

Very Small Gas Turbine Jet Engines – Current Limits and Potential for Improvement

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Abstract

Gas turbine technology offers propulsion devices with excellent thrust-to-weight ratios for a broad range of flight conditions. However, the low efficiency and the high operating costs of current small gas turbine engines substantially restrict their use in new emerging markets. This paper evaluates the performance of small gas turbines based on thermodynamic cycle data obtained from a test stand. It analyses the effect of component-based parameters and proposes measures to increase efficiency. After having carried out the analysis, the intersection between compressor diffuser and combustion chamber was identified as an area with high potential for improvement.

1. Introduction

In recent years a variety of new gas turbine jet engines in the thrust range of 1000 N and below (very small gas turbine jet engines) have been designed. Their small size and light weight make them attractive for new emerging markets like model aircrafts, UAVs, remotely piloted vehicles and autonomous flight systems. However, due to scaling effects and the lack of design guidance for key components such as the combustion chamber or the compressor diffuser vanes, the low efficiency of these engines still restricts their use in many instances.

At the Institute for Flight Propulsion of the Technische Universität München investigations on very small gas turbines have been conducted in order to evaluate the current state-of-the-art technology as well as to assess potential for improvement. The thermodynamic cycle process of a gas turbine Frank Turbine TJ 74 [1] was analyzed on a test stand (see Figure 1). However, some representative state conditions could not be determined accurately due to circumferentially uneven distributions. Numerical analyses were performed in order to account for measurement uncertainties and provide more detailed information on flow conditions. Design Parameters and component efficiencies were iteratively calculated using the GasTurb 12 performance software [2].

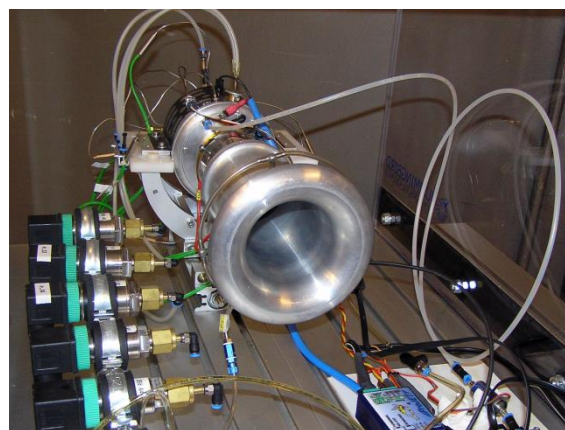


Figure 1: Test stand with gas turbine Frank Turbine TJ 74

2. Configuration of small gas turbines

The configuration of most of the small gas turbine jet engines is based on the guidelines of Kurt Schreckling [3]. Ambient air enters the gas turbine through the engine intake and proceeds to the compressor. The compressor consists of a centrifugal impeller and subsequent diffuser vanes, which deflect the air in axial direction and reduce its velocity by increasing the static pressure. The air enters the combustion zone through several holes in the inner and outer liner of the combustor. A small part of air is guided to the vaporizer sticks at the rear side of the combustor. Inside the sticks this part of air is mixed with fuel, which vaporizes on the hot wall of the sticks. The fuel-air-mixture ignites after leaving the vaporizer sticks. Downstream of the combustion chamber the exhaust gas expands in the turbine, which supplies power for the compressor impeller. The exhaust gas exits the gas turbine via the convergent nozzle, where the flow is accelerated in order to create thrust. Figure 2 shows the entire configuration of the gas turbine.

A major difference compared to larger gas turbine engines is the absence of a separate oil system for cooling and lubricating the bearings. Instead, a fuel mixture of kerosene and about 5% turbine oil supplies both the combustion and the lubrication. The mixture is split after the fuel pump so that about 5 % of the total fuel flow is channeled through the bearings [3]. This fraction unites with the main exhaust gas flow not before the turbine section – it bypasses the combustion chamber and therefore remains unburned.

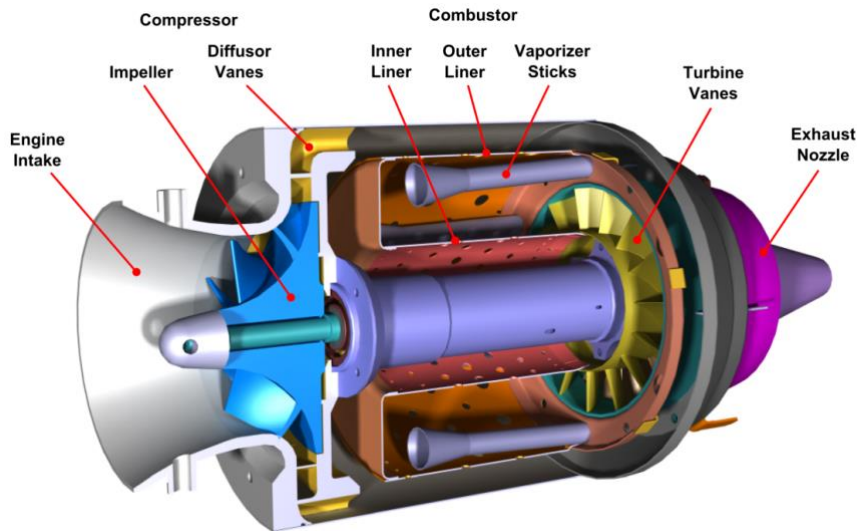


Figure 2: Configuration of a very small gas turbine

3. Performance evaluation of single-spool turbojets

In order to evaluate the performance of a jet engine, the thrust specific fuel consumption SFC as well as the specific thrust F_{sp} can be quantified. The SFC relates the fuel mass flow to the thrust output, therefore characterizing the overall efficiency of the engine (see Eq. (1)). The specific thrust is the quotient of thrust and engine mass flow making it primarily a function of the exhaust gas velocity v_9 and thus the specific kinetic energy of the exhaust (see Eq. (2)). This value is explicitly important for small engines in which size is a limiting factor and where thrust has to be created by low mass flows and high velocities.

$$SFC = \frac{\dot{m}_{fuel}}{F_N} \approx \frac{FAR}{(FAR + 1) \cdot v_9} \quad (1)$$

$$F_{sp} = \frac{F_N}{\dot{m}_0} \approx (FAR + 1) \cdot v_9 \quad (2)$$

Both equations are presented without showing the influence of flight velocity. The fuel-to-air ratio $FAR = \dot{m}_{fuel}/\dot{m}_0$ relates the fuel mass flow to the engine air mass flow $\dot{m}_0$. As Eq. (1) and (2) show, both performance values are closely correlated to each other. However, the conversion from kinetic energy into thrust becomes generally less efficient with higher velocities [4]. Therefore, engine design always has to be a compromise, because a low SFC and a high F_{sp} cannot be achieved at the same time.

The performance values are a function of design parameters and component efficiencies of the engine. For a single-spool turbojet, the SFC is a result of the compressor total pressure ratio $\Pi_{t,C}$, the turbine inlet temperature $T_{t,4}$ as well as the efficiencies and pressure losses of the individual components (see Eq. (3)). $\eta_{is,C}$ and $\eta_{is,T}$ are the isentropic efficiencies of compressor and turbine, respectively. η_{Comb} evaluates the efficiency of the combustion defining the portion of injected fuel that is actually combusted. η_{mech} is the mechanical efficiency resulting from bearing losses. Pressure losses occurring in the inlet section, the combustion chamber and the thrust nozzle are taken into account with their respective total pressure ratios $\Pi_{t,I}$, $\Pi_{t,CC}$ and $\Pi_{t,N}$.

$$SFC = f(\Pi_{t,C}, T_{t,4}, \eta_{is,C}, \eta_{is,T}, \eta_{Comb}, \eta_{mech}, \Pi_{t,I}, \Pi_{t,CC}, \Pi_{t,N}) \quad (3)$$

The correlations shown above are also valid for the specific thrust F_{sp} . However, the influence of the combustion efficiency η_{Comb} is very low as long as the turbine inlet temperature $T_{t,4}$ remains constant. In this case, a less efficient combustion would require a higher fuel-to-air ratio.

4. Cycle analysis of a small jet engine

4.1 Measurements

To evaluate the performance of a small gas turbine jet engine a Frank Turbine TJ 74 was under investigation on a test stand. Thermodynamic cycle parameters were measured at a high load reference operating point at 107 000 rpm. The measurements were taken at stationary conditions. All values were time-averaged over a period of 20 seconds. Following data was obtained (see Table 1):

Table 1: Measured Cycle Data for the TJ 74 jet engine

Static Ambient Pressure	p_0	96 kPa
Static Ambient Temperature	T_0	280 K
Net Thrust	F_N	106.4 N
Engine Air Mass Flow	$\dot{m}_0$	300.7 g/s
Total Fuel Flow	$\dot{m}_{fuel}$	5.31 g/s
Total Compressor Outlet Pressure	$p_{t,3}$	273 kPa
Static Compressor Outlet Pressure	p_3	251 kPa
Total Compressor Outlet Temperature	$T_{t,3}$	407 K
Total Combustion Chamber Outlet Pressure	$p_{t,4}$	236 kPa
Total Turbine Outlet Temperatures	$T_{t,5a}$	873 K
	$T_{t,5b}$	899 K
	$T_{t,5c}$	869 K
Total Averaged Turbine Outlet Temperature	$\overline{T}_{t,5}$	880 K

On the test stand a standardized bell-mouth air inlet duct was fitted to the engine. Air mass flow $\dot{m}_0$ was calculated from the ambient density, the inlet cross section area and a corresponding measured difference in static pressure $\Delta p = p_{inlet} - p_0$. The average temperature $\overline{T}_{t,5}$ at the turbine outlet is determined by arithmetic mean of the temperatures measured by three separate probes (a, b, c) which were circumferentially distributed within the same cross section.

To analyze the entire thermodynamic cycle of the engine, pressure losses of inlet and nozzle were estimated to one percent each. The mechanical efficiency was assumed to be 98 percent. All design parameters were iterated to match the data obtained from the test stand. The results are shown in Table 2, Column 1 at the end of this chapter. The cycle analysis yielded a specific fuel consumption SFC of 49.9 g/(kNs) and a specific thrust F_{sp} of 354 m/s.

Compared to larger engines the thermodynamic efficiency of small gas turbines is very low. The measured *SFC* is higher than the value achieved by the modern military used EJ200 engine with an activated afterburner of 48 g/(kNs) [5]. The main factor contributing to this inefficiency can be found in the combination of the low pressure ratio of 2.87 (26 for EJ200) and the turbine inlet temperature of 995 K (about 1800 K for EJ200). Both values are bound to certain constraints and therefore remain on a fairly low level. The turbine inlet temperature is limited by the sustainable material temperature of the turbine, which determines the overall lifetime of the engine. As turbines of small jet engines are generally not cooled, the turbine inlet temperature is on a far lower level compared to larger engines with a secondary air system. Additionally, as will be discussed in the following section, temperature distribution is not even over the turbine inlet cross section. As hot spots define the local maximum temperature and thus material strain, the average temperature over the cross section remains lower. For some applications the pressure ratio may be limited by the geometric size of the engine, as higher pressure ratios require larger compressors. Moreover, higher compressor outlet temperatures resulting from increased pressure ratios would require other materials like aluminum, such as steel or titanium. This would increase the engine mass as well as the manufacturing complexity and finally the overall costs of the engine. Higher pressure ratios would also require more specific power provided by the turbine. This could lead to the necessity of a second axial turbine stage which would further enhance complexity.

Several small size effects lead to higher losses, which have a further impact on performance. This is particularly relevant for turbo components. Low Reynolds numbers lead to high friction factors due to a low ratio of inertial to viscous forces. The surface-to-area ratio, which is inversely proportional to the geometrical size, increases the friction even more at small dimensions. Moreover, the influence of clearance gaps becomes more significant as they result from manufacturing tolerances and therefore do not scale with size. These factors lead to comparably low efficiencies for compressor (74.6 %) and turbine (78.5 %). Detailed analyses of small size effects and their impact on engine performance are presented in [6] and [7].

The combustion constitutes another major influence on the performance of small gas turbine. While combustion efficiencies of larger engines normally range above 99 percent in design conditions, the analysis shows a significantly lower efficiency of 85.7 percent. Even if the fuel flow through the bearings (approx. 5% of the total fuel flow) is subtracted, almost ten percent of the fuel in the combustion chamber still remains unburned.

4.2 Assessment of measured values

The measured data is sufficient to reconstruct the entire thermodynamic cycle and thus all design parameters and efficiencies. However, the reliability of the measurement has to be assessed. As the thrust and the fuel mass flow are determined via a force sensor respectively via a Coriolis flow sensor, both values can be considered reliable. This means that the specific fuel consumption can be calculated accurately. However, problems occur when measuring state values at specific cross sections between the components of the engine. Particularly total values have to be handled with care as they depend on local flow phenomena, which are often circumferentially asymmetric. Therefore a more detailed investigation on these measured values has to be performed.

Compressor outlet pressure

The measurement of the total as well as the static pressure at the compressor outlet (station 3) yields a Mach number Ma_3 of 0.35 and a corresponding velocity v_3 of 141 m/s. According to recent numerical investigations on the compressor diffuser vanes [8] these number are too high. As Figure 3 shows, the velocity of the flow leaving the diffuser varies significantly along the cross section between two vanes. This is a result of the angular momentum of the flow coming from the impeller, which causes a separation of the flow from the stator vanes. This leads to local reverse flow zones as well as to velocity peaks in the magnitude of about 240 m/s. The averaged absolute flow velocity $v_{3,avg}$ in this area is 102 m/s. This leads to the conclusion that the probe measuring $p_{t,3}$ was located at a position where the local velocity is higher than the average. With the assumption that the static pressure p_3 remains constant within the entire cross section and was therefore measured accurately, the compressor outlet total pressure $p_{t,3}$ can be corrected to the cross section averaged value 262 kPa. The change of the design parameters is shown in Table 2, Column 2.

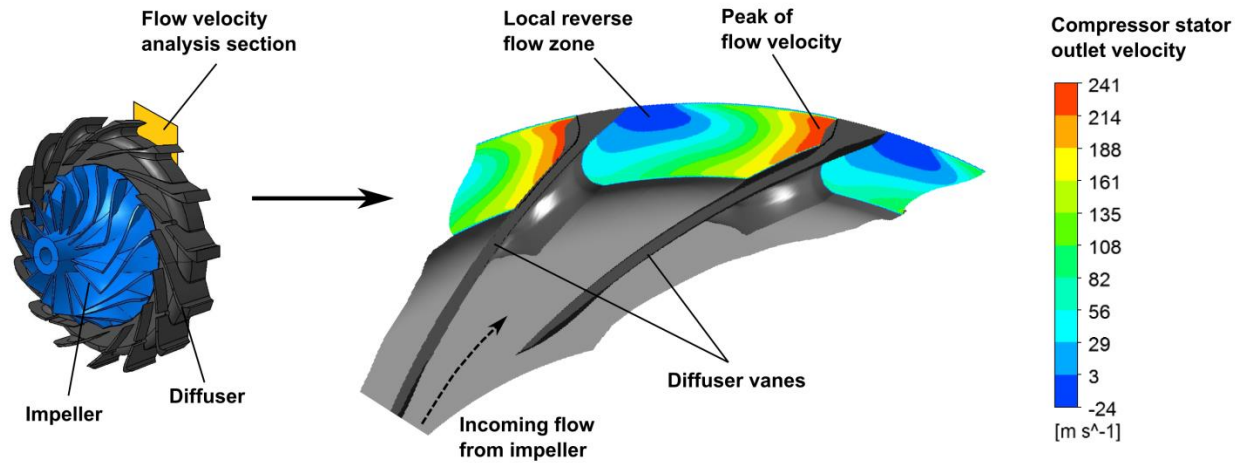


Figure 3: Velocity profile at compressor diffuser outlet (station 3)

The lower outlet pressure directly leads to a reduced compressor pressure ratio of 2.76. With the measured total combustion chamber outlet pressure the combustion chamber pressure loss is significantly reduced from 13.6 to 10.1 percent. This value appears to be more realistic as it comes closer to values obtained from numerical investigations of the combustion chamber, which estimate the pressure loss between 8 to 9 percent [9], respectively 10 percent [10].

A noticeable effect is the influence on the turbine efficiency, which decreases about 2.5 basis points. With the ambient pressure, the assumed nozzle pressure loss and the measured values for mass flow, thrust and turbine outlet temperature the position of point 5_t in the temperature-entropy diagram (see Figure 5 at the end of this chapter) is fixed. Due to the reduced specific power required for compression, the total turbine inlet temperature has to decrease. As Table 2 shows the temperature drop is 4 K. With the measured total turbine inlet pressure $p_{t,4}$ remaining constant in this analysis, the position of point 4_t shifts to $4'_t$ to lower entropy. This leads to an increase in entropy change over the turbine from $4'_t$ to 5_t which reduces the turbine efficiency. However, the measurement of $p_{t,4}$ also constitutes an uncertainty which effects the assessment of the turbine efficiency. An evaluation of this effect has already been carried out in [11].

Turbine Temperature Distribution

The total turbine outlet temperature was averaged from three measurements taken from probes circumferentially distributed within this station. Kügler et al. [9] showed that severe hot and cold spots occur at the combustion chamber outlet (see Figure 4). This is a consequence of the combustion chamber design featuring vaporizer sticks. The hot spot at the outer casing is a side-effect of the turbulence modeling in the combustion chamber flow. The model results in very high reaction rates in the boundary layer, which do not occur in reality. This effect, therefore, is not further regarded in this analysis. The combustion primarily takes place in the spaces between two sticks where vortices emerge supporting mixing and reaction. Although these hot spots level out while proceeding downstream, they are clearly observable at the turbine inlet. However, as the flow entering the combustion chamber through the outer liner still has an angular momentum [12], the hot and cold spots shift circumferentially and do no longer correspond with the position of the vaporizer sticks. Measurements of turbine inlet temperature in former publications using only one single probe were found to be either too low (only qualitatively discussed [13]) or too high (up to 1300 K [14]) by their authors. Hot and cold spots coming from the combustion appear to be a sound explanation for these results. However, too low temperatures can also result to a minor degree from combustion partially continuing through the turbine and the exhaust nozzle.

Measurements carried out by Weber [15] confirm that severe hot spots originate from the combustion. Measurements were taken at four different circumferential positions directly at the combustion chamber outlet yielding temperatures ranging from 746 to 1156 K. These values comply very well with the numerical investigation in [9], which estimates hot and cold spots within the same range. Weber also took four measurements of the temperature at the turbine outlet. Here, the circumferential differences reduced significantly compared to the combustion chamber outlet as all measurements were within a spread of about 80 K. The spread among the measured turbine outlet temperatures in the present analysis is with 30 K significantly lower (see Table 1). However, the probes were not specifically placed so that hot and cold spots may not have been explicitly detected.

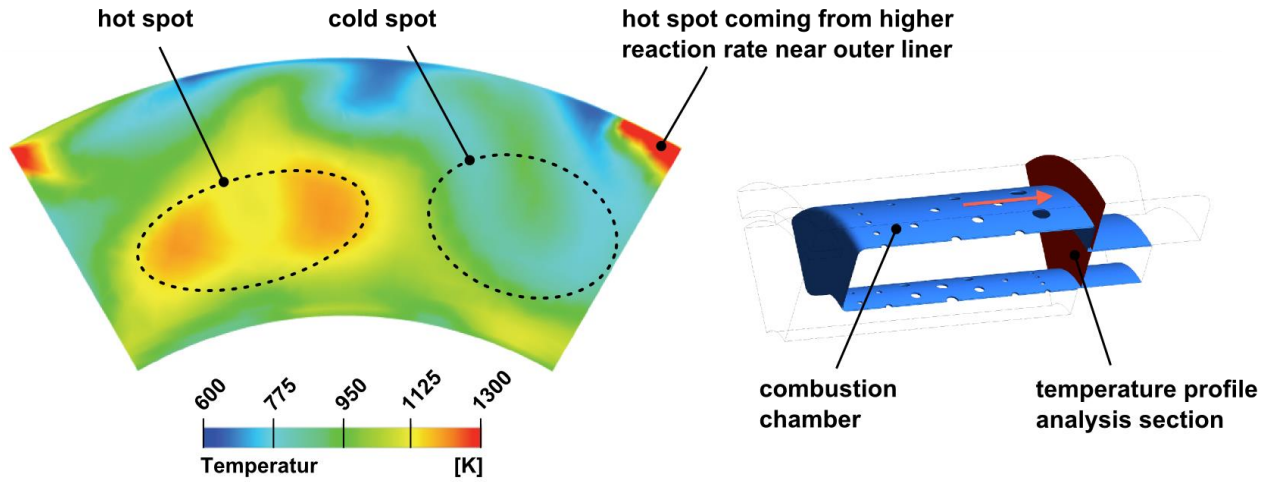


Figure 4: Temperature distribution at turbine inlet

The circumferential temperature profile illustrates that a cross section averaged temperature, which would be suitable to represent the thermodynamic cycle, cannot accurately be determined by the measurements. The numerical analysis of the combustion yields a cross section averaged total turbine inlet temperature of 971 K. The cycle calculation can be adapted to match this value. The results for the design parameters and efficiencies are shown in Table 2, Column 3. The new positions of the thermodynamic states in the temperature-entropy diagram are marked with a double apostrophe (see Figure 5). The correction for the compressor outlet velocity is still applied. As the turbine inlet temperature is reduced while the pressure remains constant, $4_t''$ moves to lower entropy. The turbine still has to provide the same power for the compressor, therefore the turbine outlet temperature decreases to 860 K. In order to maintain the thrust at a constant level, the turbine efficiency increases from 76.0 to 78.3 percent. The lower turbine inlet temperature at a constant fuel mass flow yields a decreased combustion efficiency of 82.5 percent. However, this value corresponds well with Hupfer et al. [11], who identified combustion efficiencies between 82.9 and 83.7 percent depending on the engine rating. When the bearing fuel mass flow is subtracted, the combustion efficiency in the burner alone is 86.8 percent.

Table 2: Calculated design parameters and component efficiencies

		(1)	(2)	(3)
		Directly calculated from measurement	Corrected with diffuser outlet velocity profile	Corrected with diffuser outlet velocity and turbine temperature profile
Compressor Pressure Ratio	$\Pi_{t,C}$	2.87	2.76	2.76
Turbine Inlet Temperature	$T_{t,4}$	995 K	991 K	971 K
Isentropic Compressor Efficiency	$\eta_{is,C}$	0.746	0.739	0.739
Isentropic Turbine Efficiency	$\eta_{is,T}$	0.785	0.760	0.783
Combustion Efficiency	η_{Comb}	0.857	0.857	0.825
Combustion Chamber Pressure Ratio	$\Pi_{t,CC}$	0.864	0.899	0.899

These analyses illustrate the problem that occurs when measuring data from a very small gas turbine. The complex flow conditions in the compressor diffuser as well as the combustion chamber design with vaporizer sticks cause high local gradients of state values such as velocities and temperatures. Moreover, as flow paths in small gas turbines are narrow, a sufficient high resolution of measuring probes cannot be realized. The result is an inaccurate experimental evaluation of loss sources. Potential for the improvement of components cannot be precisely assessed. This is especially true for components in the hot gas section, as thermal conditions in the turbine inlet cross section cannot be quantified accurately. Assumptions and values for combustion efficiency, combustion chamber pressure loss and turbine efficiency are therefore flawed with uncertainties.

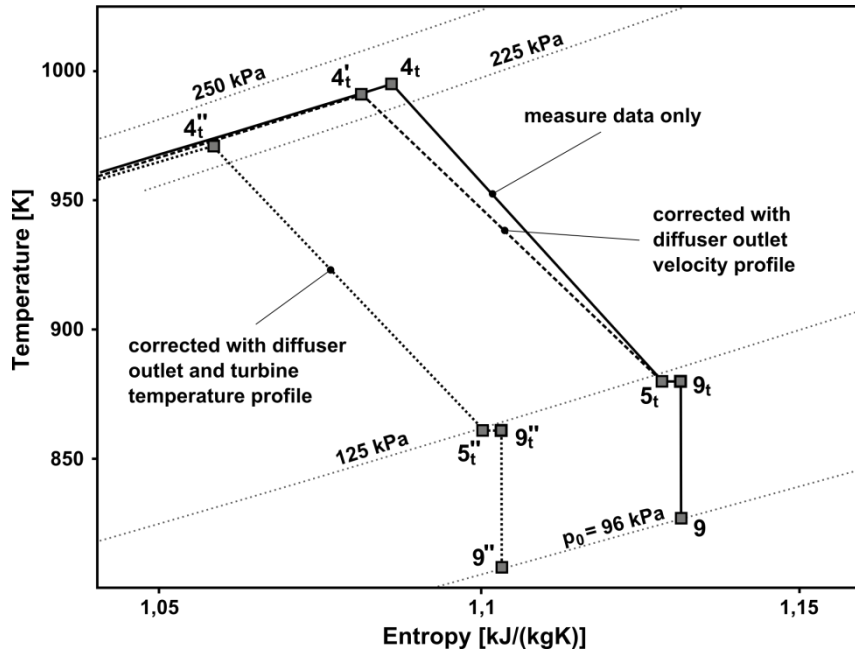


Figure 5: Temperature-entropy diagram for expansion in turbine

Current investigations therefore rely on measurements corrected with numerical analyses in order to account for these effects. For further studies within this paper the parameters calculated with the corrections for compressor outlet velocity and turbine inlet temperature are used. However, these analyses still need validation. At the Institute for Flight Propulsion, current research aims to accomplish this task. A new adjustable measurement installation is under development. The measuring position can be shifted circumferentially to record the total compressor outlet pressure across an entire segment between two stator vanes. Additionally, a three-hole pressure probe is used that can be turned in order to find the local flow direction. This installation promises a better experimental evaluation of the compressor outlet conditions. For the investigation of the turbine inlet conditions a separate combustion chamber test stand is in development, which allows a detailed measurement of temperatures, pressures and velocities over the entire cross section [16].

5. Evaluation of potential for improvement

The efficiency of a real Brayton cycle with component losses is primarily determined by the combination of pressure ratio and turbine inlet temperature. The turbine inlet temperature is limited due to the maximal allowable material temperature of the engine components. This affects stator and rotor of the turbine as well as the rearward bearing. It has already been shown that the turbine stator is exposed to a circumferential temperature profile with hot spots of about 1150 K while the cross section averaged temperature is 971 K. If a more equal distribution could be achieved, the averaged turbine inlet temperature could be increased without affecting the operational reliability of the engine in a negative way. However, as Figure 6 illustrates, a sole increase of the turbine inlet temperature does not yield better specific fuel consumption. The optimal temperature of 1003 K would reduce the *SFC* by only 0.15 percent (Point HT in Figure 6). Higher temperatures would again increase the *SFC*. This can be explained by the increase of specific thrust due to a higher enthalpy level in the exhaust. High exhaust gas velocities lead to higher specific fuel consumptions as power conversion becomes less efficient.

The driving factor for a better overall efficiency is the compressor pressure ratio. Within the range of reasonable values for small gas turbines, a higher pressure ratio always leads to reduced specific fuel consumption. However, as mentioned before, pressure ratio is limited due to constraints applicable for small gas turbines such as geometry, material temperature and single stage turbine configuration. Market research has shown that none of the existing engines in the thrust range of 1000 N and below operates at a pressure ratio larger than 4. Higher pressure ratios would furthermore lead to additional shock losses, as transonically optimized blades cause manufacturing problems at small sizes [7]. Still, a pressure ratio of 4 could reduce the *SFC* of the analyzed engine by 14.2 percent (Point HP in Figure 6).

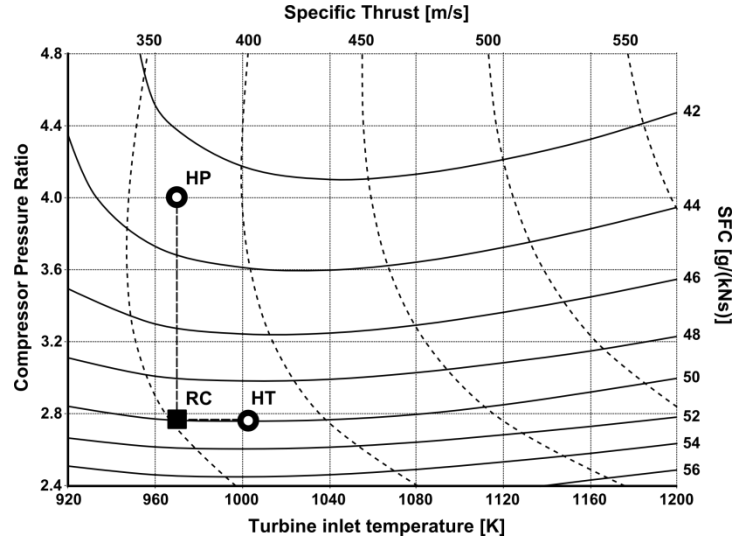


Figure 6: Performance over Design Parameters
(RC – reference cycle, HT – high temperature, HP – high pressure ratio)

5.1 Identification of loss factors

A promising method to increase pressure ratio is to reduce losses occurring within the compressor diffuser vanes. This would lead to an improved efficiency without requiring higher compressor outlet temperatures or higher work transfer from the turbine. According to the numerical investigation the pressure loss of the diffuser is in the range of 12 percent.

Figure 7 shows the compression process in the temperature-entropy diagram. While point 2_t and 3_t represent the total conditions at inlet and outlet of the entire compressor section, point 21_t shows the conditions between impeller and diffuser. The analysis shows that the efficiency of the impeller is at 84.9 percent at a pressure ratio of 3.14.

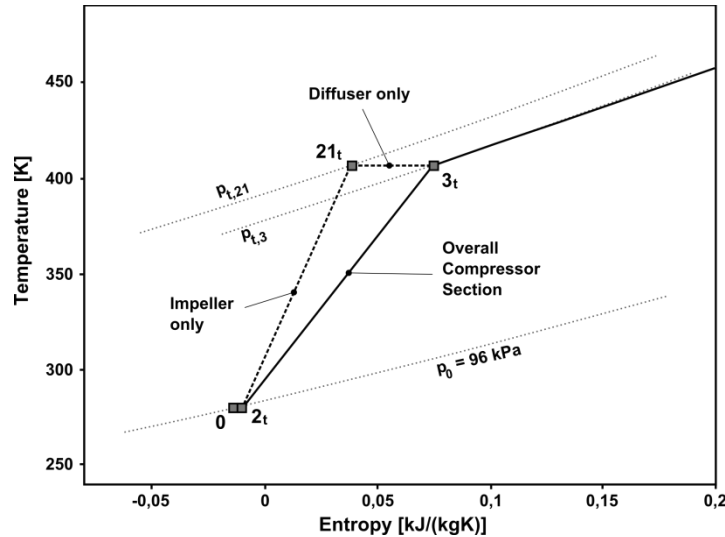


Figure 7: Temperature-entropy diagram for compression

Figure 8 shows the impact of compressor diffuser pressure loss on specific fuel consumption and specific thrust. It is clearly visible that the diffuser losses are responsible for a large part of fuel consumption. An ideal diffuser would decrease the *SFC* by 17.8 percent. Current diffusers are designed primarily with respect to simple manufacturing and low costs. They do not incorporate aerodynamically optimized vane design. Investigations of the flow through the diffuser vanes indicate that there are severe flow separations. Improved stator vane geometry can help to reduce

losses and thus increase overall efficiency. Numerical analyses on different geometries such as tandem stators are currently being carried out.

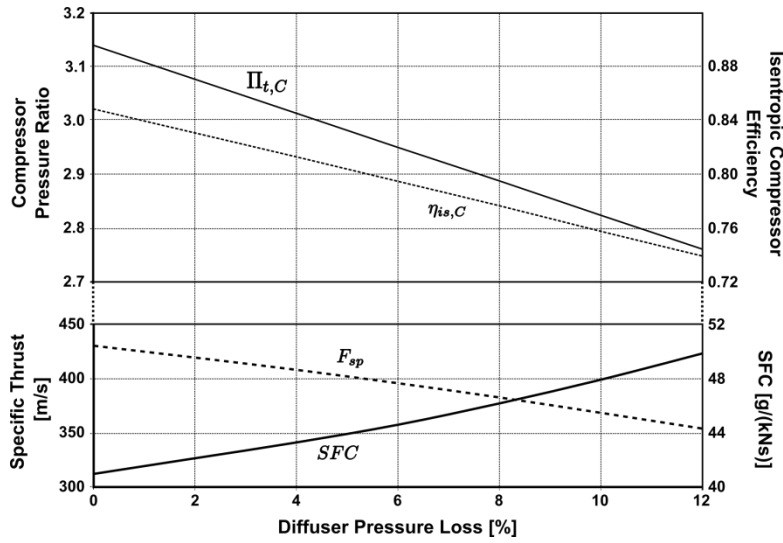


Figure 8: Effect of diffuser pressure loss

The thermodynamically relevant effect of compressor diffuser losses is the reduction of the turbine inlet pressure and therefore the achievable enthalpy gain when expanding the fluid. This is also true for the pressure losses in the combustion chamber. Hence, for an analysis of the overall performance, the origin of the losses does not matter. Figure 9 shows the result of a parametric study where both the compressor diffuser losses and the combustion chamber pressure losses are varied independently. The effect on SFC and F_{sp} is only dependent on the sum of both pressure losses.

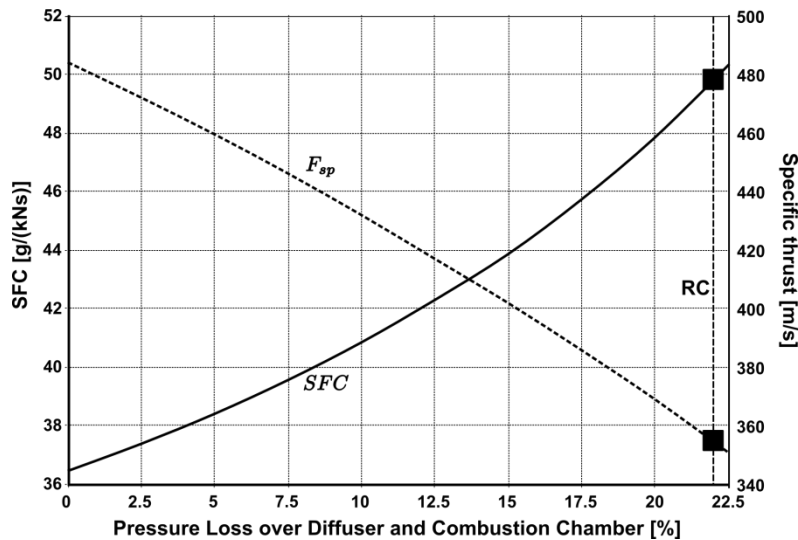


Figure 9: Combined effect of diffuser and combustion chamber pressure loss
(Reference cycle RC at 22.1 percent)

The main purpose of the diffuser is to reduce the flow velocity coming from the impeller in order to maintain a stable combustion. However, influence parameters on combustion such as velocity profiles and pressure distribution are not yet fully understood. Experimental as well as numerical analyses have shown that the diffuser does not completely remove the angular momentum from the flow. This, however, might benefit the combustion efficiency due to better mixing characteristics. The intersection between diffuser and combustion chamber therefore poses promising optimization potential. It leads to the conclusion that diffuser and combustion chamber should not be approached as individual components but rather as a single unit.

Combustion efficiency directly affects the amount of fuel mass flow necessary for maintaining the cycle. It also has a minor effect on power balance between turbine and compressor as it adds to the exhaust mass flow. Improving the combustion efficiency can be achieved via two ways. The first method is to increase the efficiency of the burner as such. New combustion chamber design with optimized liner perforation can improve mixing and ignition in the combustion zone. Combustion chamber liner design is subject of current research at the Institute for Flight Propulsion. The second possibility is to recycle the lubrication fuel mass flow into the combustion zone. This could be accomplished by hollow turbine vanes guiding inside to the vaporizer sticks. However, realizations have not yet been carried out. The same task could also be achieved with a separate oil system. This, however, appears not to be a viable option for small jet engines below 1000 N as it would increase engine size, weight, complexity and cost.

The turbine efficiency in this analysis is very low compared to larger axial turbines. This can partly be explained by small size effects such as low Reynolds numbers, clearance gaps and surface quality. However, the lubrication system of the bearings also has an impact on turbine efficiency. As the lubrication mass flow unites with the main exhaust gas flow between turbine vane and rotor, mixing losses occur. As Figure 10 illustrates, the lubrication mass flow has a radial direction leading to a 90° shear flow in the mixing zone. Moreover, the TJ 74 engine, which is analyzed in this study, utilizes a point welded steel turbine stator. Newer gas turbines are equipped with integral investment casted parts out of nickel alloys. This is primarily because of better temperature resistance. However, smoother surfaces and lower manufacturing tolerances help reducing friction losses and therefore increase turbine efficiency.

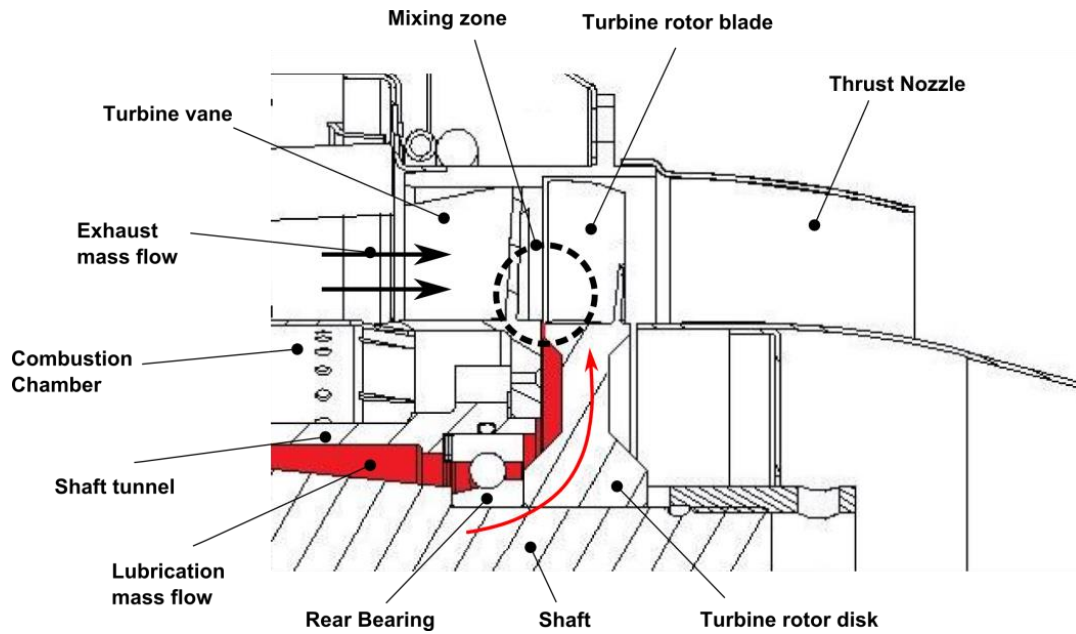


Figure 10: Mixing of lubrication and exhaust mass flow

5.2 Comparison of loss factors on performance

Improvement of components can be quantified in relative reduction of the occurring losses. Loss reduction LR_x can be evaluated with

$$LR_x = 100\% \cdot \left(\frac{x - x_{ref}}{1 - x_{ref}} \right) \quad (4)$$

where x represents the loss factor, i.e. pressure ratios if pressure losses occur or efficiencies if applicable. x_{ref} is the reference value of the loss factor from the cycle calculation above. Pressure ratios, respectively efficiencies of 1 would therefore reduce losses by 100 percent. Figure 11 shows the influence of loss reduction on specific fuel

consumption and specific thrust. The loss factors are diffuser pressure ratio, combustion efficiency, combustion chamber pressure ratio and turbine efficiency. The graphs each result from the change of a single parameter.

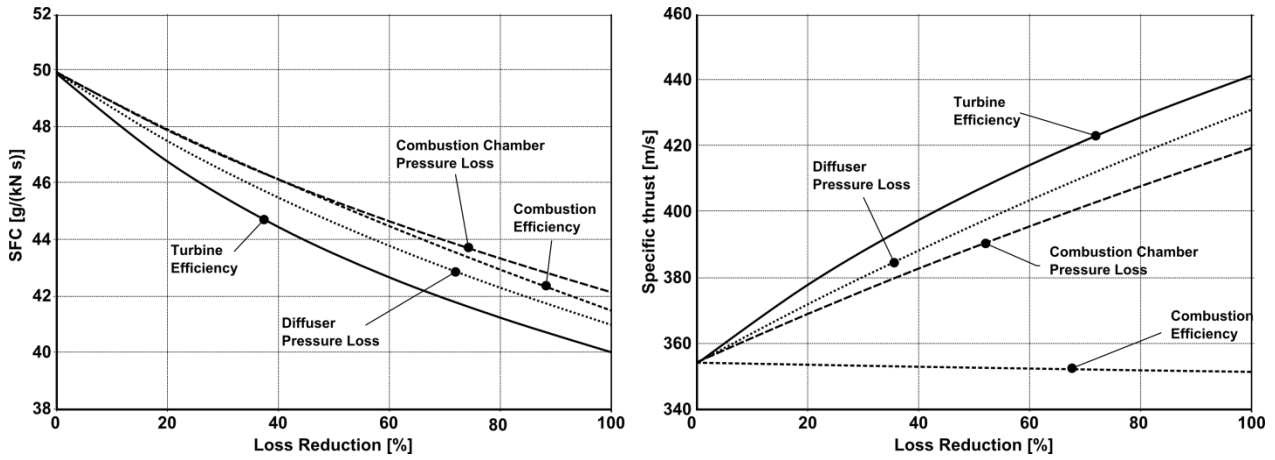


Figure 11: Influence of loss reduction on performance

The result shows that turbine efficiency theoretically poses the highest potential to increase engine performance. Partial improvement can be achieved with new turbine vane design and investment casting technique. Alternative lubrication systems might also increase turbine efficiency. However, as small size effects in turbo machinery remain, further improvement has to be considered as limited.

The effect of the other three factors is smaller but still significant. The influence of the combustion efficiency on specific thrust is an exception as an improvement only reduced the required fuel mass flow. This leads to a slightly lower exhaust mass flow requiring a higher specific work output from the turbine. The result is a reduced enthalpy level in the thrust nozzle, which entails a small reduction in specific thrust.

This analysis also shows the large potential that lies in the improvement of the intersection between compressor diffuser and combustion chamber. Higher combustion efficiency requires a better mixture of fuel and air mass flows. To a large extent mixture quality is a result of how the air mass flow is injected into the combustion zone. This, however, still requires a better understanding of mixing and vaporization processes with combustion chambers of small gas turbines. New diffuser geometry can then provide appropriate flow conditions. An optimized intersection can further lead to reduced pressure losses in both the diffuser and the combustion chamber.

5.3 Effects of improved components on optimal design parameters

Recycling of the lubrication mass flow into the combustion chamber is a method to increase both combustion efficiency as well as turbine efficiency. Assuming five percent lubrication fuel flow, combustion efficiency rises to 86.8 percent, which corresponds with a loss reduction of 24.6 percent. With the assumption, that losses within the turbine can be reduced by 10 percent by avoiding mixture losses, recycling can decrease specific fuel consumption by 8 percent and raise specific thrust by 3.2 percent.

In the following, a generic case is presented in order to evaluate the effect of component improvements on optimal design parameter, i.e. compressor pressure ratio and turbine inlet temperature. For this analysis a reduction of 30 percent for all loss factors is applied. This leads to a new compressor efficiency $\eta_{is,C}$ of 0.773, a combustion efficiency η_{Comb} of 0.878, a combustion chamber pressure ratio $\Pi_{t,CC}$ of 0.929 and a turbine efficiency $\eta_{is,T}$ of 0.848. For a constant work balance between the rotating components, this leads to an increased compressor pressure ratio $\Pi_{t,C}$ of 2.88 due to reduced losses in the diffuser. Figure 12 shows the result of this case study.

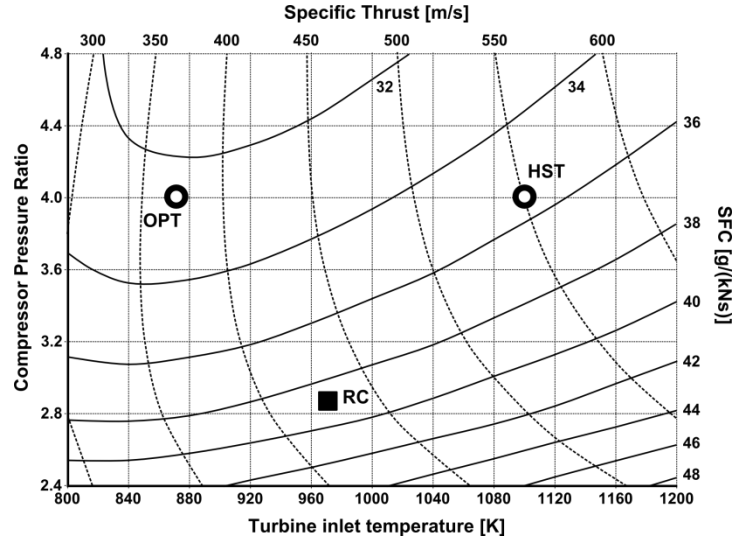


Figure 12: Performance over Design Parameters with improved components
(RC – reference cycle, OPT – optimal specific fuel consumption, HST – high specific thrust)

The loss reduction (point RC in Figure 12) leads to a reduction of specific fuel consumption by 22.4 percent to 38.7 g/(kNs) compared to the former reference cycle. The specific thrust is 429 m/s, an increase by 21.2 percent. For minimum *SFC* the optimal turbine inlet temperature is between 800 and 900 K, depending on compressor pressure ratio. For a pressure ratio of 4 and a turbine inlet temperature of 868 K a specific fuel consumption of 32.5 g/(kNs) could be achieved (point OPT in Figure 12). Compared to the reference cycle, turbine inlet temperature has to be reduced for higher efficiency. However, specific thrust increases with higher temperatures. A turbine inlet temperature of 1100 K at a pressure ratio of 4 would increase the specific thrust by 49 percent to 550 m/s compared to point OPT but also increase the *SFC* by 9.2 percent to 35.5 g/(kNs) (point HST in Figure 12). This trade-off in performance can be viable for applications with high thrust requirements. Higher temperatures demand either a reduction of hot spots through new combustion chamber design or reduced lifetime requirements. Investigations of advanced nickel-alloy materials indicate that turbine inlet temperatures of more than 1300 K are possible with a completely even temperature profile [17].

6. Concluding remarks

The investigation of a small gas turbine through experimental testing as well as numerical simulations exposed that accurate thermodynamic data from the cycle process is difficult to access. This is mainly due to circumferentially uneven velocity and temperature profiles. A sufficient high resolution of measuring probes cannot be installed because geometries are too small. Numerical analyses help identifying such uncertainties; however, they have not yet been validated. A sensitivity analysis has shown that very small changes of measured data can already lead to significant deviations of iterated design values.

Nevertheless, the thermodynamic analysis was extended in order to evaluate improvement potentials for small gas turbines. The definition of component loss reductions was used to quantify improvement potentials. It could be shown that the intersection between compressor diffuser and combustion chamber poses considerable potential as it affects diffuser as well as combustion chamber pressure losses. Moreover, combustion efficiency is strongly dependent on mixing quality, which is affected by air flow velocity and direction coming from the diffuser. In contrast to conservative approaches, both components should not be assessed individually but rather as a single unit.

Realistic assumptions for component improvement lead to specific fuel consumptions far lower than state-of-the-art engines achieve. Higher pressure ratios benefit performance but increase complexity and weight of the engine. The reduction of hot spots in the turbine section with new combustion techniques can provide higher average turbine inlet temperatures and therefore increase specific thrust without reducing turbine lifetime. A homogenous temperature and velocity profile at turbine inlet section is an important prerequisite for the success of further optimization of turbine efficiency.

Due to the diversity in applications of small gas turbines, an overall optimization of small gas turbines includes additional key aspects. Besides efficiency and thrust requirements, there are also requirements like small engine size, fuel flexibility, low complexity and acquisition cost. Small gas turbine design remains primarily an application-driven task, but still with great potential for additional improvements.

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