Turbo/Air-Augmented Rocket: A Combined Cycle Propulsion System

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A new concept of combined cycle propulsion system is proposed. In the proposed system, a turbojet (or turbofan) type engine provides compressed air as part or the whole oxidizer to the rocket engine at takeoff and part of the acceleration and climb. At the same time, the turbo-engine provides a portion of the thrust. The system performance shows little sensitivity to rocket chamber pressure, shows comparable performance to turbojet, and is capable to shifting to pure rocket mode at any desired point in the flight trajectory.

Nomenclature

BR = by-pass ratio, the ratio of mass directed to the rocket to the incoming mass
E = efficiency
F = thrust
g = standard gravitational acceleration
I_{sp} = specific impulse
M = Mach Number
$\dot{m}$ = mass-flow rate
PR = pressure ratio
P = pressure
T = temperature
TSFC = thrust specific fuel consumption
W = power per unit mass rate

Subscripts
a = air
ab = afterburner
cc = combustion chamber
$C1$ = first compressor
$C2$ = second compressor
f = fuel
it = inlet to the turbine
n = nozzle
o = oxidizer
rk = rocket
t = turbine

I. Introduction

COMBINED cycle propulsion systems have recently become the focal point for the future generation of aerospace vehicles. In this context, a combined cycle engine is an integrated air-breathing/rocket engine with multimode operational capabilities of high acceleration, much higher specific impulse (an order of magnitude), and much lower thrust specific fuel consumption (TSFC) compared to liquid-fueled rocket engines. The main advantage of air-breathing engines over the rocket systems lies in the fact that air breathers use atmospheric air as the oxidizer for takeoff and atmospheric flight, whereas rocket systems carry the oxidizer along in the whole flight period. The main drawback of air breathers for aceleration missions is their higher weight per unit thrust. Combined cycle propulsion engines have been suggested as the propulsion system for fully reusable, autonomous, economical, and flexible mission profile future aerospace vehicles, both for transatmospheric and Earth-to-orbit missions. Various aspects of combined cycles with different levels of complexity have been suggested, discussed, and some tested, Escher et al.,[1] Kors,[2] and Strack.[3] Some of the latest research and development efforts on combined cycle propulsion systems have been reviewed by Ganji et al.[4]

In this paper, a new concept of combined cycle propulsion systems is proposed. In the proposed concept, shown in Fig. 1, a turbojet (or turbofan) type engine provides compressed air as part or the whole oxidizer to rocket engine at takeoff and part of the acceleration and climb stage. At the same time, the turbo-engine provides a portion of the thrust. The system will be discussed and analyzed in Sec. II, and sample results of analysis will be presented and discussed in Sec. III.

II. Analysis of the System

A schematic diagram of the proposed combined cycle is shown in Fig. 1. The system consists of turbo-engine and rocket subsystems. The turbo-engine (turbojet or turbofan) provides the power to the second compressor for supplying air to the rocket as the whole or part of its oxidizer. The rocket may have the option of using stored oxidizer for boosting the thrust in atmosphere and also for space flight. The turbo-engine also has the option of using afterburner or duct burning (in the case of turbofan) for thrust augmentation. The system will have the capability of operation in the air-breathing mode, the rocket mode, or the mixed mode. In the mixed mode (both air-breather and rocket-operating), the rocket pressure should always be slightly below the exit pres-

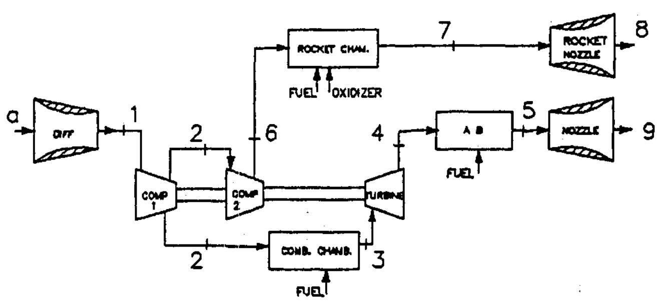

Fig. 1 Schematic diagram of the turbo/air-augmented rocket (TAAR).

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sure of the second-stage compressor. This pressure will vary along the flight path, and consequently a variable throat area nozzle will be required for optimum operation. Design of variable area nozzles is still a challenge for propulsion engineers.

In the analysis of the system, the ambient temperature and pressure (T_a, P_a), flight Mach number (M_a), efficiencies of the diffuser (E_d), first- and second-stage compressors (E_{C1}, E_{C2}), combustion chamber (E_{cc}), pressure ratio of the first- and second-stage compressor (PR_{C1}, PR_{C2}), turbine inlet temperature (T_{it}), rocket chamber temperature (T_{ch}), and the mass ratio by-passed (BR) to the rocket are independent variables. The performance parameters of interest are thrust per unit of airflow rate (F/m_a), specific impulse (I_{sp}), and TSFC. Figure 2 shows a T-S diagram of the processes in the system. In the design point analysis of this system, the basic methodology of Hill and Peterson[5] and Oates[6] has been used. Usually in this type of analysis, the specific heats are assumed to be constant and the working fluid is assumed to have the composition of air. These assumptions have not been made in the present analysis. The analysis has been made through a comprehensive computer program developed for this purpose. For further detail refer to Ganji et al.[4]

There is a limitation on the amount of air that can be compressed into the desired rocket chamber pressure. An estimate of this limitation can be obtained by assuming full expansion to ambient pressure in the turbine. This means that the turbine section of the engine will not be able to produce any thrust. To calculate this limit we have

$$W_{C1} + BR \cdot W_{C2} = (1 - BR) \cdot W_t$$

where W represents the power. The BR can be calculated from this equation:

$$BR = (W_t - W_{C2}) / (W_t + W_{C2})$$

In this relation, W_t is calculated by letting the working substance in the turbine expand to the local ambient pressure.

The performance parameters presented in the results are as follows:

$$\mathrm{TSFC} = (m_{f,cc} + m_{f,rk} + m_{o,rk} + m_{f,ab})/F$$

$$I_{sp} = F/g\,(m_{f,cc} + m_{f,rk} + m_{o,rk} + m_{f,ab})$$

where $F = F_t + F_{rk}$ and I_{sp} introduced here is the impulse per unit mass of propellant, excluding the air. For the turbo-engine, the fuel used in the calculations is assumed to have chemical formula similar to octane with the same heating value. The rocket fuel has been considered to be hydrogen, except when specifically pointed out to be hydrocarbon, in which case it will be octane. Throughout the analysis, it has been assumed that the expansion in the nozzles will be to

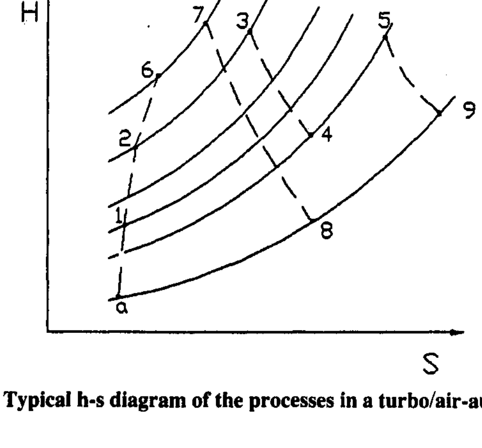

Fig. 2 Typical h-s diagram of the processes in a turbo/air-augmented rocket.

ambient pressure, although the developed program could handle the case of specified area ratio nozzle and consequently over- or underexpansion in the nozzle. No stored oxidizer has been used in this analysis.

III. Results and Discussion

Performance of propulsion systems are usually demonstrated through specific thrust (T/m_a, thrust per unit air mass flow rate for air-breathing systems), specific impulse (I_{sp}, thrust per unit propellant flow rate excluding the air), and thrust specific fuel consumption (TSFC, mass of consumed fuel per unit thrust). The behavior of these performance parameters for turbo/air-augmented rocket (TAAR) is briefly reviewed here.

Figure 3 shows the specific thrust versus the first-stage com-

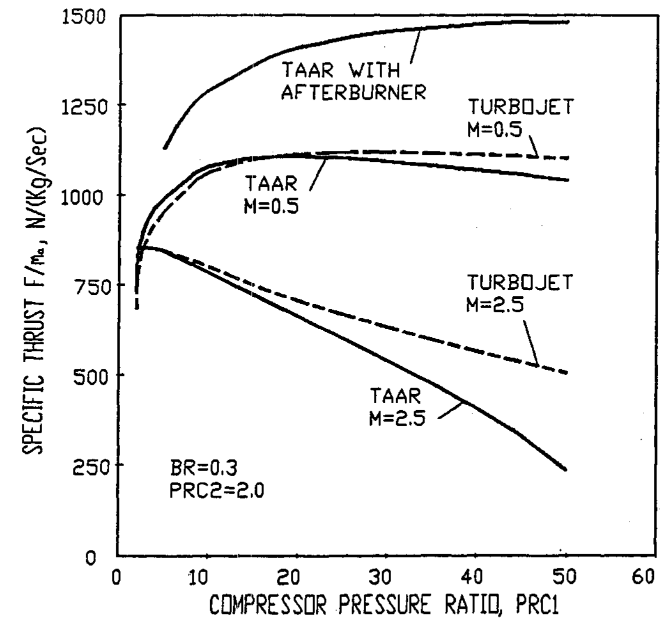

Fig. 3 Variation of the specific thrust (F/ma) vs the pressure ratio of the first-stage compressor (PR_{C1}) for a TAAR without afterburner, with afterburner, and for a simple turbojet. In this analysis, E_d = 0.9, E_{C1} = E_{C2} = 0.9, E_t = 0.9, E_n = 0.9, T_{it} = 2000 K, and T_{ab} = T_{ch} = 2500 K.

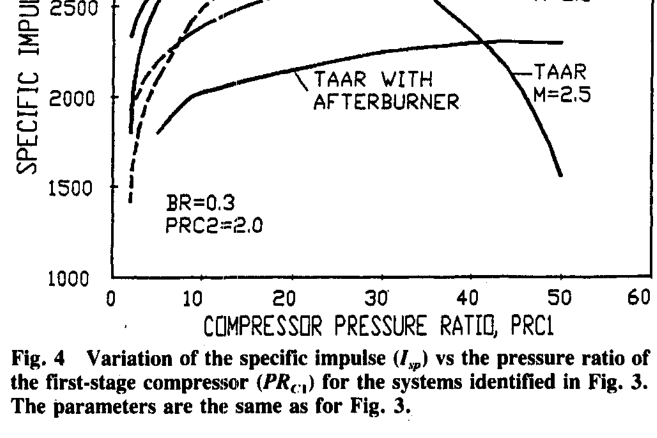

Fig. 4 Variation of the specific impulse (I_{sp}) vs the pressure ratio of the first-stage compressor (PR_{C1}) for the systems identified in Fig. 3. The parameters are the same as for Fig. 3.

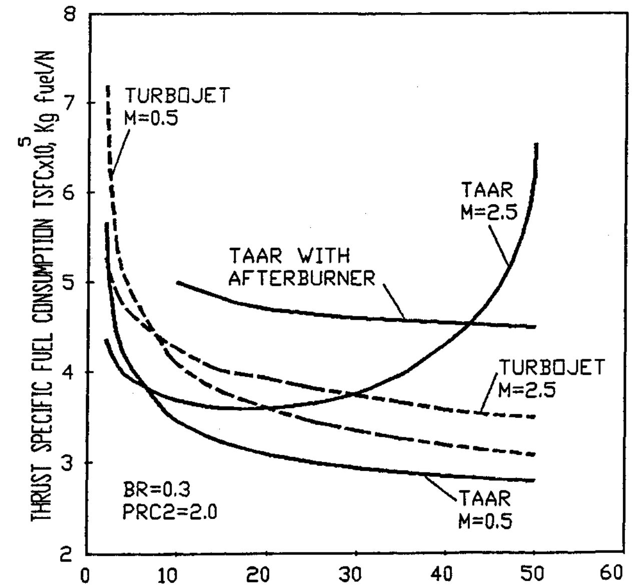

Fig. 5 Variation of the TSFC vs the pressure ratio of the first-stage compressor (PR_{C1}) for the systems identified in Fig. 3. The parameters are the same as for Fig. 3.

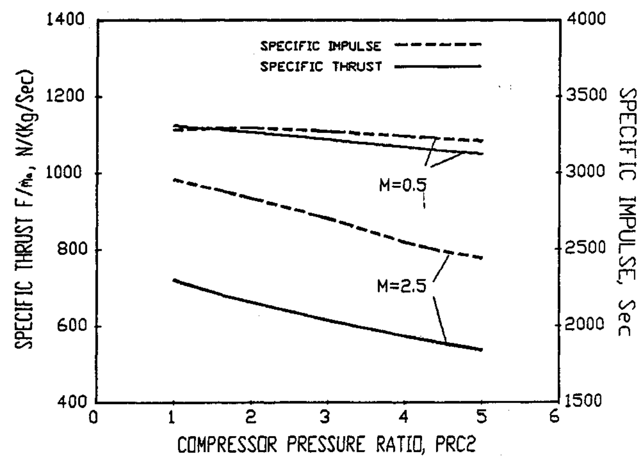

Fig. 6 Variation of specific thrust (F/m_a) and specific impulse (I_{sp}) of TAAR vs the second-stage pressure ratio for the first-stage compressor pressure ratio of PR_{C1} = 20. The rest of the parameters are the same as for Fig. 3.

pressor pressure ratio for two values of flight Mach number of TAAR, TAAR with afterburner, and simple turbojet for reference. Turbojet design parameters are the same as TAAR, excluding the second-stage compressor and rocket system. This figure shows that for TAAR for the given parameters, the specific thrust does not show a major difference with respect to a simple turbojet, especially at low pressure ratios. It is also seen that the trends are similar for typical subsonic and supersonic conditions. The properties of inlet are those at the elevation for constant dynamic pressure trajectory of 1000 psf throughout these calculations. Figure 4 shows the specific impulse versus the first-stage pressure ratio for the same systems. For subsonic conditions, TAAR has superior performance at all values of first-stage compressor pressure ratios. For supersonic speeds, TAAR shows superior performance at lower pressure ratios. In combination with Fig. 3, Fig. 4 shows that lower first-stage compressor pressure ratios will be more advantageous for this type of engine. The effect of adding an afterburner on TAAR is lowering the I_{sp} and increasing specific thrust as espected. The Mach number for afterburning condition is 0.5. TSFC of this system has a behavior that is the inverse of I_{sp}. Variation of TSFC with respect to the first-stage pressure ratio is shown in Fig. 5.

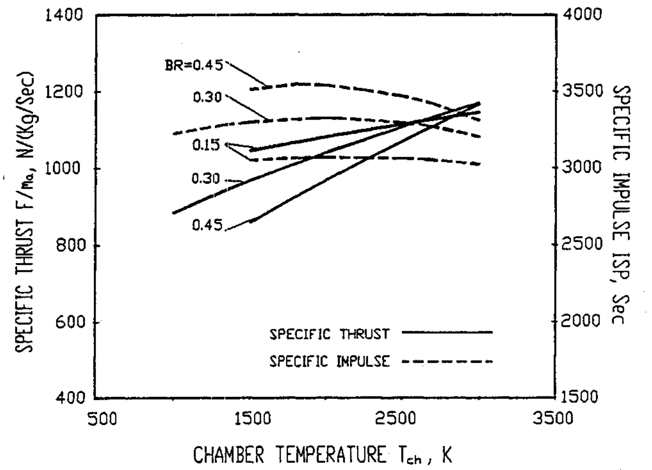

Fig. 7 Variation of specific thrust (F/m_a) and specific impulse (I_{sp}) of TAAR vs the rocket chamber temperature for three values of mass ratios by-passed to rocket (BR). PR_{C1} = 20, PR_{C2} = 2, and the rest of the parameters are the same as for Fig. 3.

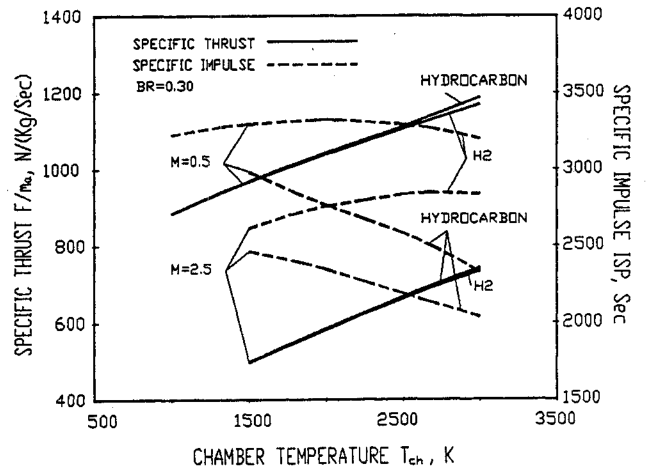

Fig. 8 Variation of specific thrust (F/m_a) and specific impulse (I_{sp}) of TAAR vs the rocket chamber temperature for hydrogen and hydrocarbon fuels at two values of flight Mach number. PR_{C1} = 20, PR_{C2} = 2.0, and the rest of the parameters are the same as for Fig. 3.

Figure 6 shows that the performance of TAAR is more sensitive to the second-stage compressor pressure ratio for supersonic flight. The overall effect is lowering the performance. As shown in Fig. 7., rocket chamber temperature does not have a pronounced effect on the specific impulse of the TAAR, whereas the by-pass ratio can moderately affect the I_{sp}. On the other hand, for temperature range of interest (below 2500 K), increase in mass by-pass ratio results in moderate reduction in specific thrust. For the design parameters under consideration, a by-pass ratio of about 30% seems to be a compromise.

Figure 8 shows the effect of using hydrocarbon fuel (octane in this case) instead of hydrogen in the rocket chamber for typical subsonic and supersonic conditions. While the type of fuel will not affect the specific thrust, it will considerably affect the specific impulse. Using hydrogen shows its clear advantage in this respect over the hydrocarbon as the fuel for the rocket. Figure 8 shows that an increase in rocket chamber temperature will result in reduction of I_{sp} for hydrocarbon fuels. For hydrogen it reaches a maximum value and slightly decreases at very high temperatures. This is because of much larger heating value of hydrogen compared to hydrocarbons.

Variation of I_{sp} along a typical flight trajectory of 1000 psf for TAAR is shown in Fig. 9. Performance of a typical turbojet with the compressor pressure ratio the same as the TAAR first-stage compressor and similar parameters for the

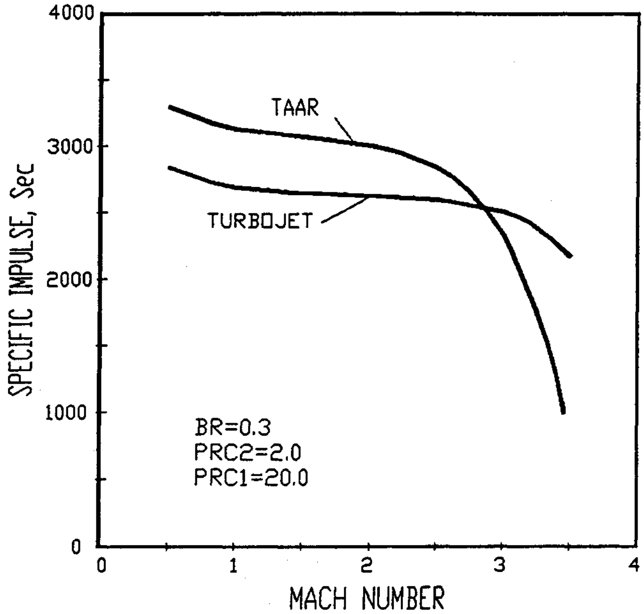

Fig. 9 Variation of specific impulse (I_{sp}) along a typical Earth-to-orbit trajectory for TAAR and turbojet. For turbojet and TAAR, PR_{C1} = 20, and the rest of the parameters are the same as for Fig. 3.

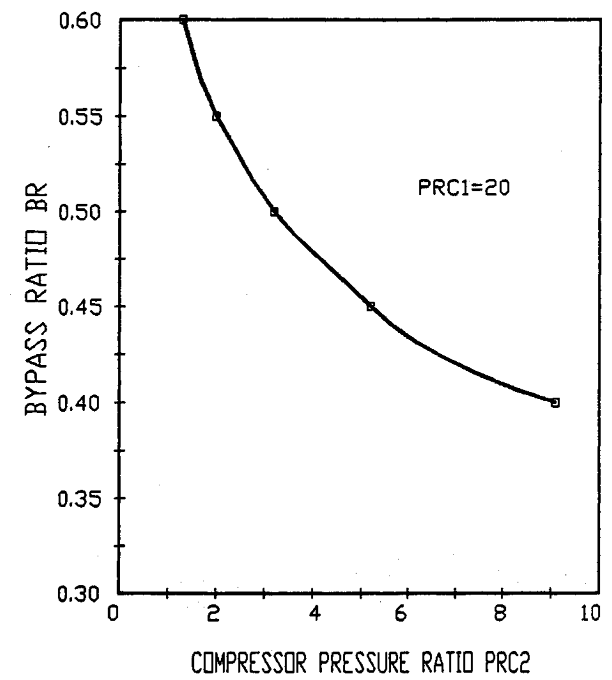

Fig. 10 Variation of limiting value of mass ratio by-passed (*BR*) to the rocket vs the second-stage compressor pressure ratio in TAAR. PR_{C1} = 20 and the rest of the parameters are the same as for Fig. 3

diffuser, combustion chamber, and nozzle is also shown. This figure shows the clear advantage of TAAR over a turbojet for subsonic and low supersonic where turbo-engines have an advantage over the other engine types.

It was shown in Sec. II that there exists a limit on the amount of air that can be by-passed to the rocket chamber. Figure 10 shows this limit with respect to the second-stage compressor pressure ratio, which represents the rocket chamber pressure. The system will usually operate at a mass by-pass ratio of a lesser value than those shown. In other words, the region of the plane below the curve is the region in which the system will be able to operate.

The results discussed in this paper are some preliminary aerothermal performance data of this new concept of combined cycle. This preliminary investigation shows some interesting features of this system that makes it comparable and in cases superior to turbojet. No attempt has been made at this stage to optimize the TAAR or the turbojet that has been used as a baseline for comparison.

The main advantage of this type of engine over a turbojet will be its dual functionality, which makes it suitable for both takeoff to midsupersonic (Mach number of approximately 3) and space flight. For midsupersonic to the high hypersonic range, turbo-engines are not suitable and sequential use of ramjet and scramjet engines have been recommended. By using H_2 for the rocket chamber fuel as shown in Figs. 3, 4, 5, and 9, the TAAR will have a comparable or superior performance to that of a turbojet engine. In all the calculations for comparison of turbojet and TAAR, the compressor pressure ratio (in the case of TAAR for the first stage), turbine inlet temperature, and component performance parameters have been taken to be the same. The turbine inlet temperature of 2000 K is for advanced, state-of-the-art, and future technology turbines, see Bahr.[7] The afterburner temperature of 2500 K is also a feasible temperature with hydrocarbon fuels. The possibility of using hydrocarbon fuels for the rocket was also studied. Selected results are shown in Fig. 8. Because hydrogen is a preferred fuel for various stages of Earth-to-orbit flight, especially for the rocket and scramjet, probably additional complexity of a hydrocarbon fuel system for the rocket will not be justified. Another advantage of hydrogen fuel that can be used in TAAR is its heat absorbing capacity in a cryogenic intercooler between the two stages of the compressor. This will certainly improve the performance of the system. The authors are working on the effect of intercooling on the performance of TAAR at the present time.

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References

[1]Escher, W. J. D., Teeter, R. R., and Rice, E. E., "Airbreathing and Rocket Propulsion Synergism: Enabling Measure for Tomorrow's Orbital Transports," AIAA Paper 86-1680, June 1986.

[2]Kors, D. L., "Combined Cycle Propulsion for Hypersonic Flight," Paper IAF-87-263, 39th Congress of the International Astronautical Federation, Brighton, UK, Oct. 10–17, 1987.

[3]Strack, W. C., "Aeropropulsion Opportunities for the 21st Century," NASA TM-88817, 1986.

[4]Ganji, A. R., Khadem, M., and Khandani, S. M. H., "Combined Cycle Propulsion Systems Analysis (CCPSA) Computer Program," Phase I Report, Contract #F33615-88-C-286, ASD, U.S. Air Force, Dec. 1989.

[5]Hill, P. G., and Peterson, C. R., *Mechanics and Thermodynamics of Propulsion*, Addison-Wesley, Reading, MA, 1965.

[6]Oates, G. C., *Aerothermodynamics of Gas Turbines and Rocket Propulsion*, AIAA Education Series, AIAA, 1984.

[7]Bahr, D. W., "Technology for the Design of High Temperature Rise Combustors," *Journal of Propulsion*, Vol. 3, No. 2, 1987, pp. 179–186.