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ADVANCED USER-FRIENDLY GAS TURBINE PERFORMANCE CALCULATIONS ON A PERSONAL COMPUTER

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ABSTRACT

Personal Computers are nowadays very powerful. They allow us to do complete gasturbine performance calculations both for the engine design point and the partload behavior. A typical helicopter engine serves as an example for a cycle study. The simple variation of compressor pressure ratio and turbine inlet temperature, however, does not yield a realistic result. Only after including the effects of variable amounts of cooling air needed for constant metal temperature and turbine efficiency as a function of aerodynamic loading one does get results that are in line with the cycle of real engines.

Off-design performance calculations need no longer use crude simplifications for reasonable calculation times. Real component maps can be used while the matching of the cycle is done by iterative methods. Transient simulations on a 486DX machine are only a matter of a few minutes. Even the effects of inlet distortion (pressure or temperature) can be dealt with by using the parallel compressor model.

The key to userfriendliness is, to hide everything from the user that he does not need to know at the very moment. The things he needs to know, however, must be presented in clear and easily understandable expressions. Results must be shown in graphics whenever it makes sense. This is the only way to recognize quickly problem areas or to convince people, that the selected design is optimal.

1. INTRODUCTION

Universities, research institutions as well as the industrial sector use large computer programs for the calculation of design and off-design performance of gas turbine engines. In general, these programs require many input data and are quite difficult to use. GASTURB is a simplified but accurate gas turbine cycle calculation program for IBM compatible PCs. It is very user-friendly and easy to use even for people who only have limited experience with personal computers.

The program is ideally suited for basic studies and for getting a fundamental understanding of the principles underlying the design and operation of gas turbines. It is also suited for professional applications in the gas turbine industry.

The differences between too simplistic and more realistic cycle optimization studies will be demonstrated for the example of a typical helicopter engine. Off-design calculations using real component maps are described in the following part.

For getting an insight into the problematic of engine stability the parallel compressor model is a very useful tool. It is incorporated into the program and allows to show the effects of distorted inlet flow on the compressor operating points in their maps. One can for example see, how a pressure distortion in the low pressure compressor is converted into a combined pressure and temperature distortion for the high pressure compressor of a two-spool engine.

Another effect that is important for engine stability is the transient excursion of the compressor operating points in their maps during accelerations and decelerations. A personal computer with a 486DX processor can do transient calculations very quickly. There is only a factor of approximately three to eight between real time and simulation time in spite of the fact, that also for the transient calculations the same detailed thermodynamic model as for steady state is being used.

Many results of the program are presented in graphics. One can easily select any parameter and plot it over any interesting quantity. Thus, one can find quickly problem areas as for example the exceedance of temperature or spool speed limits. Some important graphs like operating lines in compressor maps or specific fuel consumption over thrust are available from a click with the mouse button.

Userfriendliness of a program is not only nice to have. It undoubtedbly improves the quality of the results because in a given time one can check more alternatives more thoroughly. So investment into very sophisticated modeling of the last detail of physics can be less economical than putting more effort into the "front end" and the postprocessing of the program.

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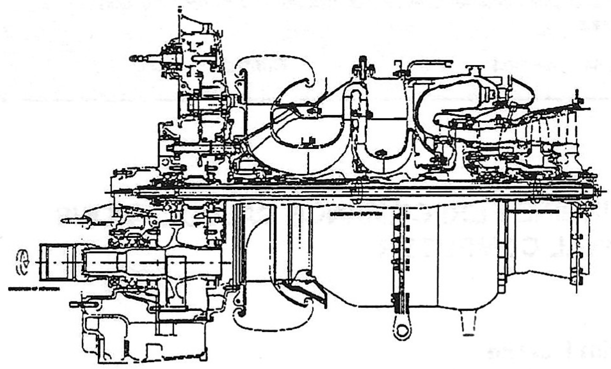

Figure 1: MTR390 CROSS SECTIONAL VIEW

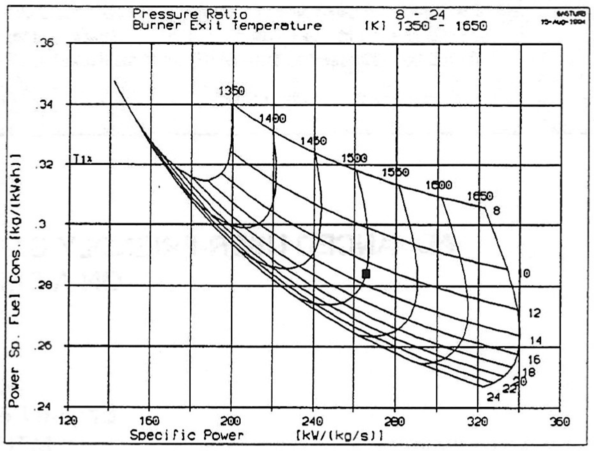

Figure 2: SIMPLE CYCLE PARAMETER STUDY

2. CYCLE OPTIMIZATION FOR A HELICOPTER ENGINE

A typical helicopter engine is composed of a gas generator and a power turbine. Figure 1 shows an example of such an engine. Another example in the same power class is the T800 which differs from the MTR390 by its integral particle separator and a two-stage gas generator turbine. Furthermore it has no gearbox for reducing the output shaft speed. The cycle study to be discussed is based on the following data that are typical for a real engine cycle in the 1000kW power class:

• Take-off power	929	kW
• Spec. fuel consumption	284	g/kWh
• Mass flow	3.5	kg/s
• Pressure ratio	13	

Selecting an optimum cycle is not a trivial task. At first one has to define what is "optimum". This depends very much on the application. In the end one always wants to minimize cost. They are dependent on many things. The cost of engine operation is mainly fuel and maintenance cost. Procurement cost is another item. Of course even the minimum cost engine has to fulfill the power requirements without exceeding given weight and volume limits.

An optimal engine has a low fuel consumption, low weight and a small volume. It is composed from a minimum number of parts with sufficient life. The selection of materials and manufacturing technology is very much dependent on the intended production price.

2.1 Simple Cycle Parameter Study

The thermodynamic cycle of this type of engine is fairly simple: it is the Joule process. The parameters to be optimized are mainly the pressure ratio of the compressor and the turbine inlet temperature. Figure 2 shows a parameter variation with constant isentropic component efficiencies (η_C=0.8, η_{HT}=0.85, η_{LT}=0.89). The graph is valid for a constant amount of high pressure turbine cooling air of 5%, independent from burner exit temperature. This, however, is unrealistic. In Figure 3 the influence of cooling air on the results is shown for a constant pressure ratio to illustrate this.

What amount of cooling air is required for a given burner exit temperature depends on the cooling technology to be applied, the material to be used and on the number of turbine stages. We will come to that in the next section.

From Figure 2 one can read, that for high specific power (which results in an engine of low weight and volume) one needs a high burner exit temperature. Moreover, the higher the temperature, the lower the specific fuel consumption.

Let us assume now, that for manufacturing cost reasons the maximum tolerable burner exit temperature is 1500K. With this constraint we read from Figure 2 for minimum specific fuel consumption the optimum pressure ratio as approximately 21.

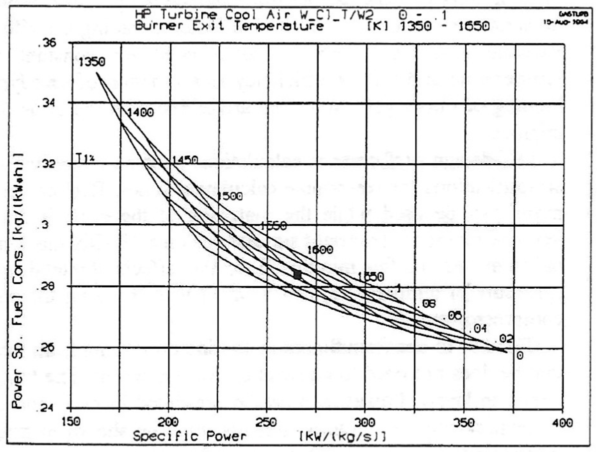

Figure 3: INFLUENCE OF COOLING AIR ON SPECIFIC POWER AND FUEL CONSUMPTION

The engines in this class are operating with a burner exit temperature in the range from 1450K to 1550K, but a much lower pressure ratio than 21. Is it possible, that they are not designed to the optimum cycle? In fact, they are optimum engines. However, a cycle parameter study as shown above is too simplistic for finding a realistic optimum.

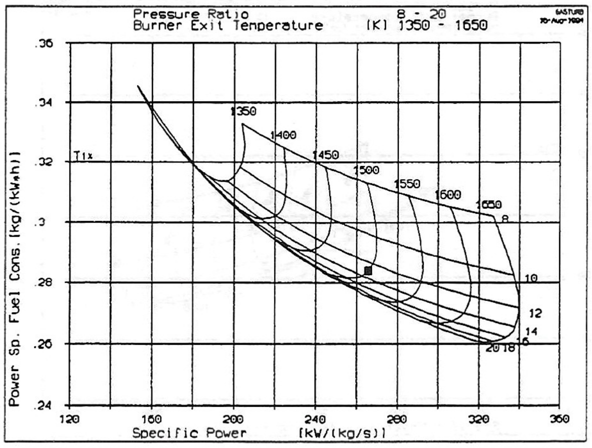

Figure 4: CYCLE STUDY WITH ONE STAGE GAS GENERATOR TURBINES

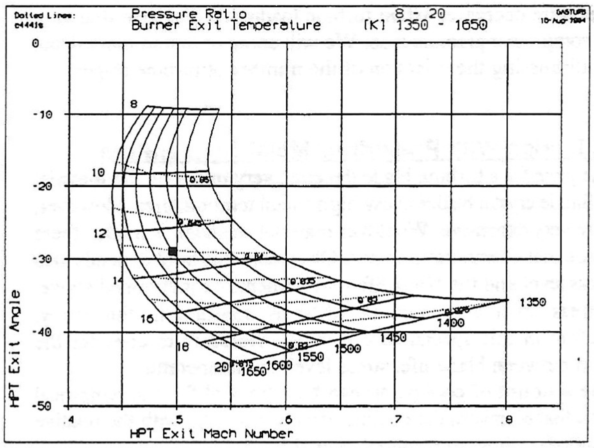

Figure 5: GAS GENERATOR TURBINE EXIT FLOW CONDITIONS

2.2 Realistic Optimization

In Figure 2 constant isentropic component efficiencies are used. This is standard practice, but not fully correct. Rerunning the parameter study with constant polytropic efficiencies leads to an optimum pressure ratio which is only slightly lower than 21. Switching over to polytropic efficiencies is not the key to the right answer. It is necessary, to go more into details. Especially on the turbine side one has to consider the number of stages. Either one uses a one stage turbine or a two-stage turbine. There is no such thing like a 1½ stage turbine. It is necessary, to run the parameter study twice, once for a one stage turbine and a second time for a two-stage turbine.

2.2.1 Engines With Single Stage Turbines

The efficiency of a turbine is very much dependent on its aerodynamic loading and on geometrical parameters. In the GASTURB program there is a preliminary turbine design routine integrated which is based on Ref. 2. The blade loss characteristics are correlated to the mean-section velocity diagrams that are assumed to be symmetrical. The aerodynamic loading is described by a speed-work parameter, defined as the ratio of the overall specific work output to the mean-section blade speed squared. Other factors affecting the calculated efficiency are the number of turbine stages, stator exit angle, Reynolds number and leaving loss.

Primarily the turbine design routine gives relations between important parameters. The absolute numbers calculated for efficiency need adjustment to the technology level considered. That can be done by adaption of an empirical loss factor.

In the following parameter studies some geometrical parameters were fixed in such a way, that for the surrounding of the point marked in all figures of section 2 (pressure ratio 13, burner exit temperature 1500K) there are reasonable velocity triangles calculated by the turbine design routine. Speed is set indirectly by selection of mean axial Mach-number, tip speed and radius ratio at the compressor inlet.

Since engine mass flow is the same for all cycles calculated, we get the same spool speeds and diameters for all engines. With turbine diameter and radius ratio held constant we get also the same turbine exit area A and the same stress level expressed in $A{*}N^2$. Because the specific power varies from cycle to cycle there will be a different absolute shaft power being calculated for each point.

Figure 4, valid for one stage high pressure turbines, is significantly different to Figure 2. The pressure ratio for lowest fuel consumption with burner exit temperature 1500K is now as low as 15. This value is much nearer to the pressure ratio of real engines than the values found to be optimum previously (i.e. 21).

In Figure 5 some features of the one stage turbine can be seen. High Mach-number and swirl cause increased losses in the turbine interduct and possibly in the low pressure turbine. Mach-numbers above 0.5 and swirl angles higher than 30° (relative to the axis) should be avoided. The dotted lines in Figure 5 are lines of constant turbine efficiency.

Another important result is only available when a preliminary turbine design is integrated into the cycle parameter study. Figure 6 shows the mean metal temperature of the rotor blade. Note that for constant burner exit temperature the latter is not constant. That comes mainly from the increase in cooling air temperature when the compressor pressure ratio is raised. An important secondary effect

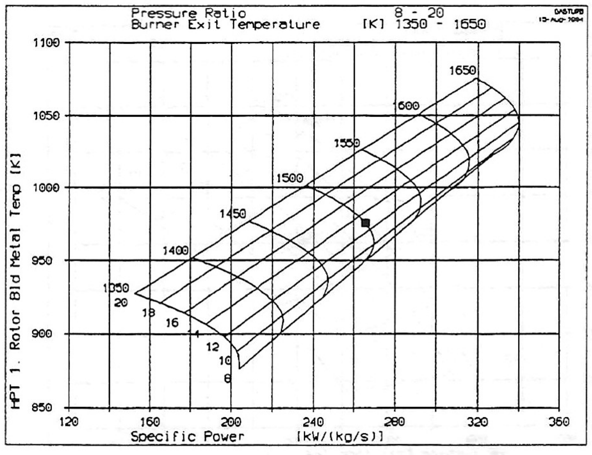

Figure 6: BLADE METAL TEMPERATURE

is, that the ratio of the rotor inlet relative temperature to burner exit temperature decreases when turbine loading is increases due to a rising compressor pressure ratio. We will come to that in more detail when discussing the selection of the number of turbine stages.

2.2.2 Design With Prescribed Metal Temperature

The price for a turbine blade depends very much on the material used. Single crystal blades allow high metal temperatures. However, they are very expensive. Whatever material selection is made, there is a blade metal temperature limit. Obviously this limit depends also on stress level and the blade life requirement. As mentioned above, the stress level expressed in $A*N^2$ is constant in this study. Therefore in this special case it is not necessary to consider the relation between blade life, stress level and temperature.

The amount of cooling air can be estimated from an empirical formula that correlates the cooling effectiveness η_{cl} with the relative amount of cooling air required (Ref. 1):

$$\eta_{cl} = \frac{W_{cl}/W_{ref}}{W_{cl}/W_{ref} + C_{cl}}$$

The constant C_{cl} (order of magnitude 0.04 to 0.07) has to be adjusted to yield reasonable results. The metal temperature is calculated as follows:

$$T_{metal} = T_{rel} - \eta_{cl} * (T_{rel} - T_{coolingair})$$

All the previous parameter studies were done with 5% cooling air. For constant metal temperature one needs a variable amount of cooling air as shown in Figure 6. The cooling effectiveness required is given with Figure 7. The amount of cooling air increases with compressor pressure ratio because the temperature of the cooling air increases.

Figure 8 corresponds with figures 2 and 4. Now the pressure ratio for minimum specific fuel consumption and burner inlet temperature 1500K is only 14. An important observation from this

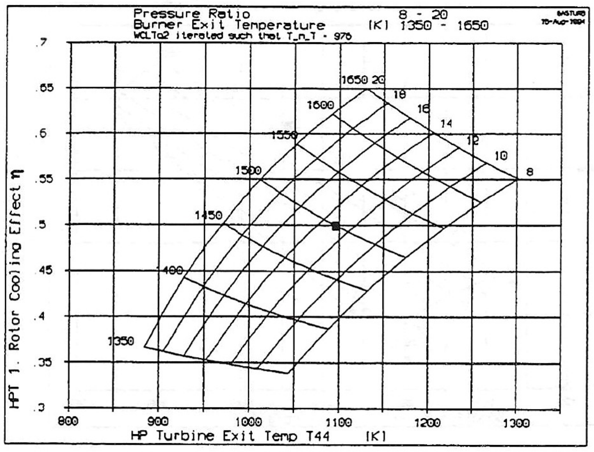

Figure 7: COOLING EFFECTIVENESS REQUIRED

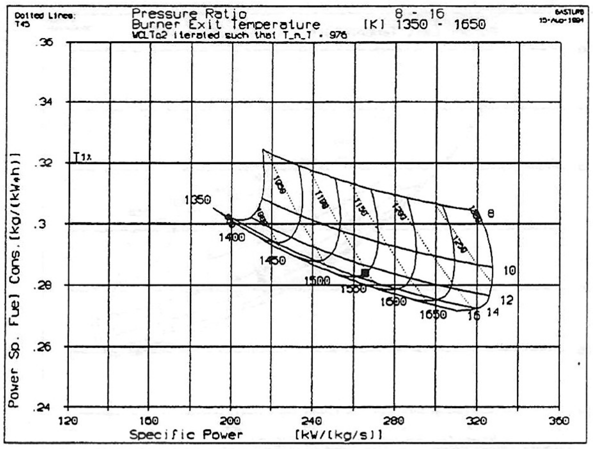

Figure 8: PARAMETERSTUDY FOR SINGLE STAGE GAS GENERATOR TURBINES WITH CONSTANT BLADE METAL TEMPERATURE

figure is, that in spite of the significant increase of the cooling air flow the specific fuel consumption is still decreasing while the burner exit temperature increases. Obviously one cannot easily infer from this parameter study what the best burner exit temperature is for minimum specific fuel consumption. One has to consider other arguments.

One of those can be a limit for the low pressure turbine inlet temperature. For cost reasons an uncooled design will be preferred. If one selects in Figure 8 a maximum LPT inlet temperature of 1100K (the dotted line passing through the reference point), then one gets as optimum engine with respect to fuel consumption a machine with pressure ratio 16 and burner exit temperature 1550K.

However, there are some other points to consider. As can be seen from Figure 7, the step from the reference cycle to the optimum cycle means an increase in cooling effectiveness from 0.5 to 0.56. That can result in a more complex cooling scheme for the rotor blade. It is questionable, whether that is cost-effective.

Increasing the pressure ratio from 13 to 16 means also a compressor of better quality that is most probably more expensive and heavier.

Taking all arguments together leads to the conclusion, that the optimum pressure ratio for a turboshaft with a single stage high pressure turbine is approximately 13 and the most practical burner exit temperature is around 1500K.

2.2.1 Engines With Two Stage Turbines

Selecting the number of turbine stages is not a trivial task. Arguments for the single stage design are less manufacturing and maintenance cost, less cooling air requirements, less weight, volume and polar moment of inertia. In contrast the two-stage design gives a better efficiency, a lower exit Mach number and less swirl in the flow downstream of the turbine.

In Figure 9 the correlations between turbine pressure ratio, efficiency and the relative temperature of the (first) rotor blade are shown for several compressor pressure ratios and the constant burner inlet temperature of 1500K. The next figure presents specific fuel consumption and specific power. The dotted lines are for

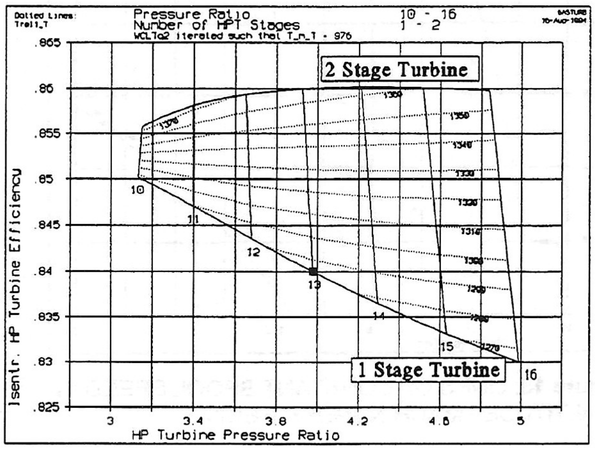

Figure 9: EFFICIENCY AND (FIRST) ROTOR RELATIVE TEMPERATURE FOR SINGLE AND TWO STAGE TURBINES

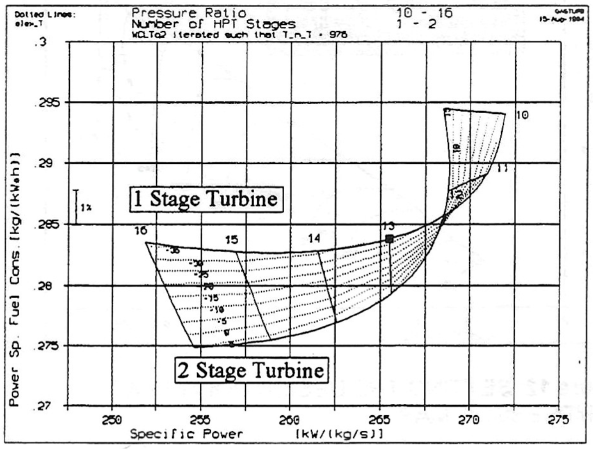

Figure 10: SPECIFIC POWER AND FUEL CONSUMPTION FOR SINGLE AND TWO STAGE GAS GENERATOR TURBINES

constant exit swirl. An optimum design for a turboshaft with a two-stage gas generator turbine would need a pressure ratio of around 15. The advantage in terms of fuel consumption is around 3% compared to a single stage design.

2.3 Summary and Concluding Remarks

A simple conventional cycle study does not yield a realistic result. Only when the constraints imposed by the component design requirements are allowed for one gets answers for compressor pressure ratio and burner exit temperature that are in line with reality.

The example shown was only one of many alternatives. The computer makes it easy, to repeat the exercise for example for a different level of component efficiencies. One can also look into the details of the power turbine design. How the inclusion of a heat exchanger into the cycle affects the results can also be studied.

The results of the GASTURB cycle calculations are lining up perfectly with those of the big performance programs used in industry. Component design calculations, however, are fairly basic. The details should not be taken as absolute values. They should be used only for relative comparisons.

As demonstrated by the figures there are many things to be checked while selecting an optimum cycle. For getting a quick overview it is essential, to present the results graphically. Only then the huge amount of data produced by a computer can be judged. One should always remember: the computer does not think - that is left to the user.

3. OFF-DESIGN CALCULATIONS

After having selected a cycle one can calculate for off-design conditions. For those real component maps are used that are stored numerically. The format of the data uses auxiliary coordinates called ß-lines as shown in Figure 11 for the example of a compressor map. Surge line data are stored separately.

3.1 Correlation of the Design Point with the Maps

The engine design cycle needs to be correlated with the component maps. Let us have a more detailed look at the problem. In the design point calculation there are certain efficiencies used for the components. The component maps need to be scaled in such a way that the design point is in line with a specific point in the map.

One can set the design point in the map easily by using the mouse, see Figure 12. Note, however, the consequence of moving the design point around in the map: The peak efficiency of the scaled map will change. If the design point is moved to a map region with low efficiency then the peak efficiency of the scaled map will increase. Normally one cannot have a very high compressor surge margin and good efficiency simultaneously.

3.2 Component Matching Procedure

The calculation of each off-design point requires an iteration. Several input variables for the thermodynamic cycle must be estimated. The result of each pass through the cycle calculations is

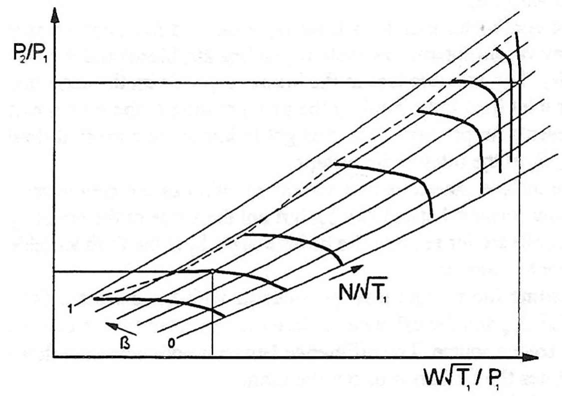

Figure 11: COMPRESSOR MAP WITH ß-LINES

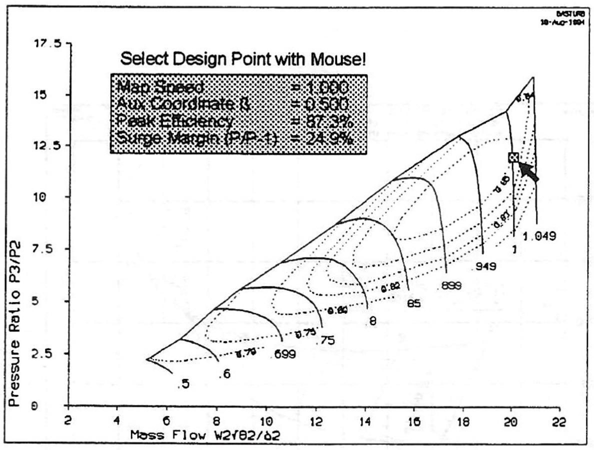

Figure 12: SETTING THE DESIGN POINT IN A COMPRESSOR MAP

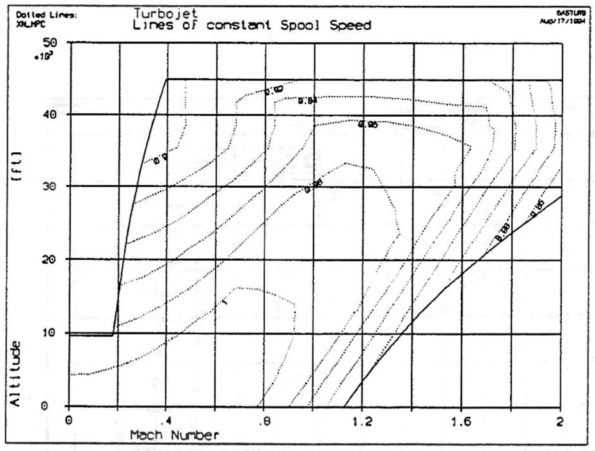

Figure 13: LINES OF CONSTANT SPOOL SPEED IN THE ALTITUDE - MACH NUMBER REGIME

a set of "errors". Inconsistencies are introduced through the use of imperfect estimates for the variables. The number of errors equals the number of variables.

The algorithm used to manipulate the variables in such a way that in the end all errors will be virtually nonexistent is a Newton-Raphson-Iteration.

The progress of the iteration can be observed on the screen. In one of the turbomachinery maps the operating point is moving around while the iteration algorithm is trying to find the solution. In a separate window one can observe the absolute values of all iteration errors as a bar chart.

Let us go through an off-design calculation for a straight turbojet with a prescribed rotational speed. We will see, which are the variables and the errors mentioned above. The calculation starts with the inlet. This provides the compressor entry conditions. The compressor map has to be read next. Although we know aerodynamic speed $N/\sqrt{\Theta}$, this is not sufficient to place the operating point in the map. An estimate for the auxiliary coordinate $ß_C$ is required. $ß_C$ is our first variable.

We can read the mass flow, the pressure ratio as well as the efficiency from the compressor map and calculate burner inlet conditions. The amount of fuel required to run the engine at the requested rotational speed is still unknown. An estimate for the burner exit temperature T_4 will yield the fuel-air-ratio. T_4 is the second variable.

The combustor exit mass flow W_4 is derived from compressor exit flow W_3, the internal air system (cooling air, bleed) and the fuel flow W_F. The pressure loss in the burner depends on the corrected burner inlet flow only. Finding the total pressure at the burner exit P_4 presents no problem. We thus get to know the corrected flow $W_4\sqrt{T_4}/P_4$ at the inlet to the turbine.

The turbine places us in a similar situation as the compressor: We know corrected speed $N/\sqrt{T_4}$, but not the value of the auxiliary map coordinate for reading the turbine map. $ß_T$ is the third variable we have to estimate.

Reading the turbine map provides us with the corrected flow $(W_4\sqrt{T_4}/P_4)_{map}$ and the efficiency. Here the corrected flow is derived from a second source. The difference between both corrected flows constitutes the first error of our iteration.

We will ignore this error for the moment and go on. The turbine exit conditions and the shaft power delivered are calculated by means of the values read from the map. A comparison of the power required to drive the compressor and the power delivered by the turbine (it is obvious that they must be equal to ensure steady state operation) reveals the second error of the iteration.

The pressure loss $(P_5-P_6)/P_5$ of the turbine exit duct only depends on the corrected flow $W_5\sqrt{T_5}/P_5$. When the afterburner is switched off then P_8 equals P_6. Furthermore W_8 equals W_6 and T_8 equals T_6. The nozzle inlet conditions thus are fully fixed.

A certain total pressure $P_{8,req}$ is required to force the flow W_8 with the total temperature T_8 through the given area A_8 while the back pressure is equal to P_{amb}. So long as the iteration has not yet converged there will be a difference between P_8 and $P_{8,\,req}$. This constitutes the third error of the iteration.

We have found three variables and three errors. The Newton-Raphson-Iteration algorithm modifies the variables in such a way that all errors equal zero.

The straight turbojet is a very simple engine. In case of a mixed flow turbofan there are two or three compressors on two spools to be matched with the turbines and the nozzle. That leads to an iteration with up to 7 variables. This, however, is not a problem for the program. It just needs a little longer to find the result.

As result for an off-design calculation one gets a lot of data. Just one example is shown in Figure 13: For that the performance of a straight turbojet was calculated throughout the flight envelope. The following limiters have been switched on: corrected speed, turbine inlet temperature and compressor exit temperature. The graph shows lines of constant mechanical speed in the altitude - Mach number regime.

4. THE PARALLEL COMPRESSOR MODEL

The flow into the first compressor of an aeroengine is often nonuniform. If the aircraft flies with a high angle of attack and/or sideslip then the total pressure will vary both circumferentially and radially. The effects of these flow disturbances can be studied with the "Parallel Compressor Model".

4.1 Setup of the Model

Let us assume for sake of simplicity that we have two zones of equal size with different total pressure at the inlet of the compressor. At the exit of the compressor there shall be a big volume with constant static pressure. We split now the machine in two

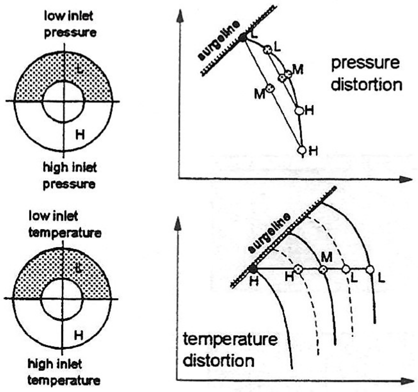

Figure 14: PARALLELL COMPRESSOR MODEL FOR TEMPERATURE AND PRESSURE DISTORTION SIMULATION

compressors of identical size. One of the compressors sucks the flow with the low inlet pressure. The other one gets the high inlet pressure flow. Both machines are assumed to have exactly the same characteristics.

As can be seen from Figure 14 the compressors are operating in different points of the map. Both points must be on the same speedline. The compressor with the lower inlet pressure, however, needs to produce a higher pressure ratio. When the difference in inlet pressure increases, then the distance between both operating points increases also. As soon as the point marked with L lies on the surge line there is according to the basic parallel compressor theory the stability boundary of the total compressor achieved. The compressor is predicted to surge in spite of the fact, that the mean operating point M is still far from the surge line.

In its simple form the parallel compressor model gives the right tendencies but does not agree very well with reality. The model can be improved by fairly simple corrections [3,4].

4.2 Pressure Distortion and Temperature Distortion

At the exit of the compressor the static pressure is circumferentially constant as mentioned above. Since the pressure ratio, however, is different, the total temperature at the exit of the compressor will not be uniform. If there follows a further compressor downstream of the first one, then the operating points in the map of the second compressor will no longer be on the same speedline. This has a more detrimental effect to the stability than a pure pressure distortion. The main differences between pressure and temperature distortion can be read from Figure 14.

The parallel compressor model as described above is integrated into the GASTURB program. One can select either pressure or temperature distortion or a combination of both for a given sector angle. The iteration required to match the cycle uses up to 11 variables, depending on the type of engine. A typical result is shown in Figure 15.

5. TRANSIENT SIMULATIONS

A complete gas turbine performance model allows also the calculation of the transient behavior. For such a model two things are required. First one needs to expand the thermodynamic description of the gas turbine. Secondly one must have some sort of control system to drive the model as required.

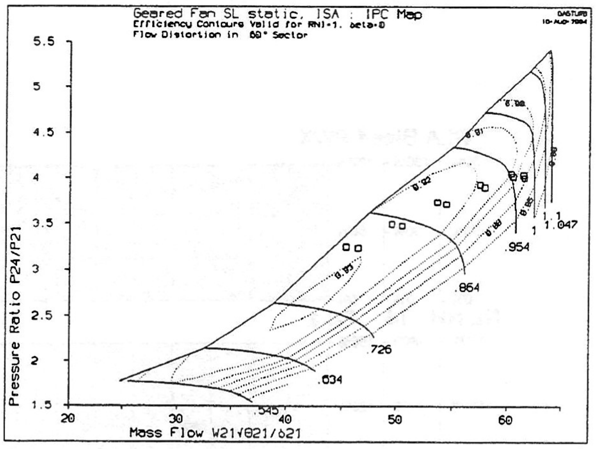

Figure 15: OPERATING POINTS FOR THE DISTORTED SECTOR AND AVERAGE OPERATING CONDITIONS IN A BOOSTER MAP OF A TURBOFAN

5.1 Additions to the Steady State Model

The most important additional item in a transient engine model is the modification of the power balance between compressors and turbines by a term that takes into account the polar moment of inertia of the spool. During accelerations, for example, there is more turbine power needed compared to a steady state operating point.

Other phenomena happening during transients have normally only a limited influence on the results. Therefore they are not described within GASTURB. The main reason for that is, that a significant amount of additional input data would be needed. The program would no longer be easy to use. The detailed thermodynamic modeling including volume dynamics, heat transfer, variable tip clearance etc. is left to bigger programs.

5.2 The Control System

A turbine engine is operated using a "Power Lever" mounted in the cockpit of the aircraft. By changing the setting of the lever the power of the engine is modulated. The position of the lever is normally described by the "Power Lever Angle" (PLA).

Actually the pilot of an aircraft wants to control the thrust of his engine directly. Because thrust is not measurable on the installed engine there is some other parameter used for setting the engine power. In GASTURB the power lever is directly connected to the mechanical spool speed. In case of a turbofan, for example, there is a linear relationship between PLA and fan speed.

In addition to the limiters used for steady state operating point definition like maximum spool speeds, temperatures and pressures there are further limiters for the transient operation required. These are acceleration and deceleration rates, minimum and maximum fuel-air-ratios in the burner and/or limitations for corrected fuel flow W_F/P_3 as a function of corrected spool speed.

In a real (fixed geometry) engine the only way to influence the operating point is to modulate fuel flow. Sensors deliver signals to the control system which compares the power delivered with the power demanded. According to the differences between actual and demanded value the fuel flow is either increased or decreased. The very simple control system included in GASTURB is of the proportional-integral-differential type. By setting the constants accordingly one can get the desired behavior.

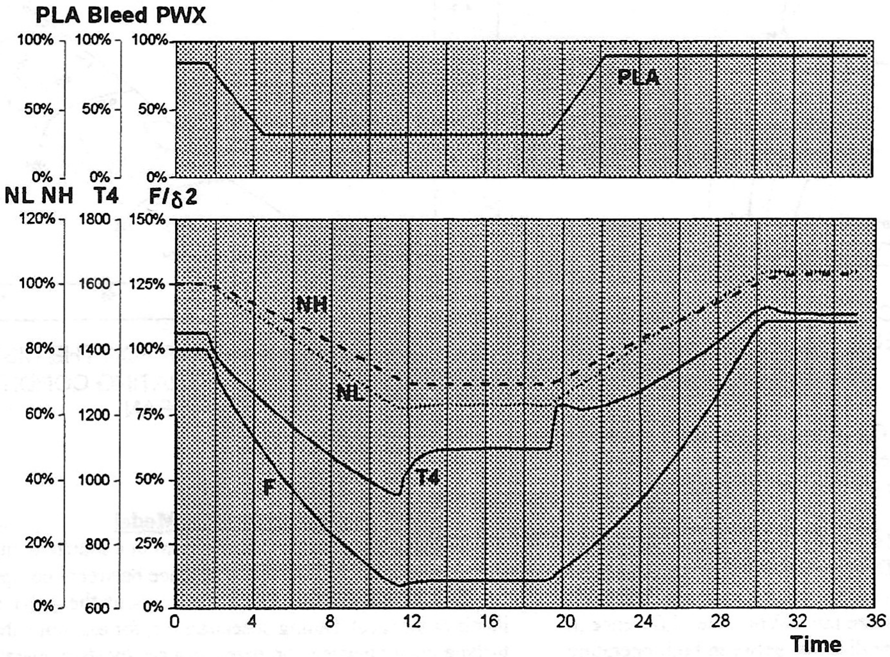

Figure 16: TRANSIENT OUTPUT SCREEN WITH MAIN PARAMETERS

A transient calculation begins after the input of PLA as a function of time. The control system then dictates the fuel flow and accelerates or decelerates the engine. One can observe on the screen the behavior of some important parameters like burner exit temperature, thrust and spool speeds, see Figure 16, for example.

After the calculation is finished one gets a variety of graphs showing all the results as a function of time or in any other combination. The operating lines in the component maps are available from a click with the mouse. It is very easy to get an insight into the transient behavior of gas turbine engines.

6. SUMMARY

The program GASTURB makes it easy to study the thermodynamics of gas turbines. It covers all important aspects of engine design cycle selection, off-design behavior, engine stability and transient operation. The number of input data required is limited to the important items. Real compressor and turbine maps are used that yield representative results.

For off-design calculations there are iterative methods implemented with up to 11 variables. The user, however, is not confronted with the difficult task of setting up the iteration. He gets an insight into the problem, but he need not bother with the details of the mathematics and he can concentrate on the physics of gas turbines.

The program simulates the most common types of aero-gas turbines. Among them are single and two spool turboshafts, turboprops and turbofans. Both mixed flow and unmixed flow engines are covered. Afterburners and convergent-divergent nozzles are available for mixed flow turbofans and turbojets. A heat exchanger can be used on the turboshaft.

Nowadays many students possess a PC or have access to one. In addition to looking at the complex formulae found in textbooks they can have a look at actual data. These are presented in graphics that make the interpretation of the results as easy as possible. However, the program is not limited to academic exercises. Actually, initially it has been developed for use in the gas turbine industry.

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