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Performance Increases for Gas-Turbine Engines Through Combustion Inside the Turbine

W. A. Sirignano* and F. Liu†
University of California, Irvine, Irvine, California 92697

In a conventional gas-turbine engine, fuel is burned in separate combustors before the heated high-pressure gas expands through the turbine to provide shaft power for the compressor, fan, propellers, helicoptor rotors, or an electric generator in a ground-based powerplant application. It is proposed in this paper that combustion be continued purposely inside the turbine to increase the efficiency and specific thrust/power of the engine. We term such a turbine with combustion a turbine-burner. A thermodynamic cycle analysis is performed to demonstrate the performance gains of turbojet engines with the turbine burner over the conventional turbojets. Ground-based gas-turbine engines for power generation are also analyzed, with the results showing even better performance gains compared without conventional engines. A mixing-layer analysis with combustion in an accelarated flow similar to the conditions in the turbine burner shows that there is also potential reduction of NO_x by using the turbine-burner compared with conventional combustors, where the burning is at a constant pressure. Challenges and related research issues that must be addressed to use the turbine-burner technology are identified in this paper.

I. Introduction

GAS-turbine engine designers are attempting to increase thrust-to-weight ratio and to widen the thrust range of engine operation. One major consequence is that the combustor residence time can become shorter than the time required to complete combustion. Therefore, combustion would occur in the turbine passages, which, in general, has been considered to be undesirable.

A thermodynamic analysis indicates, however, that significant benefit can result from augmented burning in the turbine. The analysis is presented in Sec. II. In summary, it is shown that augmented combustion in the turbine allows for 1) a reduction in afterburner length and weight, 2) a reduction in specific fuel consumption compared with the use of an afterburner, and 3) an increase in specific thrust (ST). The increase in ST implies that larger thrust can be achieved with the same cross section or that the same thrust can be achieved with a smaller cross section (and, therefore, still smaller weight). For ground-based engines, it is also shown that combustion in the turbine coupled with heat regeneration dramatically increases specific power and thermal efficiency.

Section III discusses results of a boundary-layer analysis with mixing and reaction in an accelerating flowfield that can be viewed as a simplified model for the reacting flow inside a turbine-burner. The analysis shows that the peak temperature in such a layer is reduced as the streamwise distance increases. This indicates the possibility of reduced NO_x production.

In view of the previously mentioned analyses, it is clear that mixing and exothermic chemical reaction in the turbine passages offer an opportunity for a major technological improvement. Instead of the initial view that it is a problem that resulted from attempts to burn more in the same or less volume, it should be seen as an opportunity to improve performance and reduce weight.

Combustion in the turbine passages presents a new challenge to combustion and turbomachinery engineers. Section IV attempts to identify some of the challenges for the turbine-burner technology before finally drawing the conclusions.

II. Thermal Analysis of Turbine Burners

In this section, a thermodynamic cycle analysis is performed for jet engines and ground-based gas-turbine engines to illustrate the merits of a turbine-burner vs the conventional separate combustor and turbine configuration.

A. Performance of Turbojets

Consider jet engines for propulsion purposes first. Figure 1 shows the *T-s* diagram of a jet-engine thermocycle. The farstream flow (state *a*) is compressed through the inlet and the compressor to state 03 (0 denotes the stagnation state) before going into the conventional main burner, where heat $\dot{Q}_b$ is added to increase the flow temperature to T_{04}. In a conventional configuration (dashed line in Fig. 1), the high-pressure, high-temperature gas expands through the turbine, which provides just enough power to drive the compressor and other engine auxiliaries. Fuel is injected and burned in the optional afterburner to further increase the temperature of the gas before the flow expands through the nozzle to produce the high-speed jet. The afterburner increases the power levels of the engine, but because fuel is burned at a lower pressure compared with that at the main burner, the overall cycle efficiency is reduced. Therefore, the use of afterburners increases the fuel consumption rate as well as specific thrust.

To remedy the efficiency decrease because of the use of the afterburner, one may consider adding heat in the turbine, where the pressure levels are higher than in the afterburner. For this purpose the turbine-burner is conceived. In a turbine-burner, fuel is burned in the turbine, heating the flow while doing work on the rotor at the same time. Depending on the relative magnitude of the two processes, the stagnation temperature of the flow may increase or decrease. The former is not desirable because it will cause the total temperature to exceed the material limit. For a given temperature range, we know that the most efficient cycle is the Carnot cycle, in which heating occurs at a constant temperature. It appears then

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*Professor, Department of Mechanical and Aerospace Engineering. Fellow AIAA.

†Associate Professor, Department of Mechanical and Aerospace Engineering. Member AIAA.

that the ideal option is to organize the heating so that the stagnation temperature in the turbine stays constant. An optional afterburner may still be used behind the turbine-burner before the flow goes to the nozzle, if extra thrust is needed. This process is marked by the solid line between states 04 and 7.

A simple and yet illuminating cycle analysis is performed to compare the relative performances of the conventional configuration and the new turbine-burner configuration. The calculations follow the outline in the textbook by Hill and Peterson.[1] We denote the pressure and temperatures by p and T and use subscripts to distinguish the various stations in the thermocycle. A subscript 0 denotes stagnation properties. We assume 1) no air bleeding, 2) complete expansion, 3) perfect gas and constant gas properties, 4) no auxiliary power extracted from the turbine so that the turbine power exactly balances the compressor power, 5) heating is organized in the turbine-burner to maintain constant stagnation temperature, and 6) component efficiencies are listed in Table 1.

The preceding assumptions are not accurate enough for real design purposes. However, they suffice here for the purpose of proof of concept. We attempt to use typical values of component efficiencies for modern jet engines. Some of the loss models may be simple and the values of efficiencies may vary from design to design. However, our calculations show that within some range of the efficiency values, the performances of the engines with and without the turbine-burner have the same trend as the comparisons displayed in this section. The following steps are used in the cycle calculations.

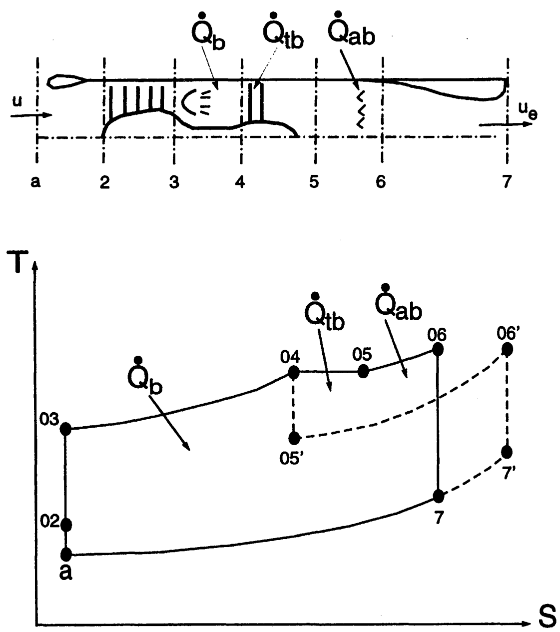

Fig. 1 Comparison of thermodynamic cycles with and without the turbine-burner.

1. Inlet and Diffuser Section: a-02

The stagnation temperature and stagnation pressure at the exit of the diffuser are calculated by the following equations:

$$T_{02} = T_a\{1 + [(\gamma - 1)/2]M^2\} \tag{1}$$

$$p_{02} = p_a\{1 + \eta_d[(T_{02}/T_a) - 1]\}^{\gamma/(\gamma-1)} \tag{2}$$

where γ is the ratio of the specific heats of air, M is the flight Mach number, and η_d is the adiabatic efficiency of the inlet and diffuser.

2. Compressor: 02-03

Engine compressor ratio π_c is a design parameter that is used as input in the performance calculations:

$$p_{03}/p_{02} = \pi_c \tag{3}$$

The stagnation temperature ratio can then be calculated as

$$T_{03}/T_{02} = 1 + (1/\eta_c)[(\pi_c)^{(\gamma-1)/\gamma} - 1] \tag{4}$$

where η_c is the adiabatic efficiency of the compressor.

The power needed to drive the compressor is then

$$\mathcal{P}_c = \dot{m}_a C_p(T_{03} - T_{02}) = \dot{m}_a C_p T_{02}[(T_{03}/T_{02}) - 1] \tag{5}$$

where $\dot{m}_a$ is the air-mass flow rate to the engine, and C_p is the constant-pressure specific heat.

3. Main Burner: 03-04

The energy equation applied to the combustion process, if we neglect the enthalpy of the incoming fuel, is

$$\dot{m}_{fb} Q_R \eta_b = C_p T_{04}(\dot{m}_a + \dot{m}_{fb}) - C_p T_{03} \dot{m}_a$$

Therefore, we get the ratio of fuel mass flow rate to the air mass flow rate in the main burner

$$\frac{\dot{m}_{fb}}{\dot{m}_a} = \frac{C_p(T_{04} - T_{03})}{Q_R \eta_b - C_p T_{04}} \tag{6}$$

where Q_R is the heating value of the fuel, and η_b is the burner fuel efficiency.

The stagnation pressure loss in the combustor can be taken into account by a stagnation pressure recovery factor π_b. Thus, we write

$$p_{04} = \pi_b p_{03} \tag{7}$$

4. Conventional Turbine Without Burning: 04-05'

Under the assumptions that there is no extra power extracted to drive the engine auxiliaries and also that there is no loss of

Table 1 Efficiencies of components used in the calculation

Component	Symbol	Value
Inlet/diffuser adiabatic efficiency	η_d	0.95
Compressor adiabatic efficiency	η_c	0.85
Main burner combustion efficiency	η_b	0.98
Main burner total pressure recovery coefficient	π_b	0.99
Main turbine adiabatic efficiency	η_t	0.95
Turbine-burner combustion efficiency	η_{tb}	0.96
Additional turbine-burner total pressure recovery coefficient	π_{tb}	0.98
Afterburner combustion efficiency	η_{ab}	0.98
Afterburner total pressure recovery coefficient	π_{ab}	0.99
Nozzle adiabatic efficiency	η_n	0.97

power because of friction of the mechanical systems, the turbine must generate enough power to balance that power consumed by the compressor

$$\mathcal{P}_t = \mathcal{P}_c \tag{8}$$

or

$$\dot{m}_t C_{pt}(T_{04} - T_{05'}) = \dot{m}_c C_{pc}(T_{03} - T_{02})$$

where we have used the subscripts c and t in $\dot{m}_c$, $\dot{m}_t$, C_{pc}, and C_{pt} to distinguish the mass flow rates, and constant-pressure specific heats in the compressor and turbine. In general, they differ for the compressor and turbine. Fuel is added to $\dot{m}_t$; air is bled from the compressor and also injected to the turbine to cool the turbine blades. As a first approximation, however, we simply assume

$$\dot{m}_t C_{pt} = \dot{m}_c C_{pc}$$

The stagnation temperature and pressure ratios across the turbine can then be calculated by

$$\frac{T_{05'}}{T_{04}} = 1 - \frac{T_{03} - T_{02}}{T_{04}} \tag{9}$$

$$\frac{p_{05'}}{p_{04}} = \left[1 - \frac{1}{\eta_t}\left(1 - \frac{T_{05'}}{T_{04}}\right)\right]^{\gamma/(\gamma-1)} \tag{10}$$

where η_t is the turbine adiabatic efficiency.

5. Turbine-Burner: 04-05

In the case where intentional controlled burning occurs in the turbine, the temperature in the turbine is held constant at the maximum turbine inlet temperature. The following equation replaces Eq. (9):

$$T_{05} = T_{04} \tag{11}$$

The energy released in combustion in the turbine-burner is not used to increase the temperature of the gas. Rather, it is converted into turbine work required to drive the compressor. This is possible because the gas is expanding through the rotor while the energy is being released so that the total temperature of the gas remains constant. Strictly speaking, this ideal condition can only be achieved when burning occurs in the rotor. Equation (11) is used here in the sense of averaging across the complete turbine when heat is added in the stator or in both the rotor and stator.

Because the total temperature is kept constant in the turbine, an appropriate amount of fuel must be burned to provide power for turbine rotor. It follows from the energy equation that the turbine-burner fuel mass flow rate is

$$\dot{m}_{ftb} = \mathcal{P}_t / Q_R \eta_{tb} \tag{12}$$

where η_{tb} is the turbine-burner fuel efficiency.

The heat added to the flow can also be estimated as $T_{av}(s_5 - s_4)$, where T_{av} is a mean temperature of the flow, and $(s_5 - s_4)$ is the entropy change across the turbine-burner. T_{av} should be a value between T_{04} and the lowest static temperature in the turbine. If we estimate a mean flow Mach number of $M_{av} = 0.7$, in the turbine, T_{av} can then be estimated as

$$T_{av} = \frac{T_{04}}{1 + [(\gamma - 1)/2]M_{av}^2} \approx 0.91 T_{04} \tag{13}$$

The entropy change and the pressure ratio across the turbine-burner can be calculated as

$$s_5 - s_4 = \frac{\dot{m}_{ftb} Q_R \eta_{tb}}{(\dot{m}_a + \dot{m}_{fb})T_{av}} = \frac{\mathcal{P}_t}{(\dot{m}_a + \dot{m}_{fb})T_{av}} \tag{14}$$

$$p_{05}/p_{04} = \exp\{-[(s_5 - s_4)/R]\} \tag{15}$$

Equation (15) accounts for the stagnation pressure loss caused by combustion in the turbine. Considering that there may be additional pressure losses because of the complicated aerodynamics in the presence of combustion, a stagnation pressure recovery factor π_{tb}, which should be less than 1, is introduced so that

$$p_{05}/p_{04} = \pi_{tb} \exp\{-[(s_5 - s_4)/R]\} \tag{16}$$

which replaces Eq. (10).

6. Afterburner Section: 05-06 or 05'-06'

In the cases where there is an afterburner or a reduced afterburner, enough fuel is added after the turbine to bring the temperature up to the maximum allowable afterburner temperature T_{06}; then expansion through a nozzle occurs. Assuming a stagnation pressure recovery factor π_{ab} and a burner efficiency η_{ab} for the afterburner, we can write

$$p_{06} = p_{05}\pi_{ab} \tag{17}$$

$$\frac{\dot{m}_{fab}}{\dot{m}_a + \dot{m}_{fb} + \dot{m}_{ftb}} = \frac{C_p(T_{06} - T_{05})}{Q_R \eta_{ab} - C_p T_{06}} \tag{18}$$

where $\dot{m}_{fab}$ is the fuel mass flow rate for the afterburner.

7. Nozzle Section: 06-7 and 06'-7'

Assuming complete expansion at the exit of the nozzle and an adiabatic nozzle efficiency η_n, we have

$$T_{07} = T_{06} \tag{19}$$

$$p_7 = p_a \tag{20}$$

The exit flow velocity is then calculated as

$$u_e = \sqrt{2\eta_n C_p T_{06}[1 - (p_a/p_{06})^{(\gamma-1)/\gamma}]} \tag{21}$$

Eqs. (17–21) are applicable for both the conventional and the turbine-burner configurations. The subscripts 05, 06, and 07 can be replaced by 05′, 06′, and 07′ when the conventional configuration in Fig. 1 is considered.

8. Turbojet Engine Performance

For jet engines designed to produce thrust, two performance parameters are of interest: specific thrust, ST, and thrust specific fuel consumption rate, TSFC. ST is defined as the thrust per unit mass flux of air

$$\mathrm{ST} = \frac{\mathcal{T}}{\dot{m}_a} = \left[\left(1 + \frac{\dot{m}_{fb}}{\dot{m}_a} + \frac{\dot{m}_{ftb}}{\dot{m}_a} + \frac{\dot{m}_{fab}}{\dot{m}_a}\right) u_e - u\right] \tag{22}$$

A higher ST means a higher thrust level for the same engine cross section and, thus, smaller engine size and lighter weight in general.

The TSFC is defined as the fuel flow rate per unit thrust

$$\mathrm{TSFC} = \frac{\dot{m}_f}{\mathcal{T}} = \frac{\dot{m}_{fb} + \dot{m}_{ftb} + \dot{m}_{fab}}{\mathrm{ST}\dot{m}_a} \tag{23}$$

A lower TSFC means less fuel consumption for the same thrust level.

A simple C language program was written to perform the preceding calculations. Figures 2–5 compare the results for four types of gas turbine combustion configurations:

1) The conventional turbojet engine without afterburner.
2) The conventional turbojet engine with afterburner.
3) Turbojet engine with a turbine-burner but no afterburner.
4) Turbojet engine with a turbine-burner and with a reduced afterburner.

We compare the engine performances at different flight Mach numbers and compressor compression ratios. In any particular comparison, one of the two parameters, the flight Mach number or the compressor compression ratio, is held fixed while the other is varied within a range. Figures 2 and 3 show the performances of the engines with respect to flight Mach number. While the conventional base engine without the afterburner gives the best fuel efficiency, its ST is limited, which means that the total thrust level cannot be increased without having a larger engine (bigger engine cross section, larger volume, and heavier weight). To obtain the extra thrust that is

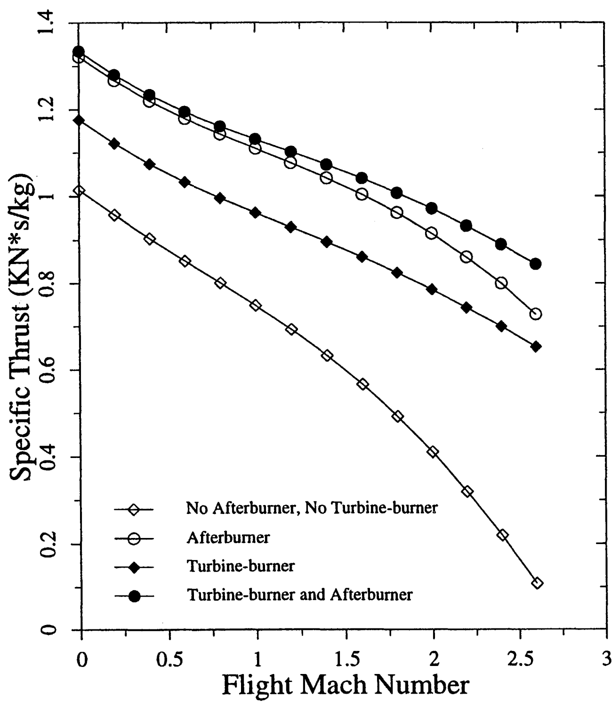

Fig. 2 ST vs flight Mach number, T_{04} = 1500 K, T_{06} = 1900 K, π_c = 20.

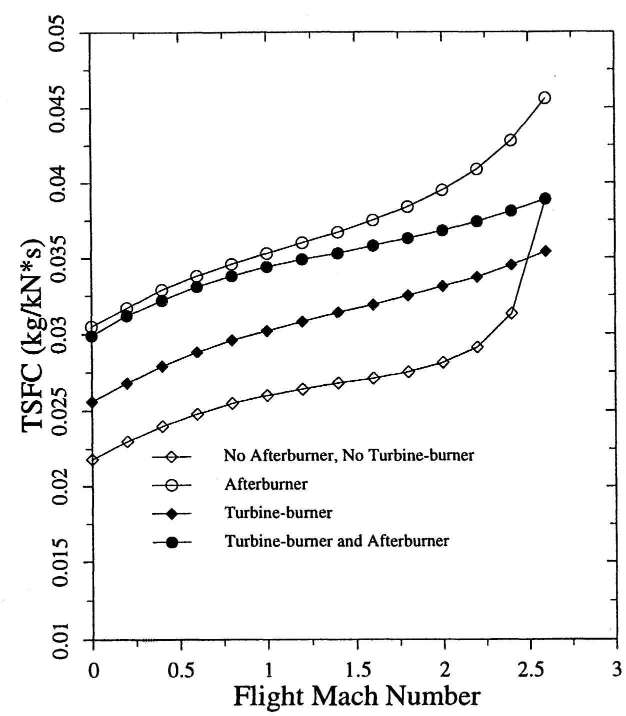

Fig. 3 TSFC vs flight Mach number, T_{04} = 1500 K, T_{06} = 1900 K, π_c = 20.

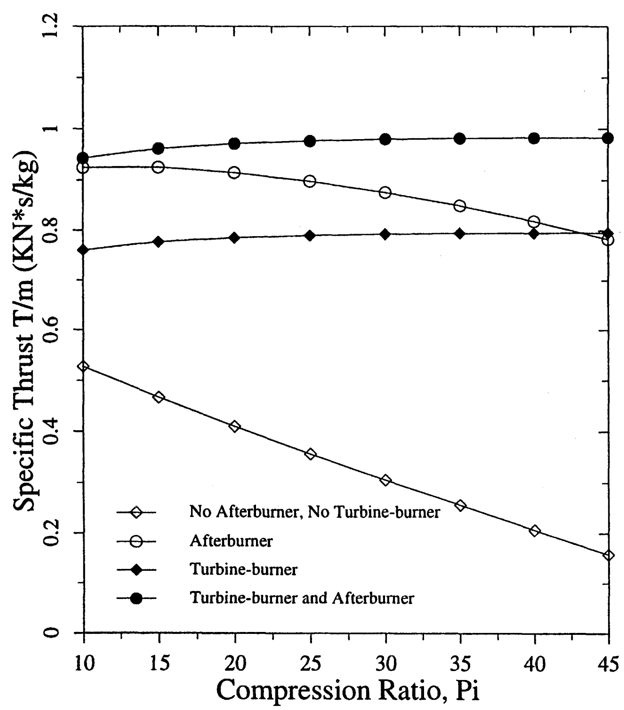

Fig. 4 ST vs compression ratio, T_{04} = 1500 K, T_{06} = 1900 K, M = 2.

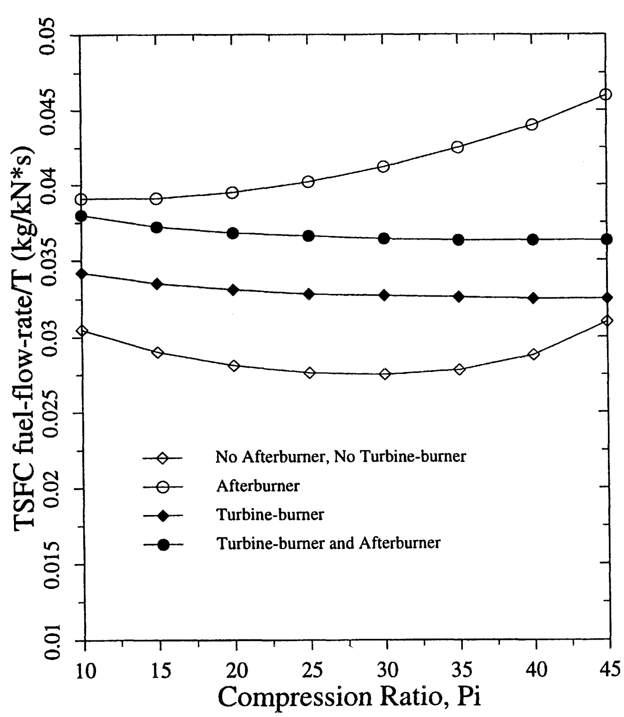

Fig. 5 TSFC vs compression ratio, T_{04} = 1500 K, T = 1900 K, M = 2.

often needed by military jet engines or even the Concord supersonic civil transport for portions of their flights, the afterburner configuration is used. The afterburner greatly boosts the engine specific thrust, but at the cost of a very high fuel consumption rate, as shown by the open circles in Figs. 2 and 3. The higher ST is achieved by burning extra fuel in a secondary combustion chamber without going beyond the maximum turbine inlet temperature. However, because the extra fuel is burned at a lower gas pressure behind the turbine, the overall cycle efficiency is reduced and, hence, the higher TSFC. In addition, the afterburner increases the length and weight of the engine even though the cross section of the engine is kept the same.

The turbine-burner concept seeks to burn extra fuel inside the turbine at a higher gas pressure and at an appoximately constant temperature, not exceeding the maximum turbine inlet temperature by taking advantage of combining the process of combustion and gas expansion in doing work to the turbine blades. This configuration significantly increases the engine ST without the large increase in the fuel consumption rate. Figures 2 and 3 show an overall 20% increase in ST, with only about 10% increase of TSFC for the turbine-burner configuration, whereas the afterburner design increases TSFC by almost 50% for a 20% ST increase. In addition, elimination of the afterburner reduces the engine length and weight.

If larger thrust levels are needed beyond the turbine-burner design, one may still use an afterbuner, using smaller levels this time because of the reduced combustion needed after fuel has already been burned in the turbine. This turbine-burner with a reduced afterburner design produces a higher specific thrust and yet consumes less fuel compared with the conventional afterburner design, as shown by the solid circles in Figs. 2 and 3.

The benefit of the turbine-burner increases with increasing flight Mach number. Figures 2 and 3 show that the specific thrust and fuel efficiency deteriorate with flight Mach number for all configurations because the compressor inlet temperature becomes high and, also, the stagnation pressure loss increases in the diffuser at higher flight Mach numbers. Higher compressor inlet temperatures increase the amount of power needed to drive the compressor and also limit the amount of fuel that can be added in the main combustor without exceeding the maximum turbine inlet temperature. At higher flight Mach numbers, the conventional base engine shows a drastic increase in fuel consumption rate because of the increased losses in the diffuser, compressor, and turbine. However, the configurations with the turbine-burner show a slower deterioration of both performance measures compared with those of the convention engine with and without the afterburner. In particular, the turbine-burner without the reduced afterburner option shows the best performance gains at higher flight Mach numbers.

We now turn our attention to the performance of the engines with respect to different design compression ratios of the compressor. As shown by Figs. 4 and 5, the relative standing of the four configurations in performance stay the same as before over a large range of compressor pressure ratios. In fact, the performance gains of the turbine-burner design increase even more dramatically with increasing compression ratios rather than with increasing Mach numbers, with or without the afterburner. The ST decreases rapidly with increasing compression ratio for the conventional base engine. Figure 4 indicates, however, that this trend is reversed by the turbine-burner design. This is because of the added energy input and the fact that the turbine-burner produces work at a lower cost of pressure drop compared with the conventional engine. At the high compressor pressure ratios, the turbine-burner configuration without an afterburner produces almost equal ST as that of the conventional afterburner design, without the large increase in fuel consumption rate associated with the afterburner configuration.

From the preceding discussion, it is clear that the turbine-burner with the reduced afterburner has the best (largest) ST with the same temperature limitations because of materials; a 10% or so improvement in the thrust level can be achieved at the higher compression ratios and higher flight Mach numbers compared with the conventional afterburner case. This implies that the turbine-burner/reduced afterburner design can obtain a larger thrust with a shorter length and lower weight engine than the afterburner case. Or, it can achieve the same thrust with both reduced engine cross section and reduced engine length compared with the afterburner design. The specific fuel consumption for the turbine-burner/reduced afterburner design is better (lower) than for the afterburner design. Obviously, this can imply a reduced weight and volume of the aircraft for a fixed range, or an increase in the range for the same amount of fuel-tank capacity.

The turbine-burner configuration without the reduced afterburner offers the best relative gain compared with the conventional engines. This design has better specific fuel consumption than the two cases with afterburners discussed earlier, whereas it produces much higher specific thrust than the conventional base design and may almost reach the same specific thrust level of the conventional afterburner design at high compressor pressure ratios. The fuel consumption rate is still slightly higher than the conventional turbojet engine with the afterburner turned off. But, because there is no afterburner, the engine length and weight can be significantly reduced. The much increased ST and the elimination of the afterburner may compensate for the somewhat higher TSFC because we can now use smaller and lighter engines for the same airplane for even regular cruise conditions. The gains in reduced drag and weight by using smaller and lighter engines can be significant for high flight Mach numbers.

For these calculations, the ambient condition is taken at an altitude of about 40,000 ft, where T_a = 216.7 K, and P_a = 18,750 Pa. The temperature at the turbine inlet T_{04} is fixed at 1500 K. In those cases where there is an afterburner, the peak temperature reached at the end of the afterburner is T_{06} = 1900 K. The preceding maximum turbine inlet temperature and the maximum afterburner temperature are chosen based on values on typical engines. To see if the effect of the turbine-burner would be sensitive to the maximum turbine inlet temperature T_{04}, we calculated the engine performances for a range of T_{04} from 1400 to 1800 K at the flight condition $M = 2$ and a design compressor compression ratio of 20. These results are shown in Figs. 6 and 7. Higher T_{04} gives higher ST, but slightly lower TSFC for all configurations. However, the relative standing of the four configurations are the same within the entire range of T_{04} considered.

The previous discussions have been limited to turbojet engines only. We also performed calculations for turbofan engines. The relative standing in performance of the four configurations remains the same. In fact, the relative performance gain of the turbine-burner becomes even wider for the turbofan engine as the bypass ratio becomes larger. Nevertheless, more systematic studies are needed for applications to subsonic civil transport engines with large fan bypass ratios.

B. Performance of Ground-Based Gas Turbines

We also consider gas-turbine engines used for electric power generation or other ground- or sea-based purposes. Nationwide, 36% of the total energy consumption is for electric generation and 27% is for transportation.[2] The efficiency of the gas turbines used in power generation are of major importance to the conservation of energy. A small percentage increase in the efficiency of such turbomachines would have significant economic implications for the energy use.

Figure 8a shows the thermocycle of a ground-based gas-turbine engine with or without a turbine-burner. Compared with the jet-engine cycle, the ground-based gas turbine is used to generate shaft power and, thus, there is no signifi-

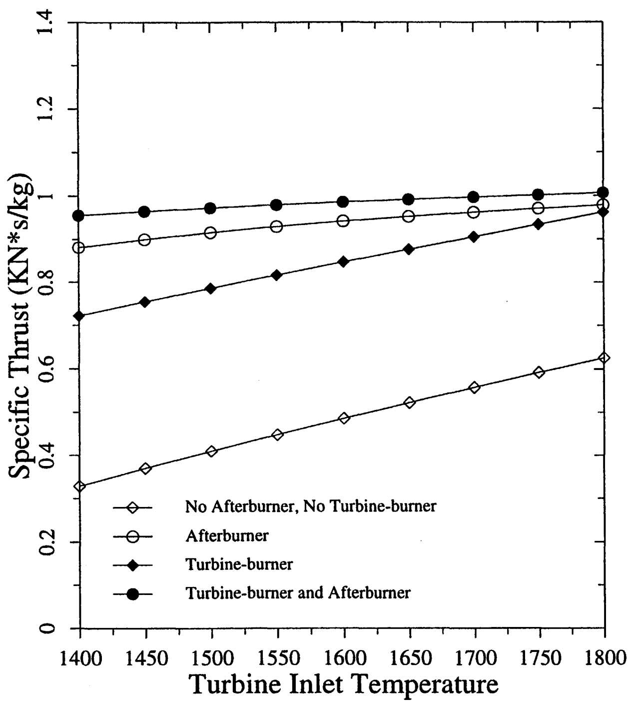

Fig. 6 ST vs turbine inlet temperature T_{06} = 1900 K, M = 2.0, π_c = 20.

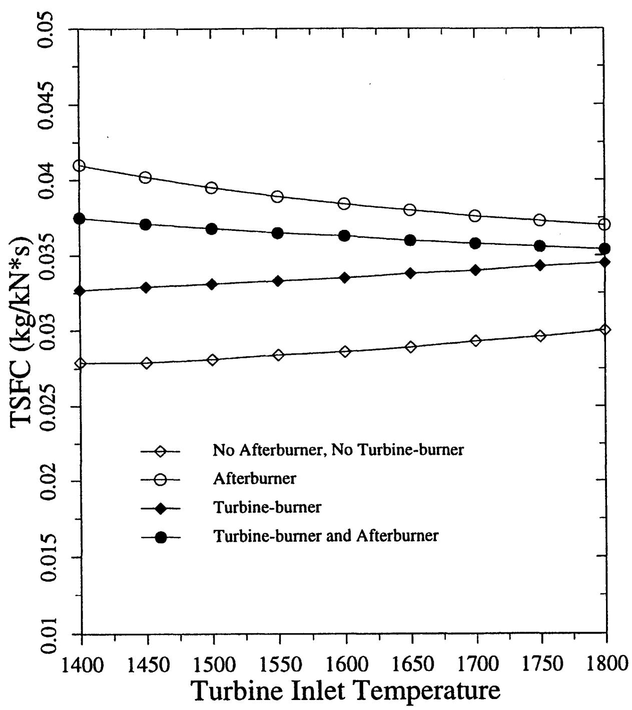

Fig. 7 TSFC vs turbine inlet temperature T_{06} = 1900 K, $\dot{M}$ = 2.0, π_c = 20.

cant nozzle section. All of the thermal energy of the high-temperature, high-pressure gas is converted to turbine shaft power. In addition, for a ground-based engine, the residual heat from the exhaust of the engine can be utilized to preheat the flow before it goes to the main combustor, as shown by segment 2 to 3 in Fig. 8a. A heat-regeneration effectiveness is defined as

$$E_x = (T_{03} - T_{02})/(T_{05} - T_{02}) \tag{24}$$

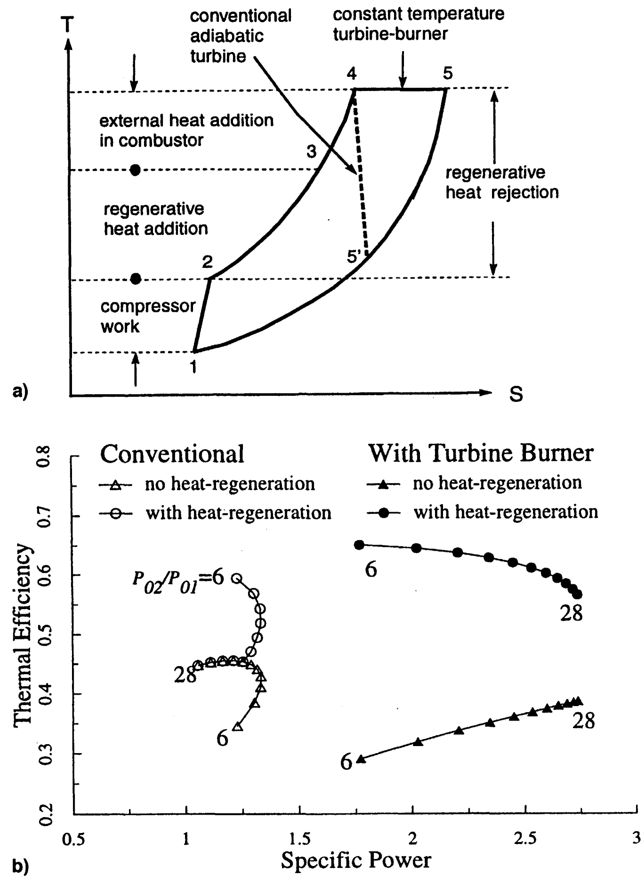

Fig. 8 Thermal cycle and efficiency, the conventional adiabatic turbine (dashed line) is replaced by a constant (average) temperature burner that functions as a combustor and the power-generating turbine, assuming 92% polytropic efficiency for the compressor, 95% for the turbine, and 95% for the heat regeneration effectiveness.

Instead of ST and TSFC, the appropriate performance measures for a ground-based engine are specific power and overall thermal efficiency, which are defined next:

$$SP = \dot{W}/\dot{m}_a C_p T_{01} \tag{25}$$

$$\eta = \dot{W}/\dot{m}_f Q_R \tag{26}$$

The calculation procedure for the ground-based machines is rather similar to that listed earlier, with some minor modifications to accommodate the heat-regeneration process and the new definition of performance measures. Figure 8b shows the comparison of the thermal efficiency and specific power for a conventional gas turbine and one with a turbine-burner with and without heat regeneration. Performances are calculated for compressor pressure ratios ranging from 6 to 28, which are marked in the figure. A 95% heat-regeneration effectiveness is assumed. Polytropic efficiencies of 0.92 and 0.95 for the compressor and turbine, respectively, are used in the calculations in this case instead of adiabatic efficiencies.

Consider first the case without heat regeneration. As pressure increases from 6, both the specific power and the thermal efficiency increase until the pressure ratio reaches 11. At that point the temperature at the entrance of the combustor becomes high enough so that heat addition in the main combustor is limited. At this time the specific power starts to decrease while the thermal efficiency continues to increase and almost reaches a plateau. Figure 8b shows that the turbine-burner configuration gives a dramatically larger specific power, although the thermal efficiency is slightly lower than that of the conventional engine. In addition, the specific power and the thermal efficiency show a continuous increase as the compressor pressure ratio increases from 6 to 28. The lower thermal efficiency of the turbine-burner option compared with the conventional configuration is a result of the lower pressure levels in the turbine-burner than in the main combustor, as noted in the discussion for the jet-engine case. However, this situation can be easily corrected by using heat regeneration to utilize the high heat content of the exhaust gas. As shown by Fig. 8b, specific power and thermal efficiencies are higher than the conventional configuration with heat regeneration. The increase in specific power is, in particular, dramatic. A more than two-fold increase in specific power can be achieved at the higher compressor pressure ratio end. Higher specific power means a smaller engine and, thus, a lesser capital investment, for the same power level. The turbine-burner configuration with heat regeneration offers an even greater potential for ground-based machines than that for jet propulsion.

Although more refined calculations must be done for real design, the relatively simple analysis in this section clearly demonstrates the potential performance gains that can be achieved by using a turbine-burner concept for both aeroengines and ground-based engines. Many research problems, however, must be solved to advance the technology of turbine-burners.

III. Combustion in an Accelerating Boundary Layer

There is a lack of fundamental treatment in the literature of multidimensional flows with mixing and chemical reaction in the presence of strong pressure gradients that support a transonic flow. Reacting, multidimensional (laminar and turbulent) low Mach number flows have been considered by many investigators using a wide variety of approaches. A limited number of efforts have been made on reacting supersonic flows. However, even the canonical case of a two-dimensional laminar reacting, mixing shear layer with a large pressure gradient has not been treated in the literature until very recently.[3] This basic example provides a good basis for proceeding to more complex situations such as elliptic/hyperbolic configurations and/or turbulent flows.

Sirignano and Kim[3] examined this simple problem of a laminar, two-dimensional, mixing, reacting layer with a pressure gradient that accelerates the flow in the direction of the primary stream. This model problem has some of the salient features of the burning turbine flow. A similar solution can be found that reduces the partial differential equations to a fifth-order system of ordinary differential equations. The reduced equations have been solved and it is seen that the local free-stream Mach number is an important parameter. Transport rates are very dependent on the Mach numbers of the free-streams.

The solutions for the velocity and for the scalar properties (enthalpy and concentration) are strongly affected by the acceleration parameter and the Mach number. It can be noted that, in the presence of exothermic reaction, significant viscous dissipation (associated with high-speed flow), and accelerating pressure gradients, there will be a nonmonotonic velocity variation across the layer. That is, a peak velocity occurs in the middle of the layer. This indicates the occurrence of large strain and vorticity (created by the cross product of pressure and density gradients). A decrease in peak temperature occurs with increasing downstream distance (see Fig. 5 in Ref. 3). This indicates the possibility of a reduction in NO_x formation. In extreme cases, extinction is threatened by the decreasing temperature.

IV. Challenges for the Turbine-Burner

High-speed combustion is itself a field in its infancy. The research on high-speed combustion has been motivated by hypersonic flight, and the emphasis has been on supersonic, steady mixing, and reacting flows.[4] Mixing and reaction in the turbine passages involve certain specific new challenges: 1) transonic speeds with the inherent complications of mixed subsonic and supersonic flows, 2) unsteady flow because of the intermittency of the rotating blades, and 3) the very large three-dimensional acceleration that can combine with the stratified mixture flowing through the turbine to produce potential hydrodynamic instability and large straining of the flow.

As a mixture of burned and unburned gases enters the turbine from the main combustor (in the practical configuration of a jet engine), a significant spatial variation in temperature, composition, and density can be expected. Because the stator vanes and the turbine blades turn the flow, large accelerations are experienced; 10^5 g in the azimuthal direction and 10^4 g in the radial direction are expected. These accelerations together with the density gradient can result in Rayleigh–Taylor instabilities.[5]

In certain configurations, one can expect that vortical cells or recirculation zones will result from these instabilities. These cells can enhance mixing and reaction in the turbine passages, but they may have a strong (possibly adverse) impact on the turbine aerodynamics and heat transfer. Models of the stratified, reacting flow with transverse acceleration forces are required. Because the radial acceleration is an order of magnitude less than the azimuthal acceleration, a two-dimensional analysis or experimental simulation would initially suffice.

Acceleration in the streamwise direction through the turbine flow passages (in a jet engine) can reach the order of 10^5 g. In reacting flow, we expect the transverse variation of the streamwise pressure gradient to be very modest compared with the transverse density variation. Therefore, the precise value of the streamwise acceleration can vary substantially in the transverse direction. Low-density regions will experience greater acceleration than higher-density regions. This directly implies that transverse derivatives of the velocity will be large, i.e., highly strained flows can result. The strained flows can have various implications. In certain configurations, the straining will enhance mixing and heat transfer by increasing interface areas. Another possibility is the widely varying velocities can result in widely varying residence times for different flow paths; this can produce flammability difficulties for regions with the shorter residence times. The transverse variation in velocity and in kinetic energy can cause variations in enthalpy and stagnation enthalpy that impact heat transfer. Note that the type of strain discussed here is different from types studied in counterflow flames. Two-dimensional finite difference, nonlinear methodologies can be employed. The results for the flow and thermal field calculations will allow the determination of the modifications of mixing, heat transfer, and flammability. Again, experimental research will provide a global observation of key phenomena and a quantitative comparison for the computations.

Turbomachinery flows are intrinsically unsteady. In fact, it is the essential mechanism for imparting or extracting work from any fluid device. Numerical methods for solving the multispecies Navier–Stokes equations can be used to calculate unsteady flows. Currently, a method has been demonstrated for time-accurate calculations involving oscillating airfoils and also cascade blades with the single-species Navier–Stokes equations and two-equation turbulence models.[6,7] This type of method can be extended for reacting multiple species to investigate the unsteady effect on combustion because of rotor–stator interactions, for example, how periodic wakes from upstream blade rows influence combustion and whether they will cause the accumulation of hot spots on blade surfaces.

The strong exothermic combustion processes will impose large changes to the flowfield, causing significant alteration of the turbine aerodynamics. For example, the thermal expansion caused by combustion could dramatically change the pressure distribution, the shock strength and location, and, thus, the overall loading of the blades. The energy released by combustion within the blade rows must be carefully calculated so that it can be absorbed and converted into turbine work by the turbine rotor to maintain the total temperature within limits for the blades. This means that the turbine stages must be loaded to their maximum capacity.

The turbomachinery designer would now face the challenges of 1) modified pressure and friction force distribution because of the combustion process, 2) increased local temperature and increased heat transfer rates, 3) the necessity to account for energy release and for species conservation in the computer codes used as design tools, and 4) aerodynamic impact of induced hydrodynamic instabilities. While there will be resistance to this development from some turbine engine designers, others will pursue this new technology. ABB Power Generation is already marketing a ground-based engine with a secondary combustor between the turbine stages. Our proposal goes further and suggests that the combustion occurs within the stator, and possibly within the rotor.

V. Concluding Remarks

A thermal analysis has shown the advantages of burning in the turbine passages of a jet engine. Increases in ST and decreases in specific fuel consumption can result. These advantages can translate to reductions in engine and fuel tank volume and weight and/or increases in range. Advantages for ground-based turbine engines are also demonstrated.

Combustion in the accelerating flow through the turbine passages presents obvious challenges concerning heat transfer to the walls and blades. A diffusion flame in the accelerating flow causes other problems. One is related to the large strain resulting from the large transverse density gradient; extinction then becomes a possibility. Another problem is related to the combination of the transverse pressure gradient and density gradient; hydrodynamic instability can result, creating vortices that could modify the aerodynamic loading on the blades. The vortices could, of course, improve the mixing of the reactants. The accelerating flow keeps the peak temperature and NO_x formation rates low.

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