, temperature and %RPM. The DP values and properties are given in the spreadsheet for user reference. Next, the estimates of *CR* and *TET* (T_3) must be made.

An increased estimate of *CR* will result in a decreased *SDMF*. The solution is quickly found; two decimal places (d.p.) is more than satisfactory resolution.

TET is then found in a similar manner, although increasing *TET* increases *TTR*.

Note that the spreadsheet includes instruction on the sense of the effect of changing *CR* and *TET* on the *SDMF* and *TTR*. A little practice soon allows the user to reach quick solutions.

SPREADSHEET FILE AVAILABILITY

Copies of the spreadsheet file are available from the author on request at a small charge to cover costs.

CONCLUSIONS

1. The spreadsheet referred to in this paper has been used by undergraduate students (for DP calculations only) who were able to achieve high order learning outcomes. Students were enthusiastic about this learning method since it saved them many tedious hours of calculations. The spreadsheet allows the student to see gas properties throughout the cycle as they change key design choices.
2. The spreadsheet is primarily aimed toward students of aeronautical engineering and can be used for classwork and self-study. It would be particularly valuable if the spreadsheet were given to the student after some experimental work running a gas turbine engine.
3. The spreadsheet is suitable for final-year undergraduate level and postgraduate propulsion specialist students, and would be of use to students of thermodynamics and turbomachinery in general. It would also be a good example to postgraduate students in engineering design of the complexity of even the most abstract performance modelling with a system as complex as a jet engine.
4. The new off DP method used in this paper uses the semi-dimensional mass flow parameter, $\dot{m}\sqrt{\Delta T}/p_2$, to determine compressor pressure ratio. This avoids the use of 'nested' iterative calculations and simplifies the solution process.

APPENDIX

```
Turbojet Design Spreadsheet by N. Reffold                    Written:       10/28/93

First we define the Design Point...
 Mach No:          0.84        Ta:      255.70  K                 0.5405  bar
          and calculate TAS (Ca) :      269.73  m/s

Next, some efficiency and pressure loss targets...
                         Efficiencies for shaft :          0.99          and combustor :      0.98
          Polytropic efficiency for compressor :          0.91           and for turbine :     0.88
                                 Intake PR max :          0.97      Combustor OPL / % :      4.00

Air Properties...          r means gamma ---->          r/(r-1)      (r-1)/r     SGC R :    287.00
Cp comp :           1005  Gamma (r)           1.40        3.50       0.2856  Cp's in J/kgK
Cp turb :           1148  Gamma (r)           1.33        4.00       0.2500

Fuel LCV:          43.00  ( MJ/kg)

Now the principal design choices.       Compression Ratio:          8.00        TET/K:    1200.00

That completes the design point input.          Now, hit Page Down to see results.

  Pressures / bar           Temperatures / K                      Performance Results
   p air           0.54       T air           255.70
     p0            0.86         T0            291.90    Comp. specific power  / (kW/kg) :      272.06
     p1            0.83         T1            291.90       Comp.  adiabatic efficiency :         0.87
     p2            6.67         T2            562.60                    Fuel/air ratio :      0.01799
     p3            6.40         T3           1200.00     Turbine  adiabatic efficiency :        0.89
     p4            2.33         T4            960.62          Exit density / (kg/cu.m) :      0.5318
     p5            2.33         T5            960.62             Exit velocity / (m/s) :      561.32
   pcrit           1.26  flow choked                                      Fs / (Ns/kg) :      531.53
  p5 stat          1.26     T5 stat           823.39                     SFC / (kg/hN) :      0.1218


Enter mass flow to size the engine :          92.50  kg/s

 Fuel flow :          1.66  kg/s                          Thrust ...         50.05  kN
                                                                           11.25  thou. lbs

Nozzle ...      area/m² :         0.32  diam/m :          0.63

Now, hit Page Down to complete off design point computation...
          DP Summary       #######################    Off Design Point     #########################
        Mach :       0.8416       0.5000      % RPM :         90                     Ts5 :      659.33
       p air :       0.5405       1.0100         p0 :       1.20            Exit density :      0.6363
       T air :       255.70       288.00         T0 :     302.39           Exit velocity :      502.30
          p1 :         0.83         1.16        TAS :     160.25
          T1 :       291.90       302.39        FAR :    0.01186
        flow :        92.50        99.07        ps5 :       1.20  choked
          CR :        8.000        5.500
          p2 :         6.67         6.39
       eta c :         0.87         0.87              ######## Results ###########
          T2 :       562.60       519.27      Fs / (Ns/kg) :      402.77
   T3 (TET) :      1200.00       961.00     SFC / (kg/hN) :      0.1060
          p3 :         6.40         6.14         Thrust /kN :      40.41
          T4 :       960.62       769.22                            9.08  thou. lbs
       eta t :         0.89         0.89
          p4 :         2.33         2.23
      SDMF 3 :       228.24       228.27  Increase CR  to decrease SDMF 3
         TTR :       0.8005       0.8004  Increase CR to increase Turbine Temperature Ratio.
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C. N. Reffold served as a Royal Air Force fighter navigator for 15 years before retiring through invalidity. His service included tours on several front-line squadrons as well as posts on the Qualified Weapons Instructors staff and as a Staff Navigator. After leaving the RAF, the author taught engineering at Coventry University, specializing in design, experimental methods and applied mechanics. He was latterly course tutor of a B.Eng. course in aerospace systems engineering. Mr Reffold joined the Design Group of the School of Mechanical, Materials and Civil Engineering at the Royal Military College of Science in January 1994.

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C. N. Reffold served as a Royal Air Force fighter navigator for 15 years before retiring through invalidity. His service included tours on several front-line squadrons as well as posts on the Qualified Weapons Instructors staff and as a Staff Navigator. After leaving the RAF, the author taught engineering at Coventry University, specializing in design, experimental methods and applied mechanics. He was latterly course tutor of a B.Eng. course in aerospace systems engineering. Mr Reffold joined the Design Group of the School of Mechanical, Materials and Civil Engineering at the Royal Military College of Science in January 1994.