RI/RD78-215

(NASA-CP-15008) TRIPROPELLANT ENGINE STUDY Final Report, Aug. 1977 - May 1978 (Rocketdyne) 146 p HC A07/MF A01 CSCL 21H

N78-32181

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NASA

FINAL REPORT

TRIPROPELLANT ENGINE STUDY

by

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Rockwell International

Rocketdyne Division

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

SEPTEMBER 1978

George C. Marshall Space Flight Center

Contract NAS8-32613

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle	5. Report Date
TRIPROPELLANT ENGINE STUDY Final Report	September 1978
	6. Performing Organization Code
7. Author(s) D. B. Wheeler and F. M. Kirby	8. Performing Organization Report No. RI/RD78-215
9. Performing Organization Name and Address Rocketdyne Division Rockwell International 6633 Canoga Avenue Canoga Park, CA 91304	10. Work Unit No. 11. Contract or Grant No. NAS8-32613
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration George C. Marshal Space Flight Center Marshal Space Flight Center, AL 35812	13. Type of Report and Period Covered Final (August 1977-May 1978) 14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

This study examines the potential for converting the Space Shuttle Main Engine (SSME) to a dual-fuel, dual-mode engine using LOX/hydrocarbon propellants in mode 1 and LOX/H_2 in mode 2. Various engine system concepts were formulated that included staged combustion and gas generator turbine power cyles, and LOX/RP-1, LOX/CH_4, and LOX/C_3H_8 mode 1 propellants. Both oxidizer and fuel regenerative cooling were considered. All of the SSME major components were examined to determine their adaptability to the candidate dual-fuel engines. Technology demonstration programs are recommended in those areas that this study determined to be critical.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Dual Fuel Dual Mode Staged Combustion Gas Generator Turbine Power Cycles High Chamber Pressure LOX/Hydrocarbon Propellants	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	139	

* For sale by the National Technical Information Service, Springfield, Virginia 22151

NASA-C-168 (Rev. 6-71)

FOREWORD

The Tripropellant Engine Study was conducted at the Rocketdyne Division of Rockwell International under NASA Contract NAS8-32613. Mr. Dale Blount, NASA-Marshall Space Flight Center, was the study Project Manager. At Rocketdyne, Mr. Frank M. Kirby was study Program Manager and Mr. David B. Wheeler was the Project Engineer.

The study effort was supported by the following team of technical specialists at Rocketdyne:

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INTRODUCTION

The advanced vehicle studies that have been conducted for NASA indicate the advantages of a high-pressure oxygen/hydrocarbon engine. Single-stage-to-orbit vehicle studies also show the potential for engines that operate in dual mode with sequential burn of oxygen/hydrocarbon and oxygen/hydrogen. Feasibility of an engine to operate in dual mode must be determined before committing to a dual-mode vehicle concept.

The Space Shuttle Main Engine (SSME) is a high-pressure oxygen/hydrogen engine that potentially could be modified for a dual-mode operation. Such a modification would minimize development cost of a dual-mode engine by maximizing utilization of existing hardware.

The objectives of this study program are to: (1) investigate the feasibility of a tripropellant engine operating at high chamber pressure; (2) identify the potential applicability of SSME components in the duel-mode engine; (3) define engine performance and engine concepts for both gas generator and staged combustion power cycles; and (4) provide plans for experimental demonstration of the performance, cooling, and preburner or gas generator operation.

The approach taken in this study was to investigate various high P_c engine configurations derived from the SSME that will allow sequential burning of LOX/hydrocarbon and LOX/hydrogen. Both staged combustion and gas generator pump power cycles were considered. Engine cycle concepts are formulated for LOX/RP-1, LOX/CH_4, and LOX/C_3H_8 propellants. Each system also must be capable of operating sequentially with LOX/H_2. Flowrates and operating conditions were established for this set of engine systems and the adaptability of the major components of the SSME were investigated.

Control systems for dual-mode operation were established and SSME control system components were evaluated for their applicability. The final objective was to identify high chamber pressure engine system concepts that make maximum use of SSME hardware and best satisfy the dual-mode booster engine system application.

Based on the results of the engine system concepts and SSME component adaptability studies, recommendations were made for additional testing to complement the already planned experimental program using the existing test facility and 40K test hardware.

SUMMARY

The results of these studies have shown that the conversion of an SSME engine to a high chamber pressure, dual-mode fuel engine will require major modifications to the hardware and/or the addition of a significant number of new engine components. However, the study has shown numerous possibilities for the use of SSME hardware derivatives in a single-mode LOX/hydrocarbon engines. It was also shown that a reduced chamber pressure version of a staged combustion SSME is operationally feasible using the existing fuel-rich preburners and main chamber injectors. Certain turbomachinery modifications or additions are required for a total low chamber pressure (2300 psia) engine system. This study also has shown that the engine system concepts applicable to the dual-mode systems are somewhat narrowed since the operational constraints of two systems must be considered.

Some general conclusions were established that would be useful in future single-fuel or dual-fuel LOX/hydrocarbon engine systems:

1. There is insufficient energy (fuel) available to obtain a staged combustion power balance with all preburners fuel rich at a chamber pressure of 3230 psia and a turbine inlet temperature of 2000 R. Staged combustion power balances can be achieved for all fuel rich preburners at either reduced chamber pressure or increased turbine inlet temperatures.

2. A LOX/hydrocarbon staged combustion power balance is achievable with both preburners LOX rich or LOX-rich LOX turbine and fuel-rich fuel turbine.

3. There is insufficient energy (LOX) available to obtain staged combustion power balance with all preburners LOX rich for LOX/H_2 mode 2 operation.

4. Gas generator cycles are not power limited. If H_2 is employed to cool either the chamber or nozzle, the required flow is generally adequate for fuel flow to the gas generator. This also provides minimum turbine drive performance loss.

5. Higher chamber pressures are achievable with gas generator cycles than with staged combustion cycles with equal pump discharge pressures.

6. Regenerative cooling with RP-1 limits maximum chamber pressure to 2000 psia.

7. LH_2 regenerative cooling gives minimum coolant flowrate, ΔP, and does not require switching coolants from mode 1 to mode 2.

8. LOX regenerative cooling limits LOX/H_2 mode 2 operation to a maximum chamber pressure of 2000 to 2500 psia due to high pressure drop.

9. CH_4 was found to be the best hydrocarbon evaluated for a regenerative coolant.

The above-stated conclusions are some of the more significant results of this study. A general comparison of the important features of the engine cycles and propellant combinations and their impact on the major engine components and systems are shown in Table 1.

Initially, the objective of the test planning task of this study was to identify critical areas for experimental verification of the adaptability of SSME engine components to the dual-fuel engines. Since the study results indicated limited use of SSME engine components in a dual-fuel engine, a revised objective was established to identify general technology items that arose during the study and that require either technology demonstration or development. The results of the proposed test plans will provide information important to all new dual-mode tri-propellant or single-mode LOX/hydrocarbon booster engines.

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TABLE 1. TRIPROPELLANT DUAL-FUEL ENGINE SYSTEM CONFIGURATION COMPARISON

SYSTEM OPTION	CHAMBER	INJECTOR	TURBINES	PUMPS	PREBURNER/GG	POWER CYCLE	PERFORMANCE
O_2/RP 1 SYSTEMS STAGED COMBUSTION		LARGE FUEL DENSITY DIFFERENCE BETWEEN MODES INJECTOR DESIGN COMPROMISES REQ'D	REQUIRES AT LEAST ONE OX RICH PREBURNER	$\boxed{\text{DIFFERENT FUEL TURBOPUMP REQ'D FOR MODE 1 \& MODE 2 IN EVERY CASE}}$	REQ LIQ/LIQ INJECTION	REQUIRES MIXED LOX RICH AND FUEL RICH PREBURNERS FOR HIGH P_c. ALL LOX RICH PREBURNERS NOT FEASIBLE IN MODE 2. ALL FUEL RICH PREBURNERS REQUIRES REDUCED P_c OR INCREASED TURBINE INLET TEMP	(RELATIVELY POOR PERFORMANCE) I_s S/L = 310 AT P_c = 3220
O_2 COOLED	P_c LIMITED IN MODE 2 REQUIRES NO COOLANT SWITCHING	ADDS HEAT & AVOIDS FORMING LOX/RP GEL					
RP 1 COOLED	P_c LIMITED IN MODE 1 POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM		POSSIBLE COKING PROBLEM		
H_2 COOLED	NO COOLANT SWITCHING REQUIRED	MUST ACCOMMODATE H_2 INJECTION	REQUIRES SEPARATE TURBINE FOR H_2 COOLANT IN MODE 1	REQUIRES SEPARATE PUMP FOR H_2 COOLANT	REQUIRES UNHEATED LIQUID/LIQUID INJECTOR		I_s S/L = 304 AT P_c = 3220
GAS GENERATOR			ALL TURBINES OPERATE FUEL RICH			G.G. CYCLES NOT POWER LIMITED. ALLOWS HIGHER P_c WITH PUMP DISCHARGE PRESSURE EQUAL TO STAGED COMBUSTION	I_s S/L = 301 AT P_c = 3220
O_2 COOLED	P_c LIMITED IN MODE 2 NO COOLANT SWITCHING REQUIRED						
RP 1 COOLED	P_c LIMITED IN MODE 1 POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM		POSSIBLE COKING PROBLEM		
H_2 COOLED	NO COOLANT SWITCHING REQUIRED. COOLANT FLOW ADEQUATE FOR G.G. FUEL REQUIREMENTS		REQUIRES SEPARATE TURBINE FOR H_2. REQ NO SWITCHING OF TURB DRIVE FLUIDS	REQUIRES SEPARATE PUMP FOR H_2 COOLANT	REQUIRES UNHEATED LIQ/LIQ INJECTION		
O_2/CH_4 SYSTEMS		SMALLEST FUEL DENSITY DIFFERENCE BETWEEN MODES - MINIMIZES INJECTION DESIGN PROBLEMS					
STAGED COMBUSTION			REQUIRES AT LEAST ONE OX RICH PREBURNER	HIGH PUMP DISCHARGE PRESSURES	REQ LIQ/LIQ INJECTION	REQUIRES MIXED LOX RICH AND FUEL RICH PREBURNERS FOR HIGH P_c. ALL LOX RICH PREBURNERS NOT FEASIBLE IN MODE 2. ALL FUEL RICH PREBURNERS REQUI[illegible] REDUCED P_c OR INC[illegible]D TURBINE INLET TEM.	
O_2 COOLED	P_c LIMITED IN MODE 2						
CH_4 COOLED							I_s S/L = 320 AT P_c = 3220
GAS GENERATOR			ALL TURBINES OPERATE FUEL RICH		REQ LIQ/LIQ INJECTION	G.G. CYCLES NOT POWER LIMITED. ALLOWS HIGHER P_c WITH PUMP DISCHARGE PRESSURE EQUAL TO STAGED COMBUSTION	
O_2 COOLED	P_c LIMITED IN MODE 2						
CH_4 COOLED							I_s S/L = 310 AT P_c = 3220
O_2/C_3H_8 SYSTEMS		LARGE FUEL DENSITY DIFFERENCE BETWEEN MODES					
STAGED COMBUSTION			REQUIRES AT LEAST ONE OX RICH PREBURNER	HIGH PUMP DISCHARGE PRESSURES	REQ LIQ/LIQ INJECTION	REQUIRES MIXED LOX RICH AND FUEL RICH PREBURNERS FOR HIGH P_c. ALL LOX RICH PREBURNERS NOT FEASIBLE IN MODE 2. ALL FUEL RICH PREBURNERS REQUIRES REDUCED P_c OR INCREASED TURBINE INLET TEMP	
O_2 COOLED	P_c LIMITED IN MODE 2						
C_3H_8 COOLED	POSSIBLE COKING PROBLEM P_c LIMITED IN MODE 1	POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM		POSSIBLE COKING PROBLEM		I_s S/L = 320 AT P_c = 3220
GAS GENERATOR			ALL TURBINES OPERATE FUEL RICH			G.G. CYCLES NOT POWER LIMITED. ALLOWS HIGHER P_c WITH PUMP DISCHARGE PRESSURE EQUAL TO STAGED COMBUSTION	
O_2 COOLED							
C_3H_8 COOLED	POSSIBLE COKING PROBLEM P_c LIMITED IN MODE 1	POSSIBLE COKING PROBLEM	POSSIBLE COKING PROBLEM		POSSIBLE COKING PROBLEM		I_s S/L = 300 AT P_c = 3220

TASK I - PERFORMANCE DETERMINATION

The purpose of this task was to generate propellant performance data, combustion gas thermodynamic properties, and turbine drive gas parameters as required to support the other tasks. Main chamber theoretical performance predictions for the various mode 1 propellant combinations were predicted based on the JANNAF ODE program which yields theoretical I_s and c*. Delivered performance is calculated based on assumed efficiencies of $\eta_{c*} = 0.98$ and $\eta_{CF} = 0.9859$ applied to theoretical performance for the LOX/hydrocarbon combinations and $\eta_{c*} = 0.9975$ and $\eta_{CF} = 0.9859$ for LOX/H_2. Parametric curves of theoretical sea level and vacuum specific impulse as a function of mixture ratio, chamber pressure, and area ratio are shown in Fig. 1 through 6 for the three mode 1 propellant combinations. A table of the predicted theoretical performance for sea level and vacuum operation is presented in Table 2 for the four propellant combinations of interest. The mixture ratio was selected based on peak I_s. Fuel densities are also presented as an indicator of relative system volumes.

A preliminary study was conducted to verify the validity of mass averaging the specific impulse when H_2 is injected into the main chamber along with the LOX/hydrocarbon. Both vacuum and sea level specific impulse values were considered. I_s values computed by mass flow averaging were compared with theoretical (ODE) results for the O_2/RP/H_2 propellant system. Figure 7 presents the results of this comparison.

Sea level I_s values computed by mass averaging are generally quite close to the theoretical values, usually within 1.5 seconds. Vacuum results have a somewhat greater spread.

In general, the mass-averaged I_s values are sufficiently close to the theoretical values to permit their use in system definition studies.

Several mechanisms have been suggested to explain the difference in I_s computed by mass averaging and theoretical values. Mass averaged I_s values are lower than theoretical. This may be due to exothermic reactions that occur in the combustion

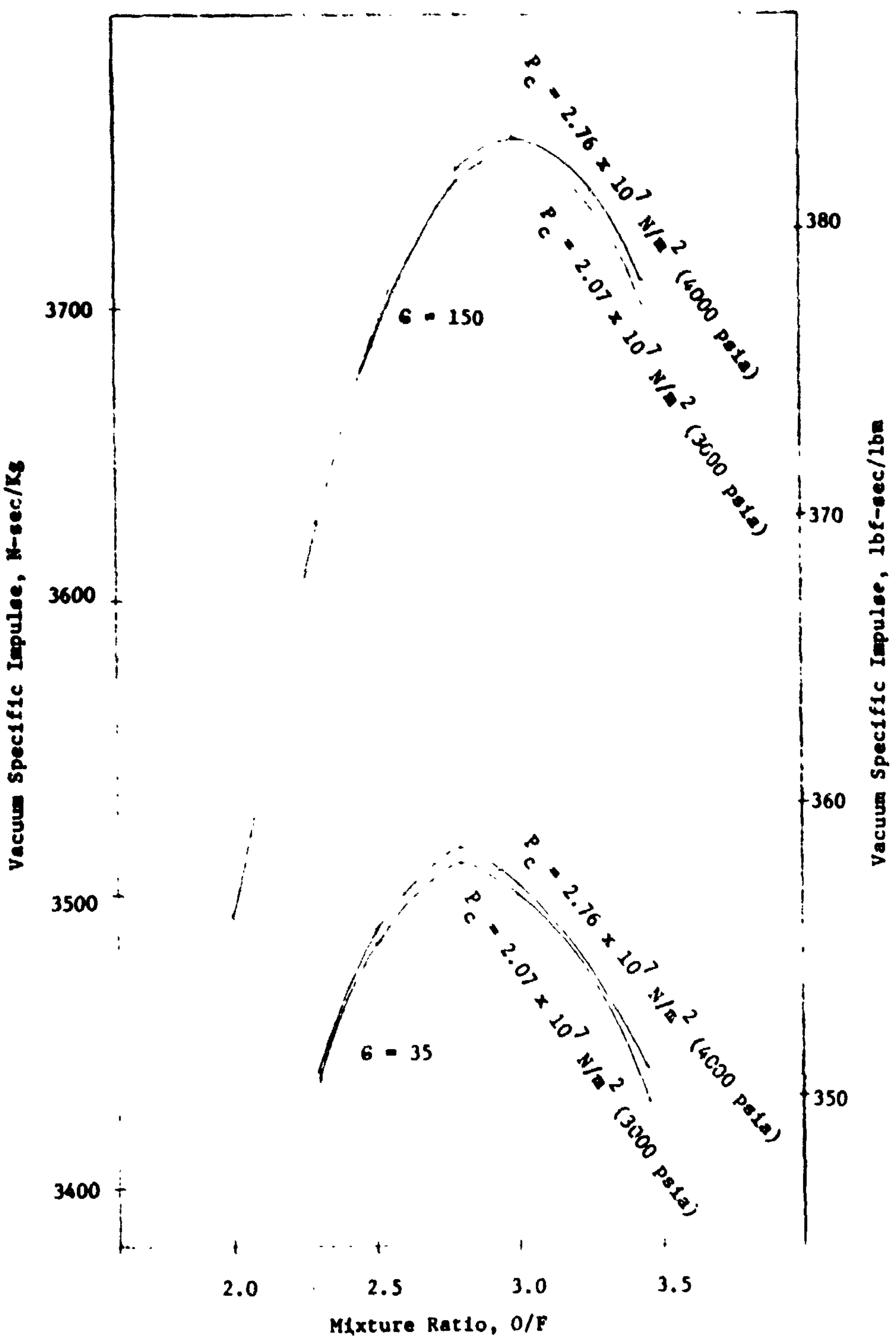

Figure 1. Theoretical Vacuum Performance for Oxygen/RP-1

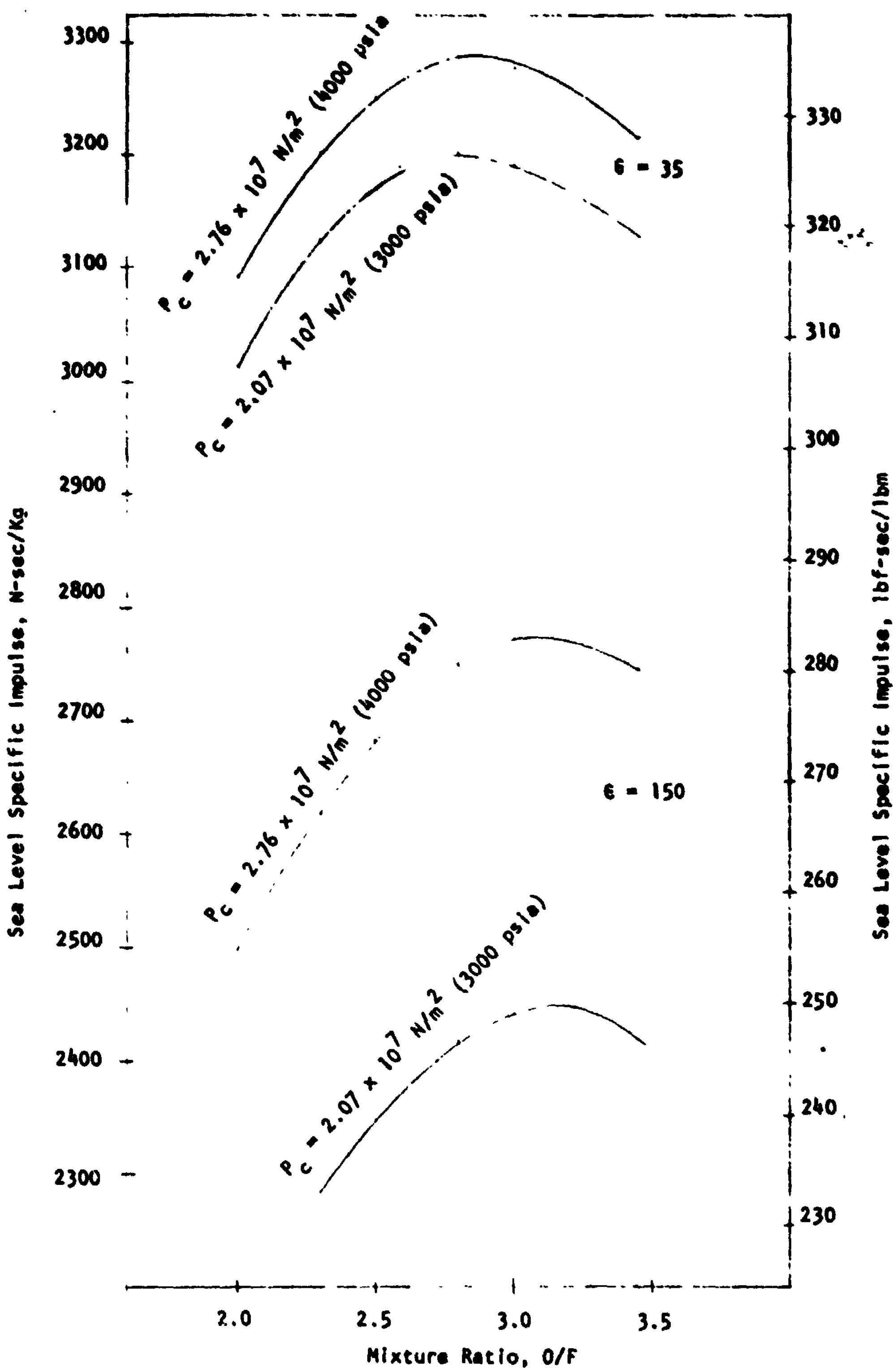

Figure 2. Theoretical Sea Level Performance for Oxygen/RP-1

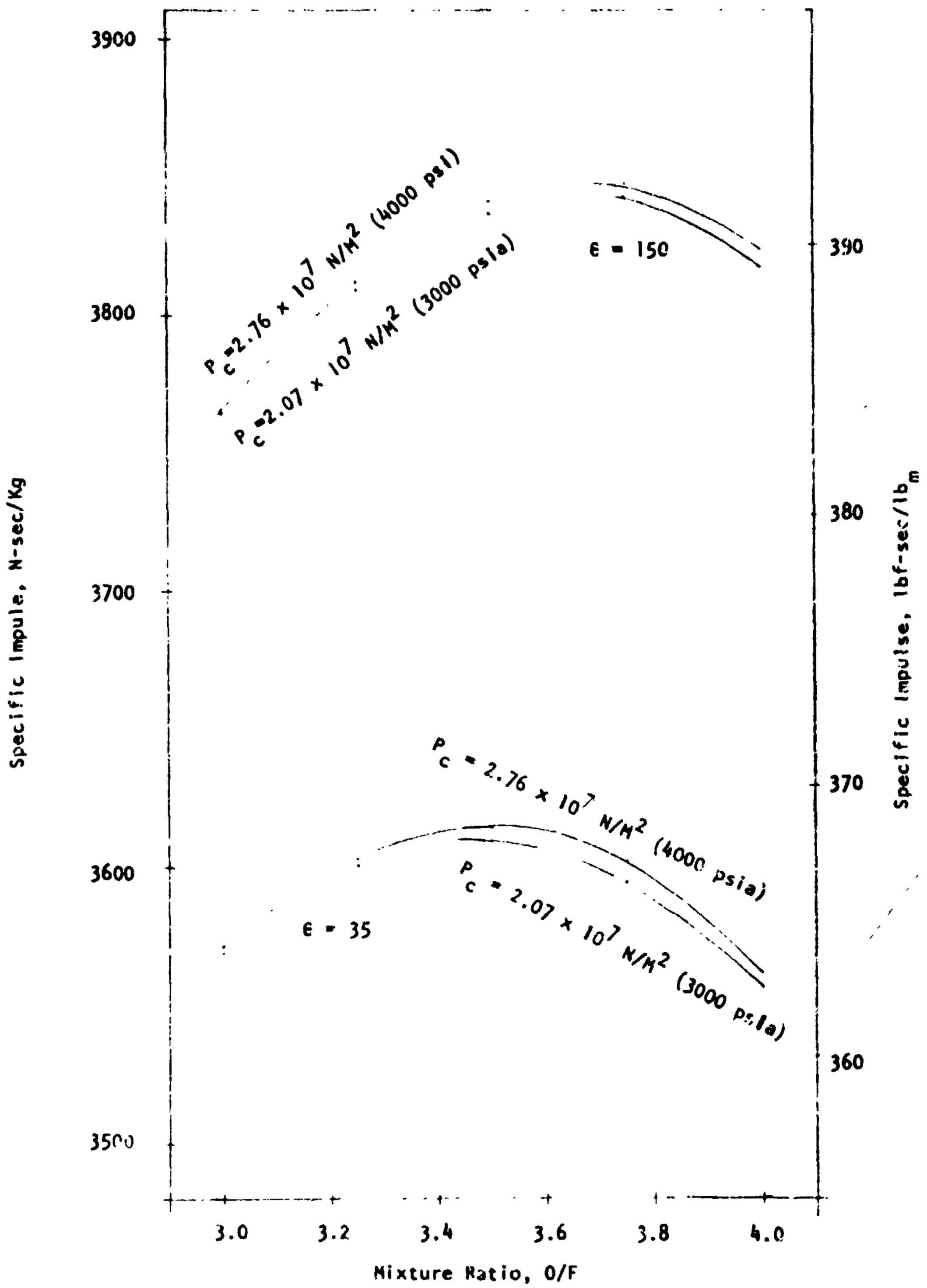

Figure 3. Theoretical Vacuum Performance for O_2/Methane (O_2/CH_4)

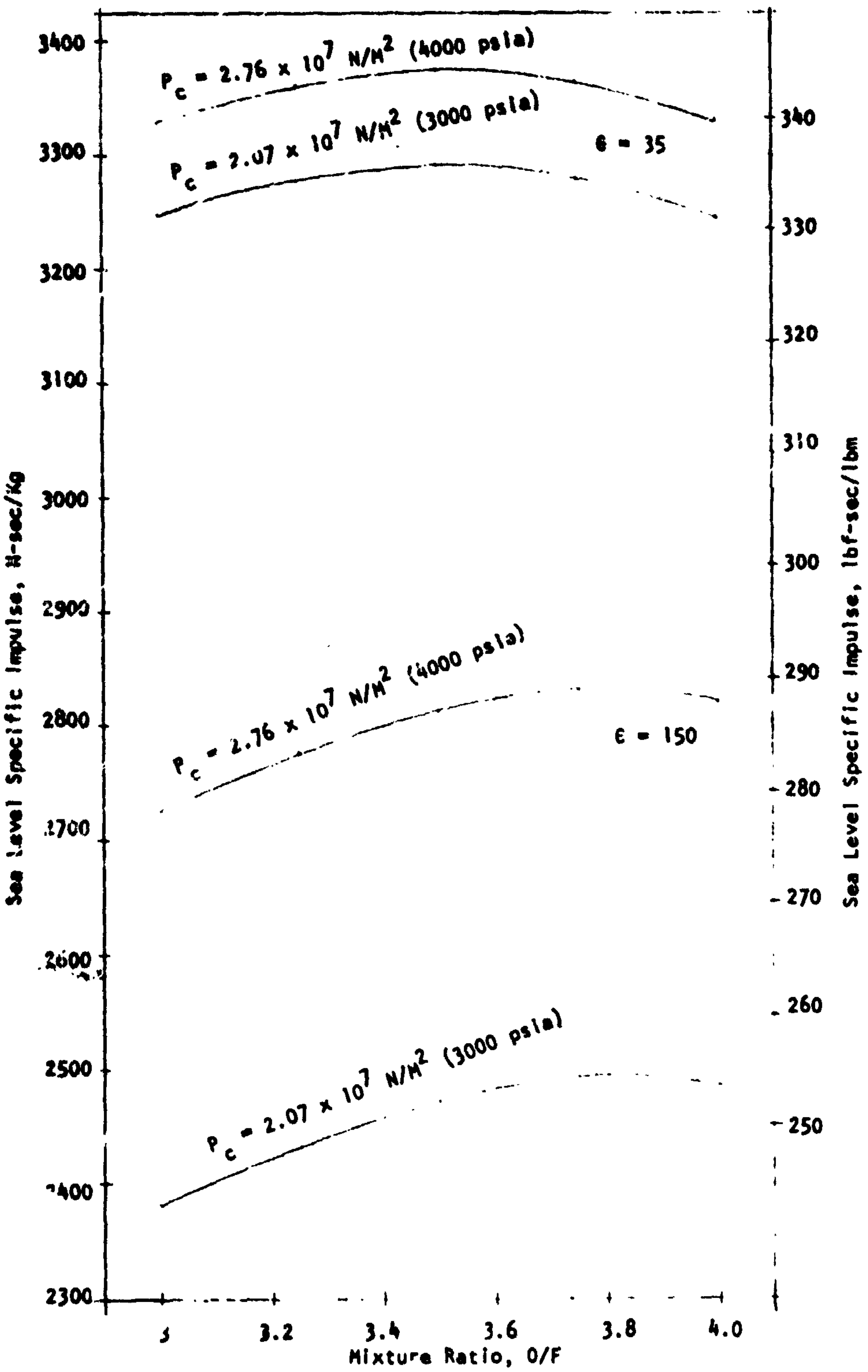

Figure 4. Theoretical Sea Level Performance for O_2/Methane (O_2/CH_4)

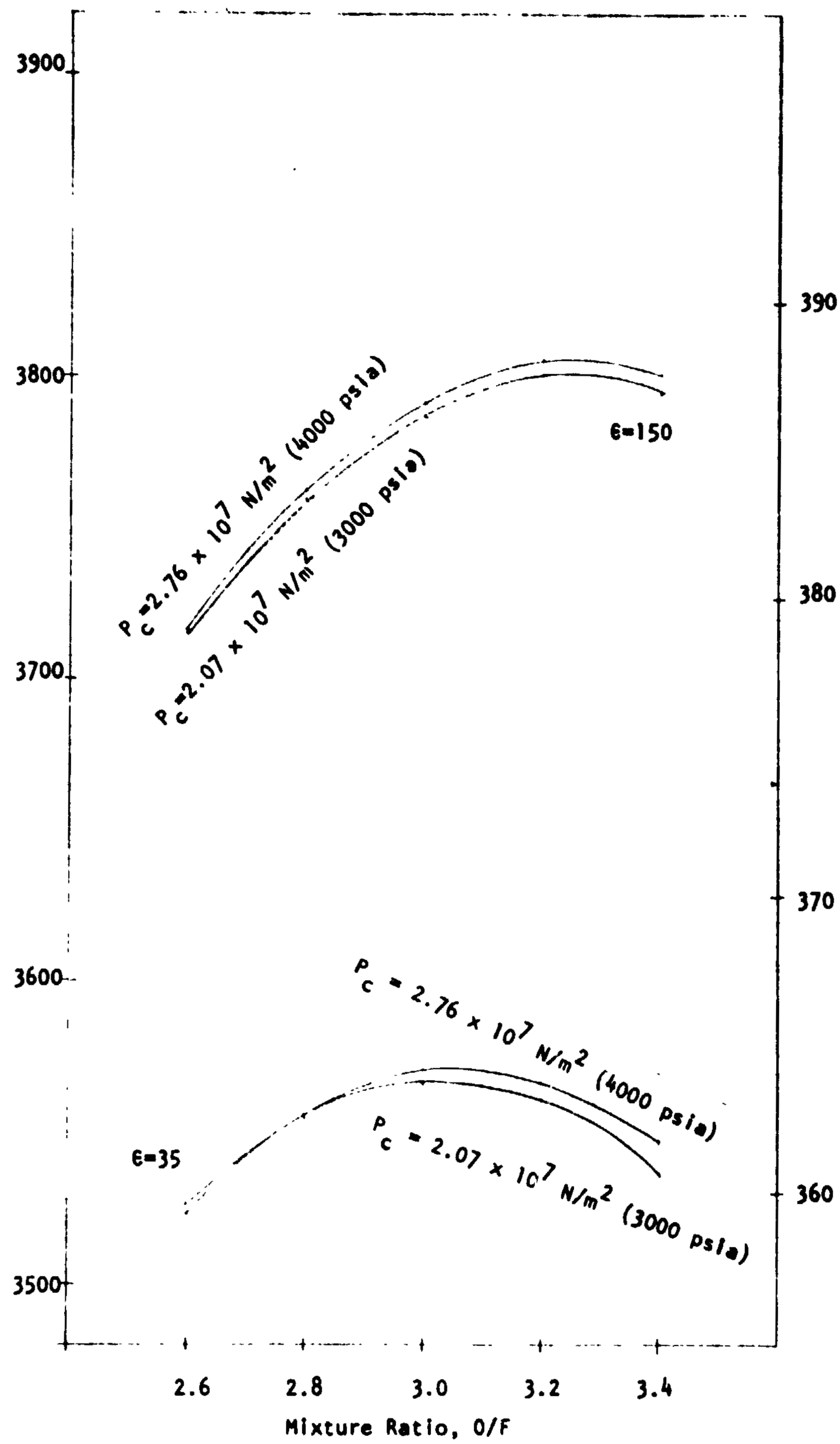

Figure 5. Theoretical Vacuum Performance for Oxygen/Propane (O_2/C_3H_8)

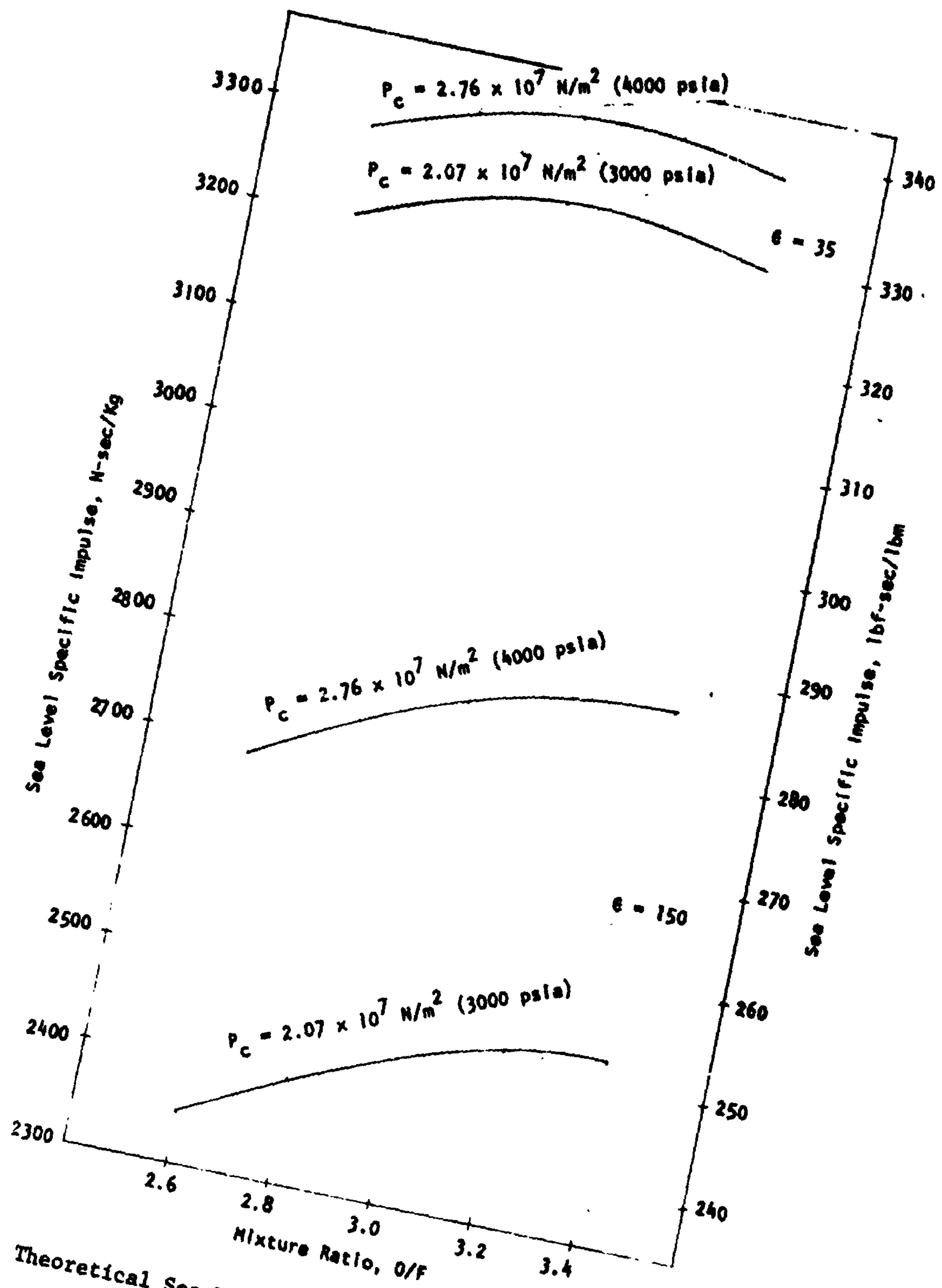

Figure 6. Theoretical Sea Level Performance for Oxygen/Propane (O_2/C_3H_8)

TABLE 2. PROPELLANT COMPARISON THEORETICAL PERFORMANCE

P_c = 3230 psia; ε = 35:1

	Main Chamber Mixture Ratio	I_s (Sea Level), seconds	I_s (Vacuum), seconds	ρ_{fuel}, lb/ft^3	ρ_{bulk}, lb/ft^3	Fuel NBP, R	Fuel Freezing Temperature, R
O_2/RP-1	2.8:1	328.7	358	50.0	64.0	882	405
O_2/CH_4	3.5:1	338	368	26.4	51.7	201	163
O_2/C_3H_8	3.0	333.5	364	36.6	57.6	416	154

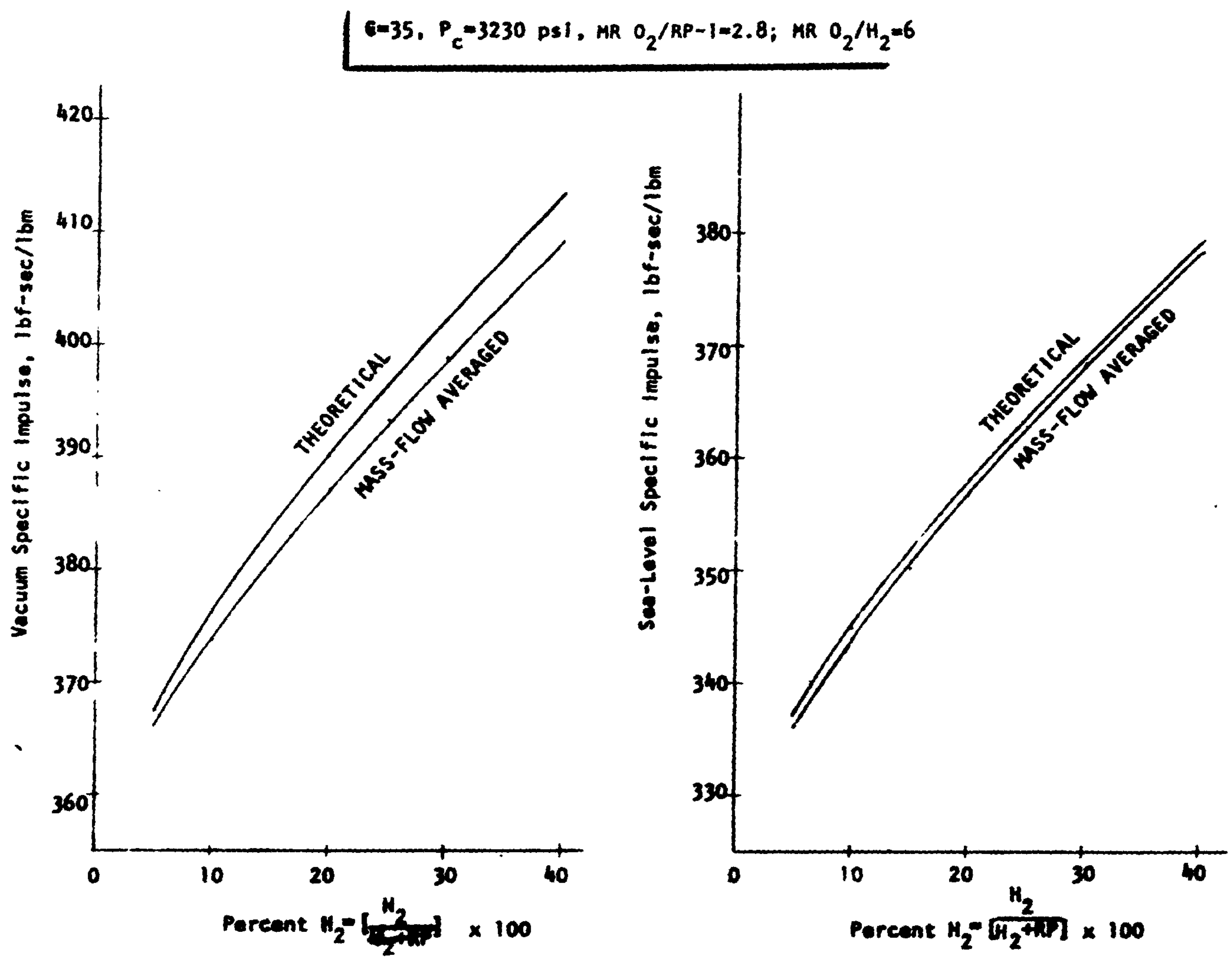

Figure 7. Comparison of Vacuum and Sea Level Performance for O_2/RP-1/H_2 Calculated by Two Methods

chamber between propellant components that are not modeled by the assumptions implicit in mass averaging. These reactions arise because the composition of the combined constituents in the chamber is not the same as the mass-averaged composition.

In performing the mass-average calculations, specific impulse values are selected for the two bipropellant pairs (O_2/H_2,O_2/RP) at one specific area ratio (ε). The values are then used to predict the performance of the combined system. However, each propellant combination results in different gas properties (e.g., γ) which result in different exit pressures at fixed ε. This pressure mismatch is ignored in mass-averaging I_s values, and may explain some of the difference between these values and theoretical ones. If the gases of the two different combustion systems were expanded to the same pressure ratio (e.g., the tripropellant system pressure ratio), the difference between the theoretically calculated specific impulse and the mass-averaged value will be approximately the same for vacuum and sea level operation.

Finally, it should be noted that the O_2/H_2 and O_2/RP specific impulse values used to predict O_2/RP/H_2 performance are each based on a specific mixture ratio (o/f). In theoretical computations of tripropellant performance, the actual proportions of O_2 to H_2 and RP are fixed by the chemistry in the combustion chamber. These proportions may differ from those assumed, resulting in differing performance values computed.

This comparison and the proposed explanations represent only a cursory analysis and a much more thorough investigation would be required to reach a more comprehensive conclusion. However, the mass-averaging method gives a conservative estimate of the predicted performance and is certainly adequate for these preliminary system requirements and comparison studies.

In the gas generator turbine power cycles, the turbine exhaust gases were assumed to be directed into the main nozzle at an area ratio compatible with the turbine exhaust pressure and expanded to the nozzle area ratio. The resulting I_s values for the secondary flow are presented in Table 3.

Turbine drive gas characteristics for oxidizer-rich and fuel-rich conditions are presented in Table 4. The function of γ and pressure ratio is specifically $1 - (1/PR)^{\gamma-1/\gamma}$ and is a measure of the energy available as the gas expands in the turbine. High values of C_p and $f(\gamma, PR)$ tend to minimize turbine flow for a given horsepower requirement.

TABLE 3. GAS GENERATOR CYCLE SECONDARY FLOW SPECIFIC IMPULSE

Propellants	I_s (sea level), seconds	I_s (vacuum), seconds
O_2/H_2	248.2	282.7
O_2/C_3H_8	122.2	142.7
O_2/CH_4	121.5	141.6

TABLE 4. TURBINE DRIVE GAS CHARACTERISTICS

	Turbine Pressure Ratio (PR) = 1.6:1				Temperature = 2000 R			
	LOX Rich				Fuel Rich			
	MR	γ	C_p	$f(\gamma,PR)$*	MR	γ	C_p	$f(\gamma,PR)$*
O_2/H_2	86	1.293	0.296	0.1	1.14	1.345	1.78	0.113
O_2/RP-1	33	1.31	0.294	0.105	0.41	1.135	0.66	0.054
O_2CH_4	37.5	1.289	0.284	0.1	0.43	1.16	0.875	0.0627
O_2/C_3H_8	34	1.288	0.283	0.099	0.44	1.147	0.691	0.584

*$f(\gamma,PR) = 1 - (1/RP)^{\gamma-1/\gamma}$

TASK II - THRUST CHAMBER THERMAL ANALYSIS

The objective of this task was to provide the heat transfer and cooling analysis support for the selected engine systems to be studied.

The initial effort was devoted to establishing the feasibility of cooling the SSME chamber and nozzle with the candidate hydrocarbon fuels or hydrogen. Further studies evaluated LOX cooling, H_2 cooling of an extendible nozzle, and any cooling variations that might offer some improvements to the engine systems being studied.

H_2 Cooling With LOX/Hydrocarbon Combustion

The current SSME chamber and 35:1 development nozzle design were analyzed to determine the coolant bulk temperature rise, coolant pressure drop, and wall temperatures as a function of H_2 coolant flowrate for LOX/RP-1 combustion.

Two coolant circuits were analyzed. Both use hydrogen as the coolant. The first is an uppass circuit where the nozzle and chamber are cooled in parallel. The second circuit is a downpass series circuit. Changes in the chamber coolant channel and nozzle tube geometry have not been considered.

The following conditions have been used for the O_2/RP-1 propellants:

$$P_o = 3237 \text{ psia}$$
$$T_o = 6512 \text{ F}$$
$$\dot{w}_g = 1455 \text{ lb}_m/\text{sec}$$
$$MR = 2.8$$
$$MW = 24.48 \text{ lb}_m/\text{lb}_m\text{-mole}$$

The heat transfer coefficient profile for the O_2/RP-1 propellant combination is obtained by correcting the SSME O_2/H_2 heat transfer coefficient profile uniformly by the flowrate and property ratios as given in the following equation:

$$\left.h_g\right)_2 = \left.h_g\right)_1 \frac{k_2}{k_1} \left[\frac{\dot{\omega}_g}{\dot{\omega}_g} \frac{2}{1} \frac{\mu_1}{\mu_2}\right]^{0.8} \left[\frac{Pr_2}{Pr_1}\right]^{0.4}$$

where subscript 1 = O_2/H_2 parameters

subscript 2 = O_2/RP-1 parameters

This is the normally accepted ratioing technique based on Nusselt number correlations. Using this ratio, the O_2/RP-1 heat transfer coefficient is 58% of the O_2/H_2 heat transfer coefficient. The major factor in the reduction in the heat transfer rate is the lower thermal conductivity (k) of the O_2/RP-1 propellant combination. The O_2/H_2 and O_2/RP-1 heat transfer coefficient profiles for the main chamber are shown in Fig. 8. The analogous profiles for the 35:1 [illegible] are shown in Fig. 9. The effect of burning the hydrogen coolant with the O_2 [illegible] RP-1 in the chamber has not been considered in determining these profiles.

The possibility of a carbon layer buildup on the chamber and nozzle wall could significantly reduce the heat flux to the coolant but, because of the uncertainty of its sustained existence, the conservative approach is taken and the carbon layer will not be assumed for the engine balance analysis. The effect of the carbon layer on the coolant requirement is presented here to show the magnitude of its effect.

For cases where a carbon layer is assumed, the following equation (Ref. 1) is used to calculate the carbon layer resistance:

$$x/k = e^{10-0.51\,G} \quad (\text{Btu/in.}^2\text{-sec-F})$$

Coolant bulk temperature rise and pressure drop are calculated using Regenerative Cooling Design/Analysis computer program models that have previously been set up for SSME analyses. The pressure drops computed using these models are increased by 10% to account for parasitic losses (inlet manifold, exit manifold, etc.). Using the data generated by these models, two-dimensional wall temperatures for the chamber and nozzle are calculated using models that are set up on the time-sharing computer system.

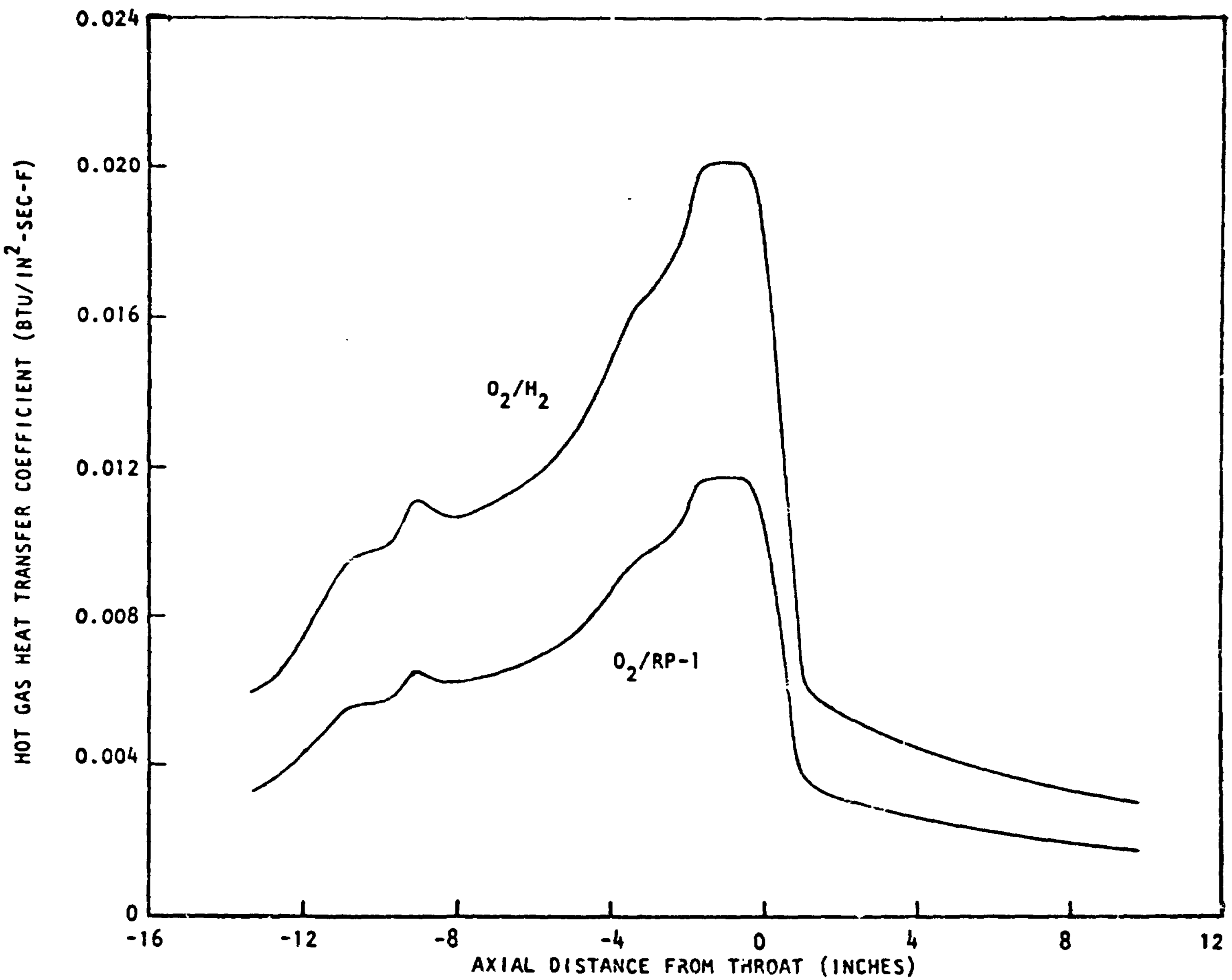

Figure 8. Main Combustion Chamber Heat Transfer Coefficient Profiles for O_2/H_2 and O_2/RP-1 Propellant Combinations (P_c = 3237 psia)

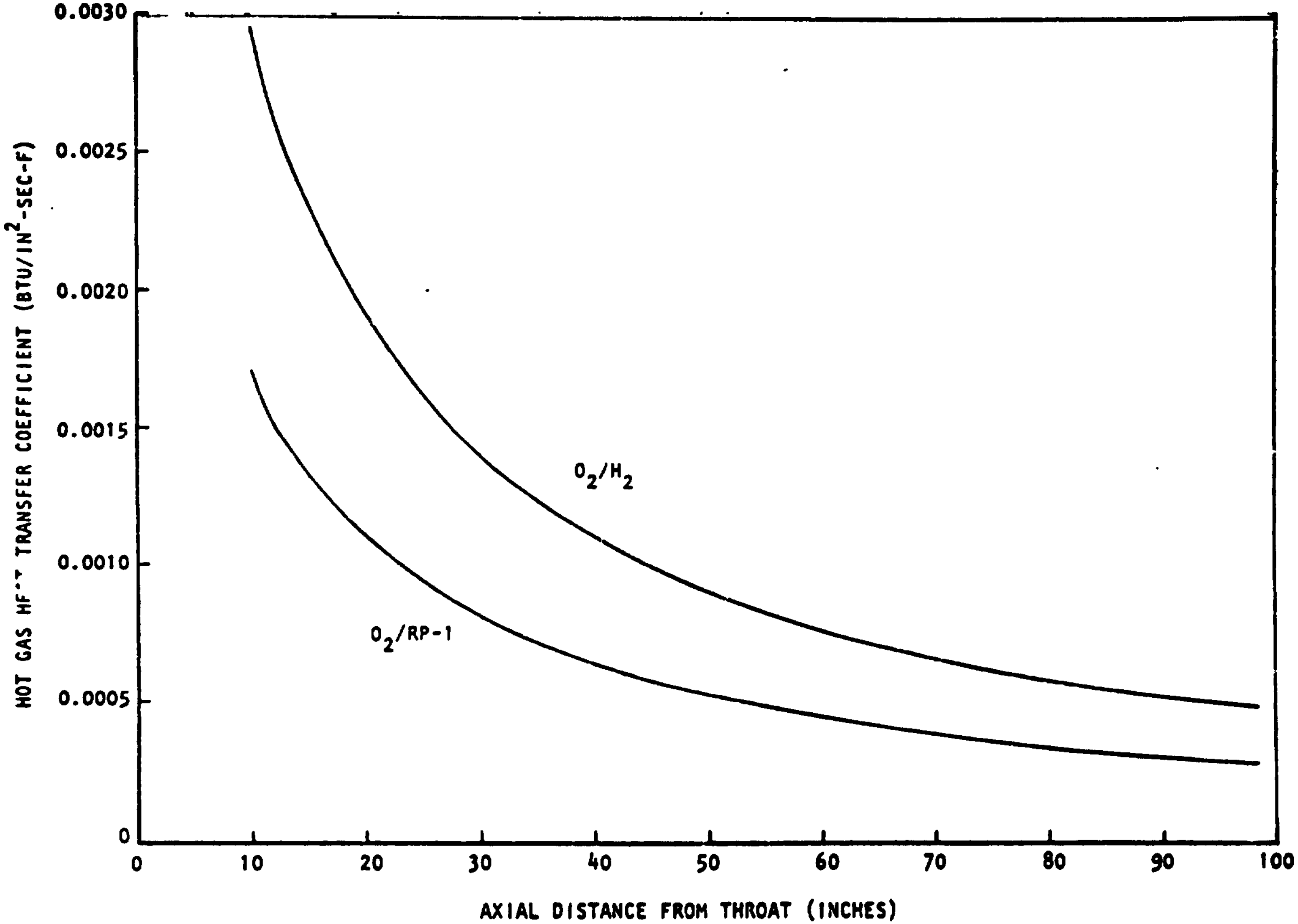

Figure 9. 35:1 Nozzle Heat Transfer Coefficient Profiles for O_2/H_2 and O_2/RP-1 Propellant Combinations (P_c = 3237 psia)

For the uppass circuit with the chamber and nozzle cooled in parallel, the hot-gas wall temperature as a function of flowrate is shown in Fig. 10a. Assuming no carbon coating, the nozzle would require a coolant flowrate of 18.5 lbm/sec to maintain the same maximum temperature as for the O_2/H_2 SSME flight nozzle. When assuming a carbon coating, the required nozzle flowrate is only 1.5 lbm/sec.

When assuming no carbon coating, the temperature -limiting location for the O_2/RP-1 chamber occurs in the combustion zone. To cool this region to the same temperature as in the combustion zone of the O_2/H_2 SSME requires a coolant flowrate of 15.3 lbm/sec. When assuming a carbon coating, the temperature-limiting location is in the throat (-1") and requires a coolant flowrate of 10.7 lbm/sec to cool the chamber to the same temperature as the O_2/H_2 SSME.

The coolant bulk temperature rise for the chamber and nozzle is given in Fig. 10b. The coolant pressure drop is given in Fig. 10c. The pressure drop is shown for inlet pressures of 5000, 6000, and 7000 psia for both the nozzle and chamber. When assuming a carbon coating, the nozzle pressure drop is extremely low (~2 psi) because of the low required coolant flowrate.

For the downpass circuit with the chamber and nozzle cooled in series, the hot-gas wall temperature as a function of coolant flowrate is shown in Fig. 11a. Since it is a series circuit, the minimum required flowrate is controlled by the wall temperature in the throat region of the chamber. Assuming no carbon coating, a coolant flowrate of 19 lbm/sec is required to maintain the same temperature in this region as for the O_2/H_2 SSME. The cooling in the throat region is hampered by the fact that the benefits of curvature enhancement to the coolant coefficient are not realized in the downpass circuit. The maximum nozzle temperature occurs near the maximum expansion ratio (94") instead of near the nozzle-to-chamber attach point because of the large bulk temperature rise. When assuming a carbon coating, the required coolant flowrate for the downpass circuit is 14.6 lbm/sec.

The coolant bulk temperature rise is shown in Fig. 11b. The coolant pressure drop (chamber plus nozzle) is shown in Fig. 11c for three inlet pressures. When assuming no carbon coating, the inlet pressure should be maintained to at least near 4000

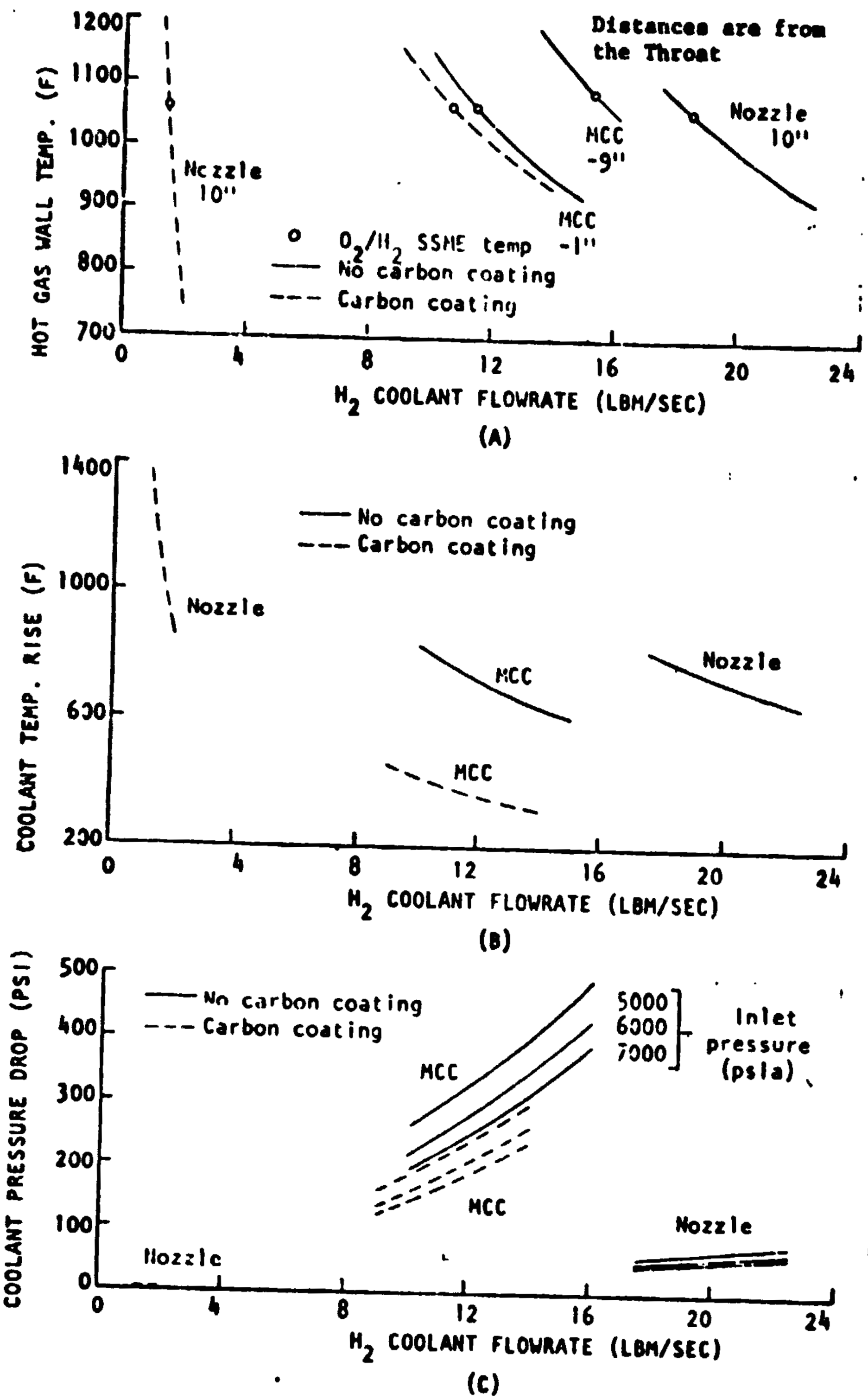

Figure 10. Hot-Gas Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop as Functions of Coolant Flowrate for Uppass Parallel Coolant Circuits for the SSME Main Combustion Chamber and 35:1 Nozzle O_2/H_2 Hydrocarbon Combustion

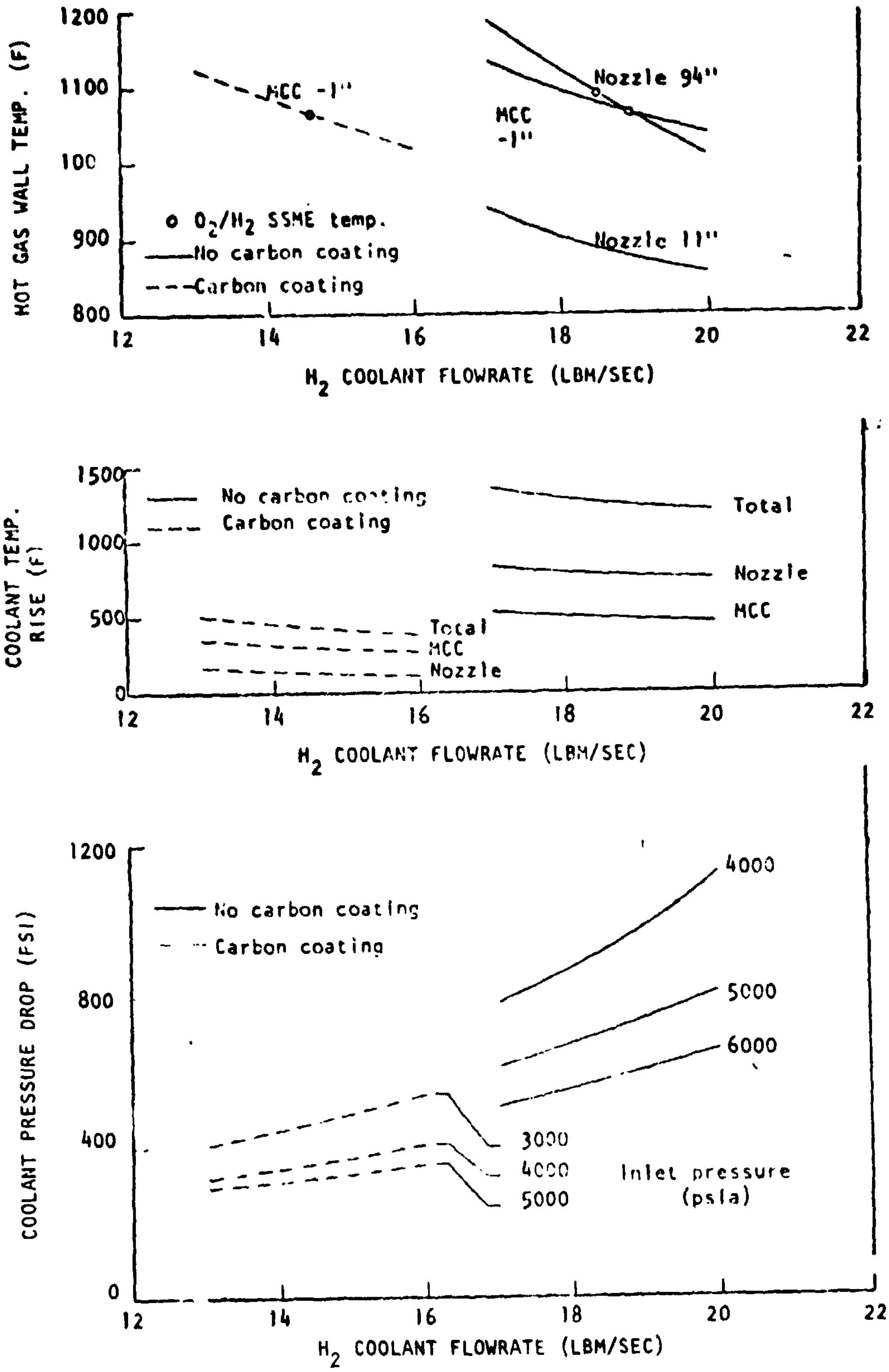

Figure 11. Hot-Gas Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop as Functions of Coolant Flowrate for Downpass Series Coolant Circuits for the SSME Main Combustion Chamber and 35:1 Nozzle O_2/Hydrocarbon Combustion

psia to avoid choking in the coolant channels of the chamber. Additionally, it is desirable to keep the coolant pressure greater than the hot-gas pressure in case of a leak through the hot-gas wall of a coolant channel.

The coolant flowrate requirements are summarized below.

H_2 COOLANT AND O_2/RP-1, O_2/CH_4, OR O_2/C_3H_8 COMBUSTION

		Nozzle $\dot{\omega}_c$, lbm/sec	Chamber $\dot{\omega}_c$, lbm/sec	Total, Flowrate, lbm/sec	Fraction of O_2, H_2 SSME $\dot{\omega}_c$, lbm/sec
Uppass Parallel	No Coating	18.5	15.3	33.8	0.44
	Coating	1.5	10.7	12.2	0.16
Downpass Series	No Coating	19.0	19.0	19.0	0.25
	Coating	14.6	14.6	14.6	0.19

The uppass parallel cooling circuit was selected for the candidate engine system and these results were assumed to be valid for O_2/C_3H_8 and O_2/CH_4 combustion cases with H_2 coolant. This assumption is reasonable since, at comparable chamber pressures, the combustion gas flowrates and properties combine to give similar heat flux profiles. Hence, the coolant parameters would not change.

Hydrocarbon Fuel Cooling With LOX/Hydrocarbon Combustor

Previous studies (Ref. 2 and 3) have shown that the use of RP-1 as a coolant is limited to approximately 2000 psia chamber pressure because of bulk temperature rise limitations and the resultant coking of the fuel which occurs at 600 F. Based on these results, it was decided that only CH_4 and C_3H_8 would be considered as a coolant. Studies identical to that previously described for the H_2 coolant were then conducted for these two systems. The results are presented in Fig. 12

through 15 for the 3237 psia chamber pressure conditions. This analysis has also been conducted for a 4000 psia chamber pressure for the LOX/CH_4 propellant combination. These results are shown in Fig. 16 and 17.

Oxygen Cooling With O_2/Hydrocarbon Combustion

The oxygen cooling requirements have been determined for O_2/hydrocarbon combustion in the SSME main chamber and the 35:1 nozzle. A chamber pressure of 3237 psia and the current SSME combustion chamber and coolant channel geometry with a parallel uppass circuit were assumed. The results of this analysis are presented in Fig. 18 and 19. To maintain a maximum hot-gas wall temperature of 1050 F (typical of SSME conditions) requires a coolant flowrate of 220 and 275 lb/sec in the chamber and nozzle, respectively. At these flowrates, a coolant temperature rise and ΔP of 450 F and 4500 psi, respectively, occur in the chamber and 625 F and 500 psia, respectively, in the nozzle. These values are based on a 8500 psia inlet pressure. Engine balance results indicate that an inlet pressure of approximately 8600 psia is required to achieve a chamber pressure of 3237 psia, (concepts No. 10 and 11). If the inlet pressure drops much below that value, there is the danger of the flow choking in the cooling jacket. It was found in the analysis that at 8000 psia inlet pressure, choking would occur at a flowrate of 210 lb/sec or greater. Because of the very high inlet pressure and pressure drop requirements that result for the chamber due to the SSME channel geometry constraint, it was decided to conduct an additional analysis to determine how much the inlet pressure and pressure drop requirement could be reduced by adopting a more nearly optimum channel configuration for the combustion chamber. A chamber was configured using a narrower channel and thinner hot-gas wall and the cooling analysis was repeated. The results, presented in Fig. 20 for an inlet pressure of 7000 psia, show that the combustion chamber coolant flowrate can be reduced to 150 lb/sec with a resulting pressure drop of 2100 psia. The pressure drop could be reduced further by increasing the channel height to increase the coolant flowrate for a given mass velocity. This will reduce the coolant temperature rise and the resultant pressure drop. Since the chamber coolant inlet pressure and pressure drop drive the pump discharge pressure requirement, there is no incentive to redesign the nozzle channel geometry.

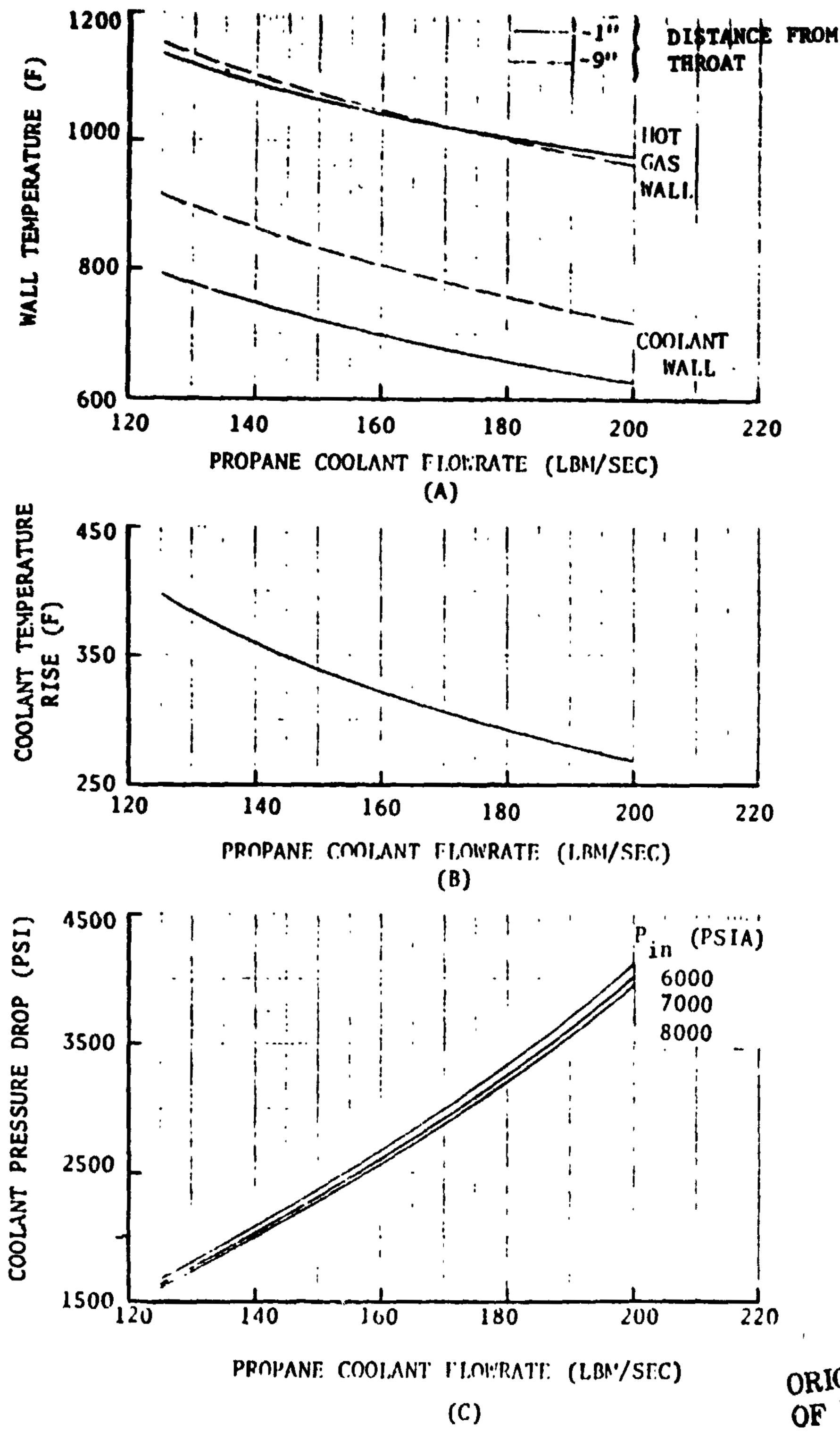

Figure 12. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for an O_2/Propane Combustor (P_c = 3230 psia) Uppass Propane Cooled

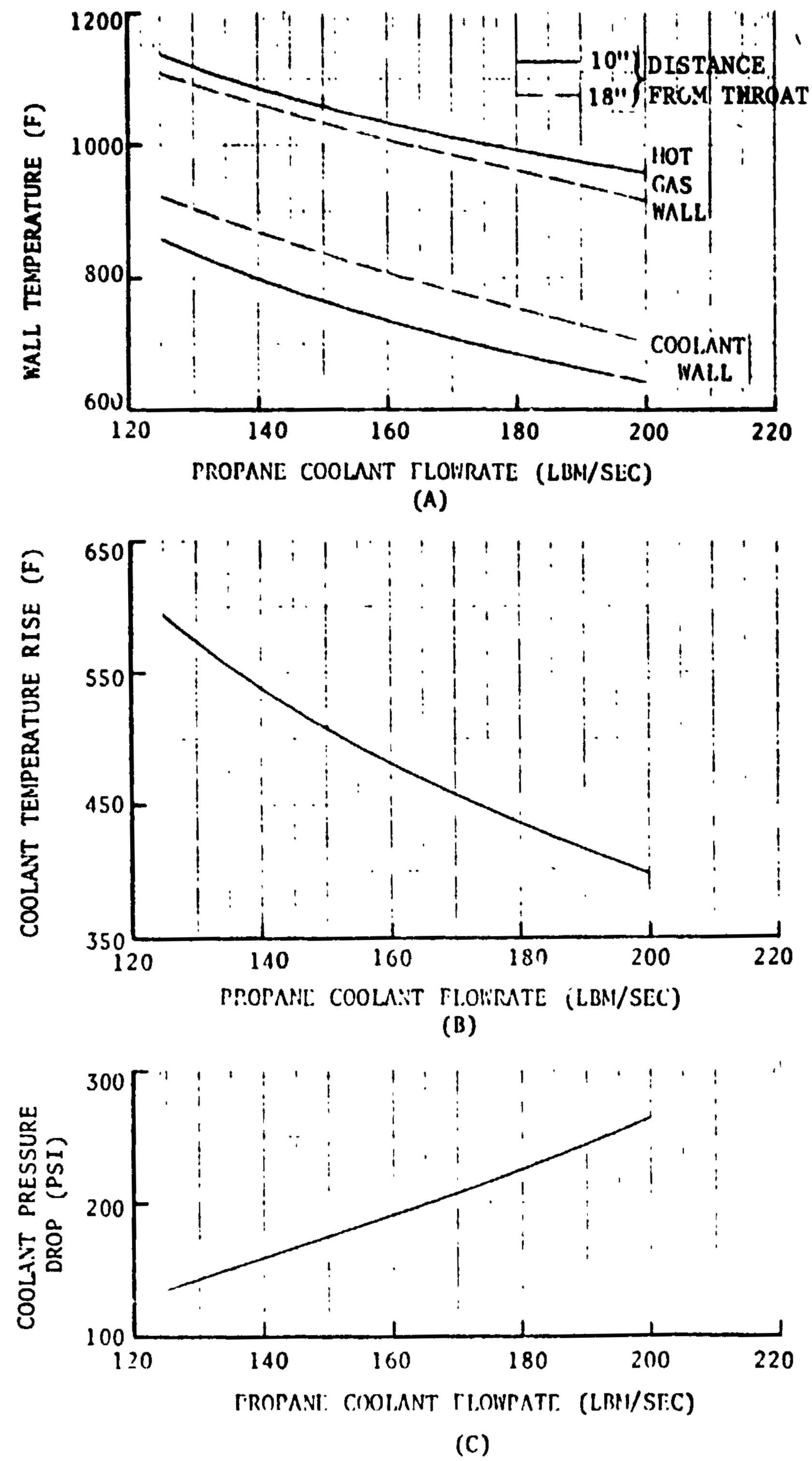

Figure 13. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for an O_2/Propane 35:1 Nozzle (P_c = 3230 psia) Uppass Propane Cooled

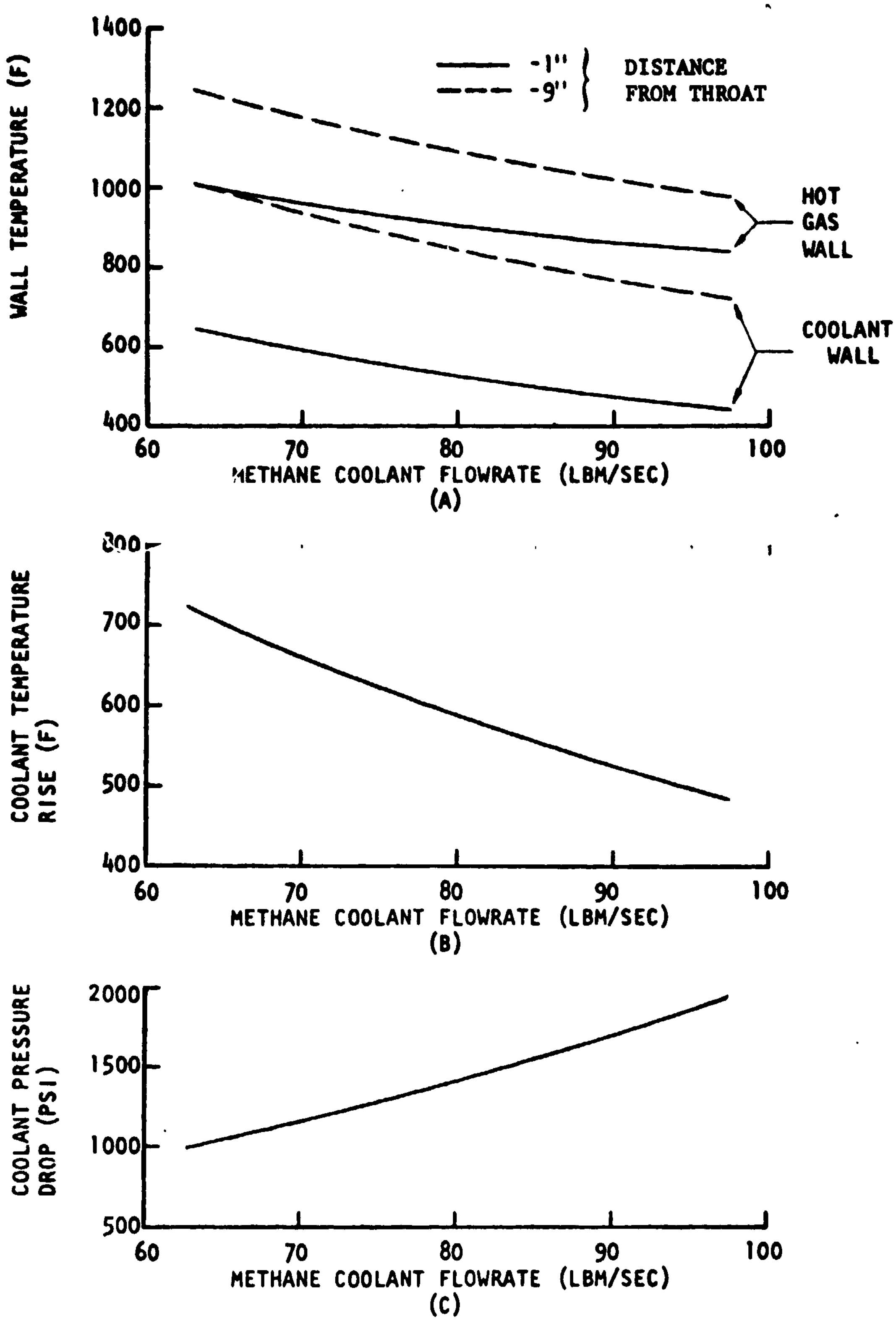

Figure 14. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c=3230 psia O_2/CH_4 CH_4-Cooled Uppass Chamber

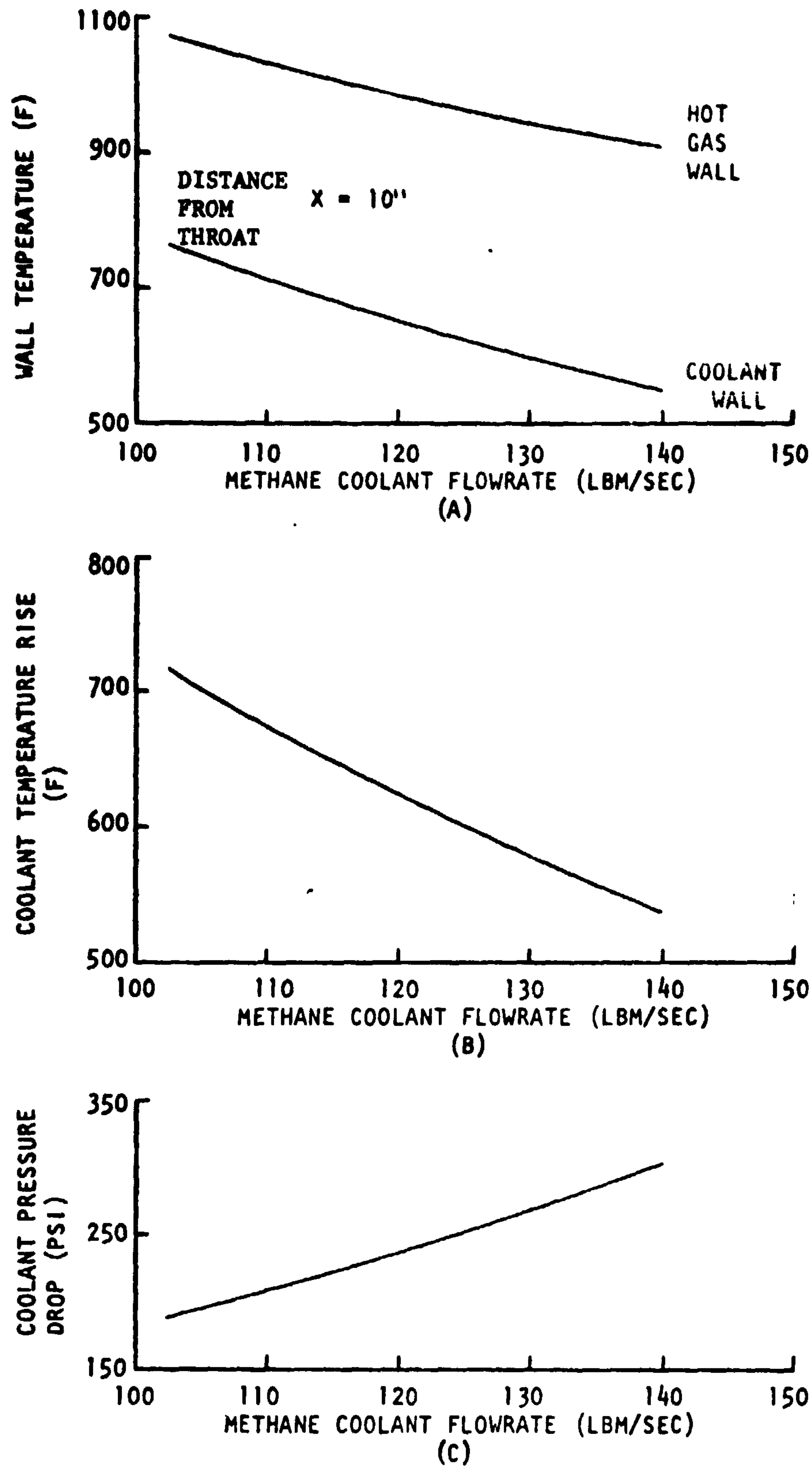

Figure 15. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c=3230 psia O_2/CH_4 CH_4-Cooled Uppass 35:1 Nozzle

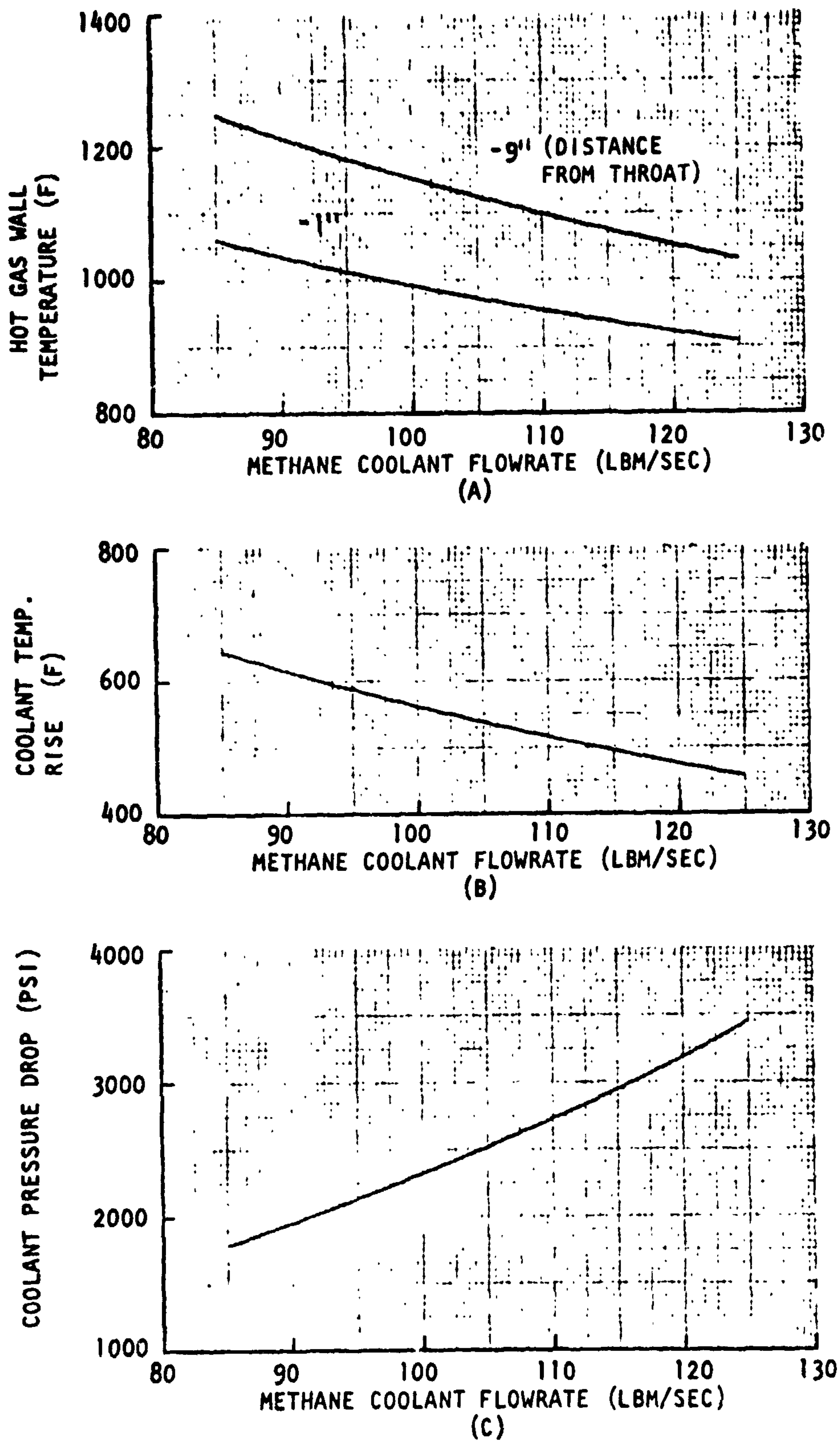

Figure 16. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c = 4000 psia O_2/CH_4 CH_4-Cooled Uppass Chamber

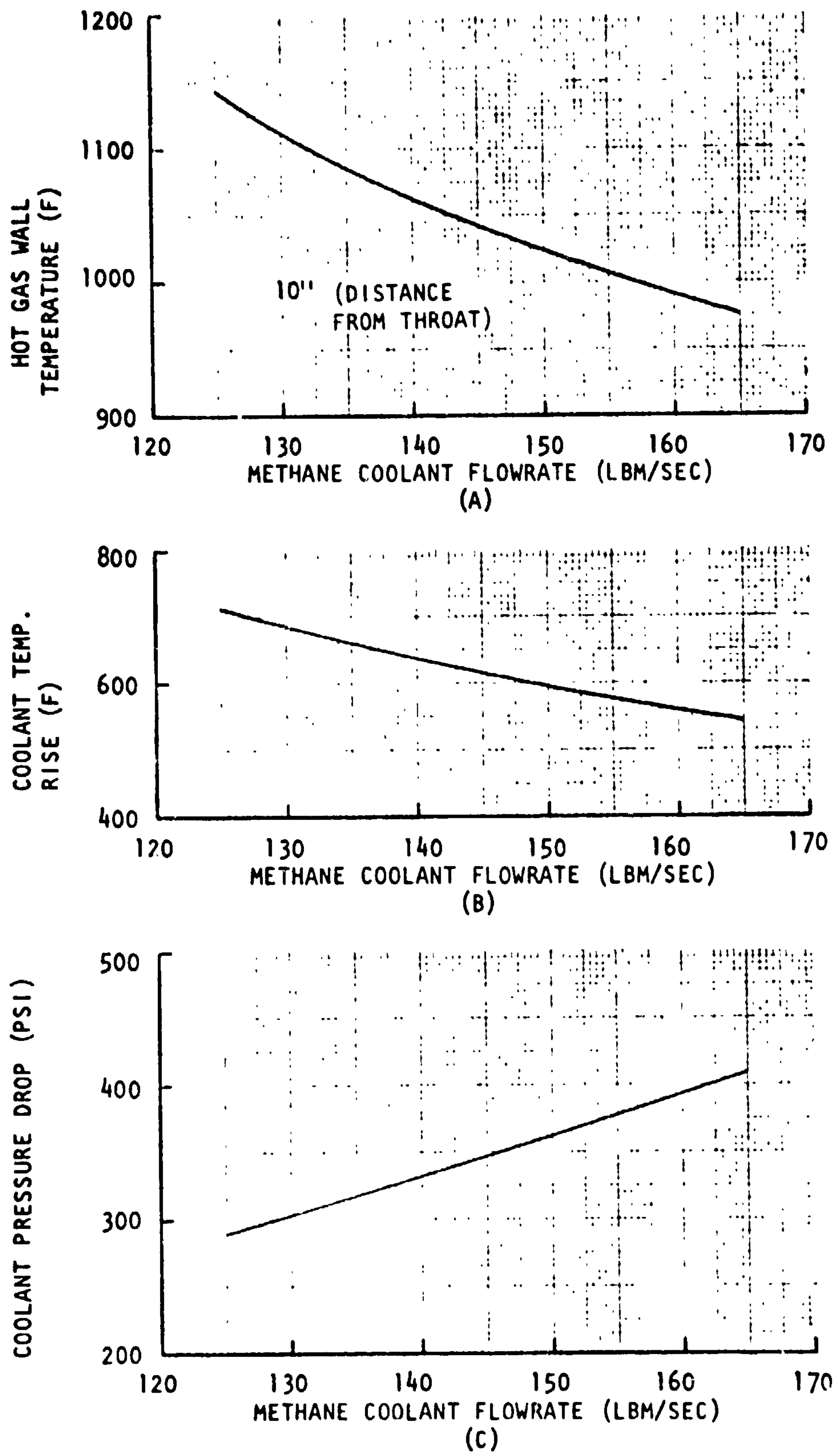

Figure 17. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c=4000 psia O_2/CH_4 CH_4-Cooled Uppass 35:1 Nozzle

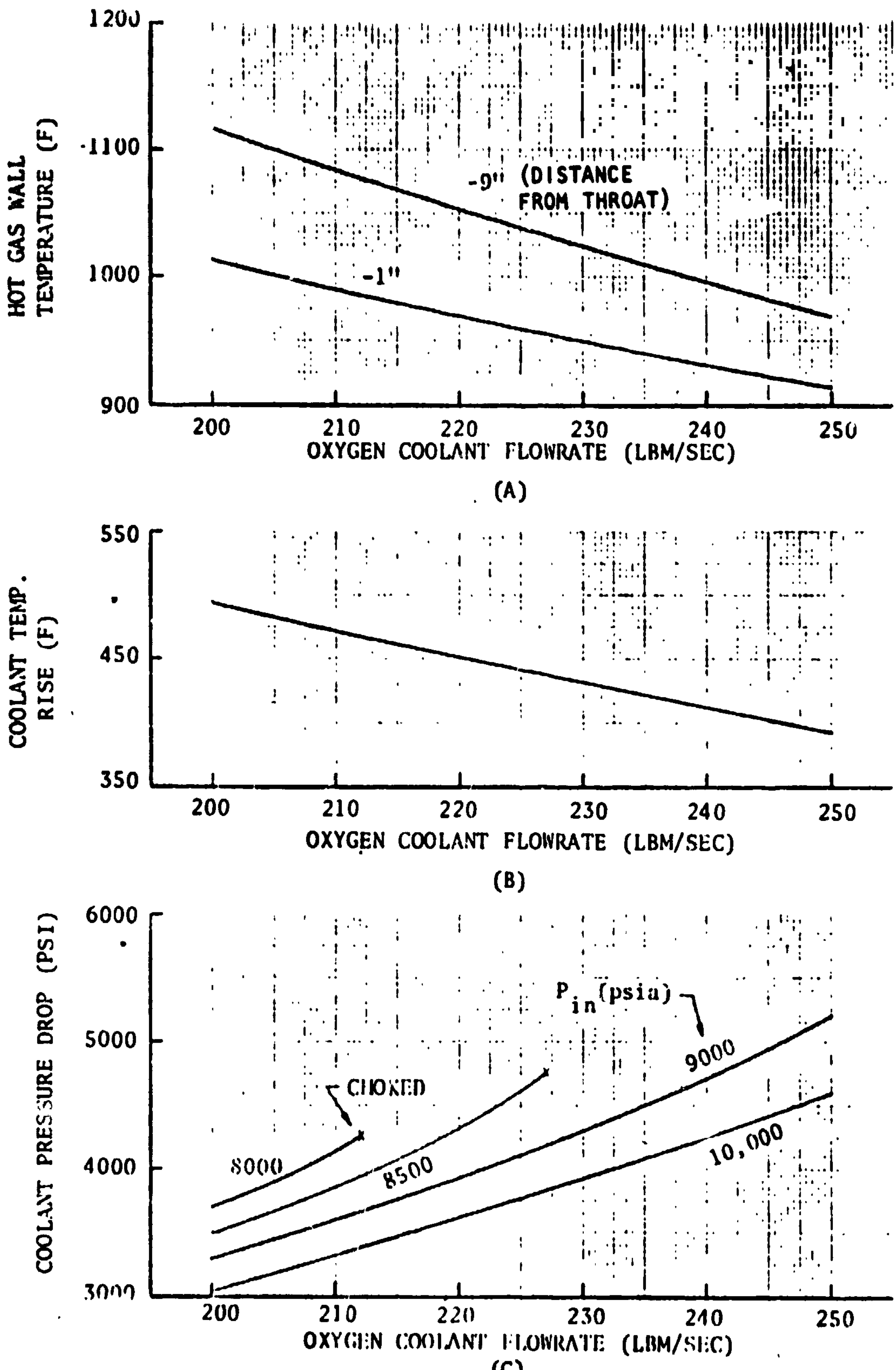

Figure 18. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c=3237 psia O_2/Hydrocarbon O_2-Cooled Uppass Chamber (Present SSME Channel Design)

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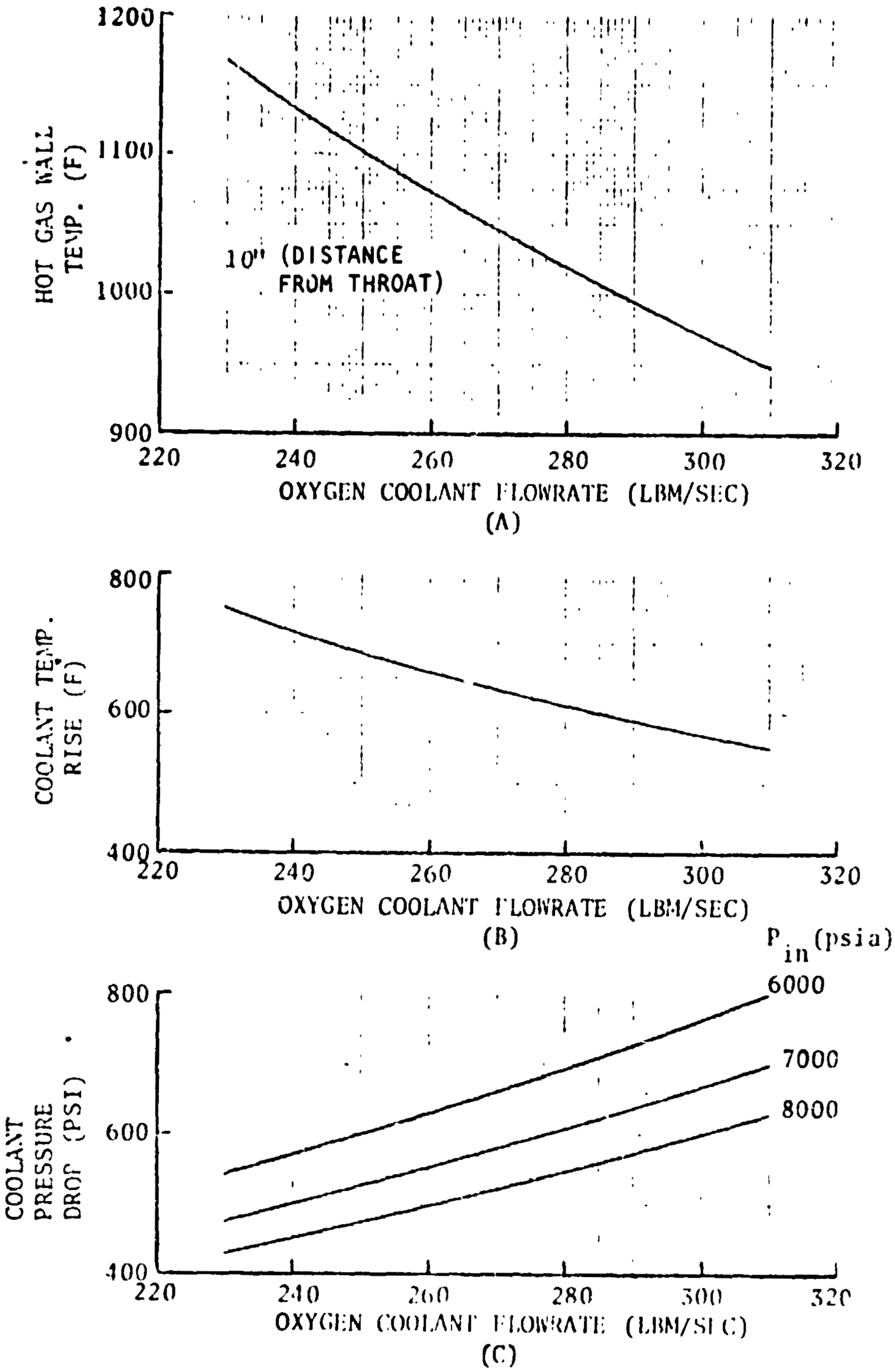
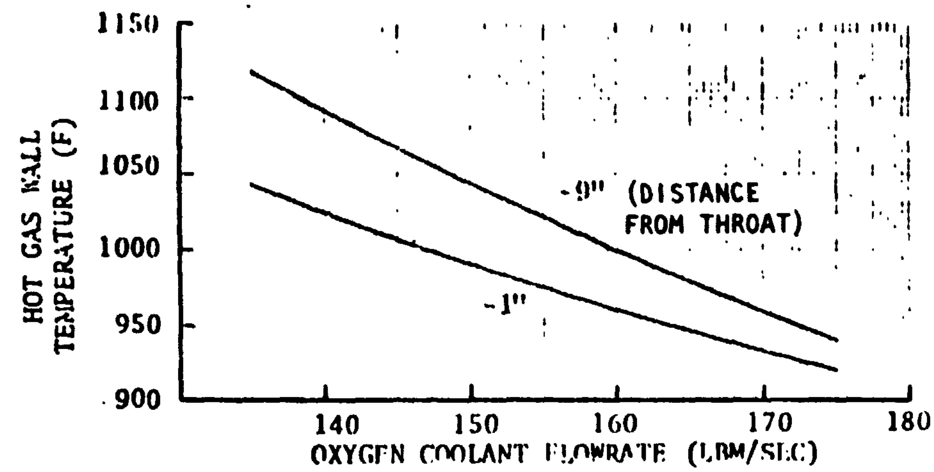
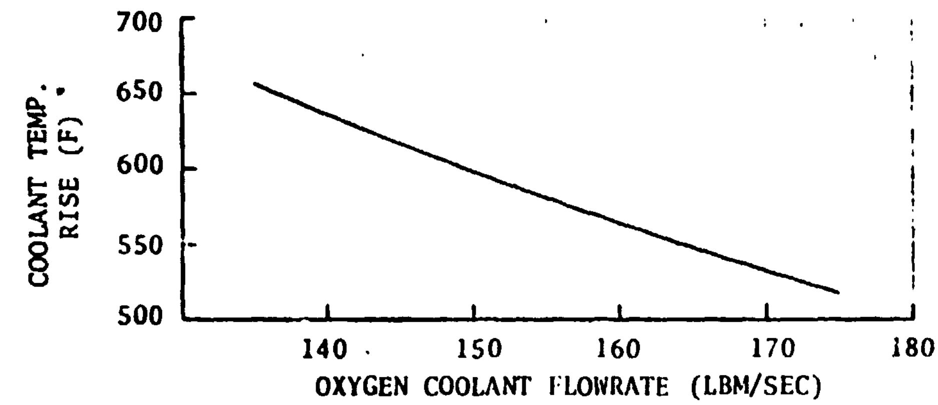


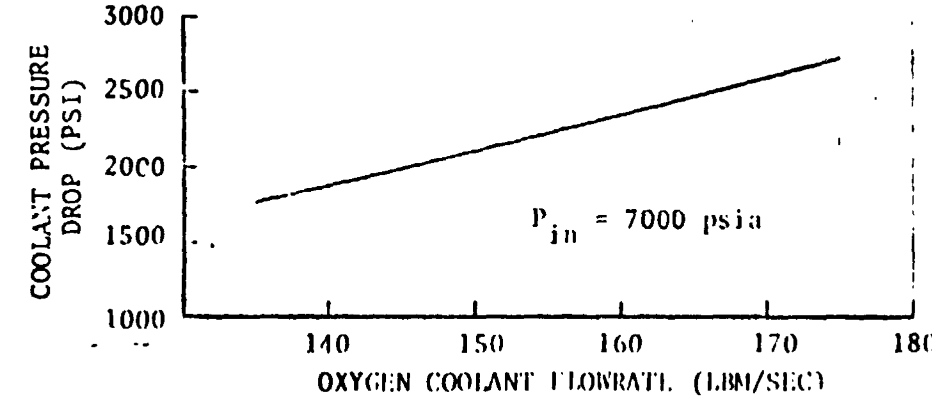
Figure 19. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c = 3237 psia O_2/Hydrocarbon O_2-Cooled Uppass 35:1 Nozzle



(A)



(B)



(C)

Figure 20. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a P_c=3237 psia O_2/Hydrocarbon O_2-Cooled Uppass Chamber (Channel Redesign)

O_2 Cooling With O_2/H_2 Combustion

The main incentive in considering a LOX-cooled chamber is that it would not be necessary to switch coolants between mode 1 and mode 2 operation. A brief study was conducted to determine if LOX cooling would be applicable for the SSME operation in mode 2 (O_2/H_2 combustion). O_2/H_2 heat transfer rates are 60 to 70% higher than O_2/hydrocarbon rates at the same chamber pressure. It was found that with the current coolant channel geometry, the maximum chamber pressure would be limited to less than 2000 psia. If the channel height were doubled in the chamber to permit an increase in coolant flowrate, the maximum chamber pressure would be 2500 psia with a flowrate of 700 lb/sec and a ΔP of 5400 psi. A chamber pressure of 3237 psia could be achieved only with a complete redesign of the chamber coolig geometry and would still require an excessively high coolant pressure drop. With these results, it is apparent that LOX cooling is not a feasible candidate for a dual-fuel engine using LOX/hydrocarbon in mode 1 and LOX/H_2 in mode 2 since the mode 2 operation is greatly limited.

H_2 Cooling of an Extendible Nozzle

The extendible nozzle contour used in this analysis along with the SSME development nozzle contour is shown in Fig. 21. For this analysis, no effort was made to optimize the extendible nozzle contour. The chamber throat radius is 5.15 inches (SSME main chamber).

The hot-gas heat transfer coefficient as a function of the nozzle expansion ratio is shown in Fig. 22. From ε = 77.5:1 the heat transfer coefficient is for the SSME flight nozzle at FPL (P_c = 3237 psia). From ε = 77.5:1 to ε = 150:1, the heat transfer coefficient is obtained by extrapolation of the curve from ε = 5:1 to ε = 77.5:1.

For this analysis, a constant-diameter tube is assumed. The number of tubes was varied to minimize the coolant flowrate while maintaining a reasonable size tube. The geometry selected consists of 2520 (7 x 360) tubes with an unformed diameter of 0.16 inch. A tube-wall thickness of 0.009 inch was assumed. The tube material is A-286 (same as the SSME nozzles).

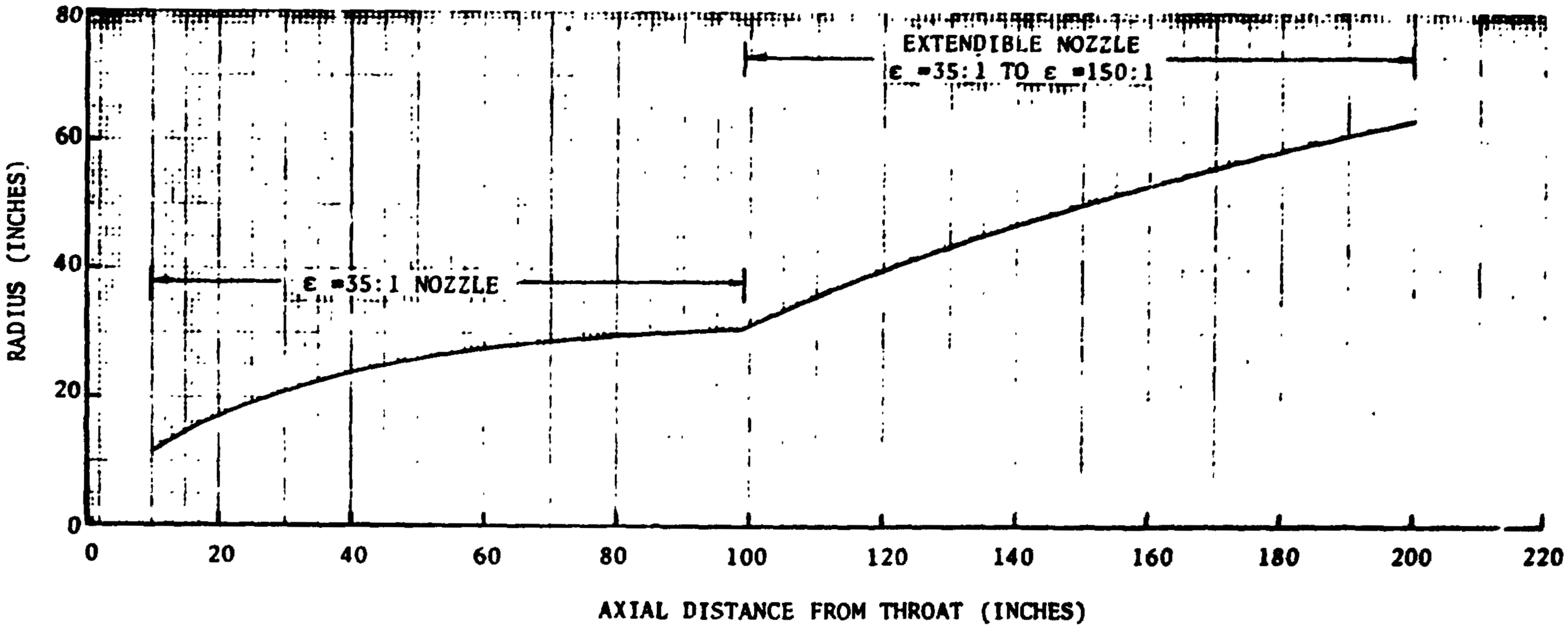


Figure 21. Extendible Nozzle Contour

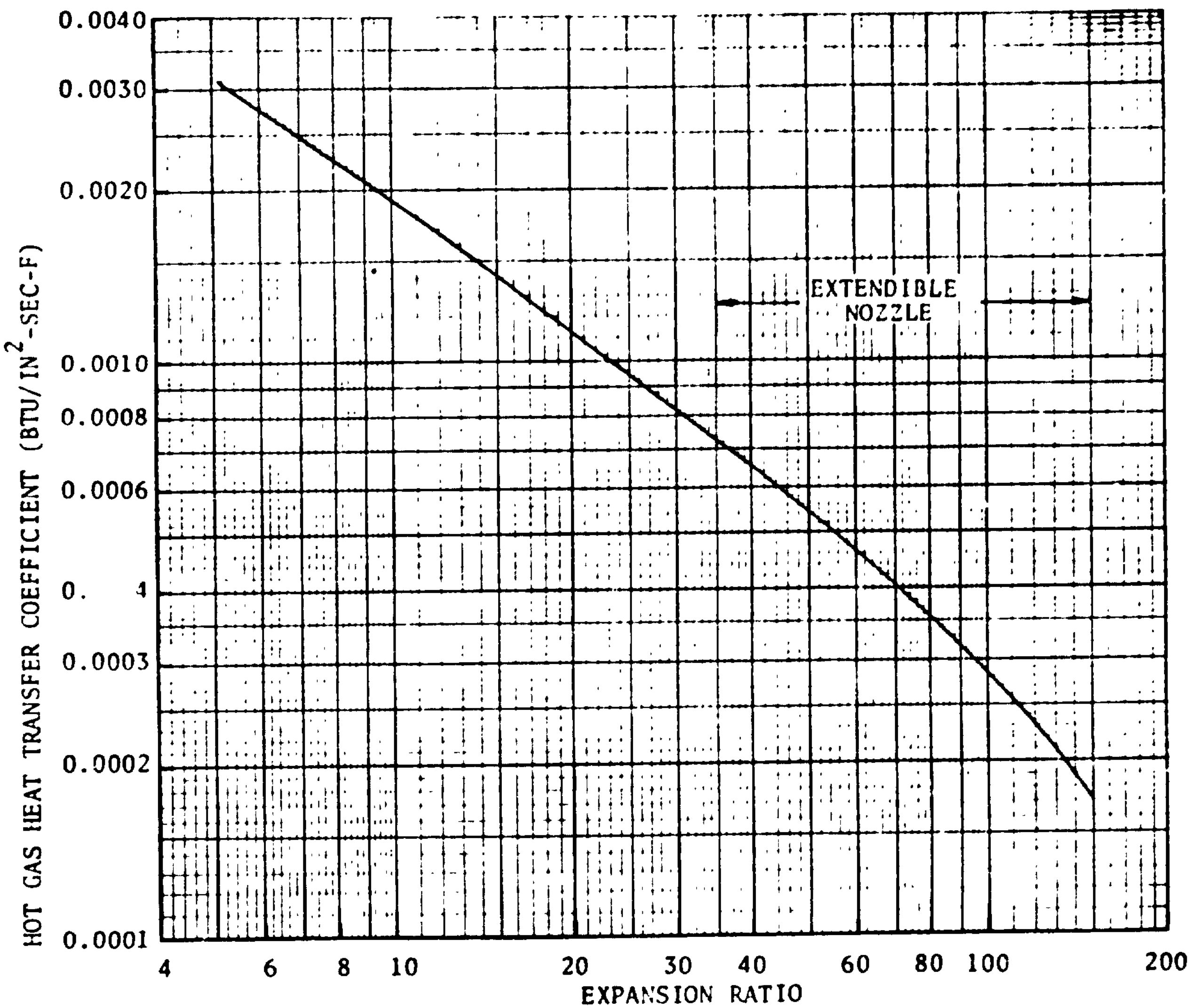


Figure 22. Hot-Gas Heat Transfer Coefficient Profile (P_c = 3237 psia) O_2/H_2 Combustion

A downpass coolant circuit has been selected for this design. This provides the low-temperature coolant in the maximum heat flux region (minimum expansion ratio). A coolant inlet pressure of 7000 psia and an inlet temperature of -360 F are assumed.

The maximum hot-gas wall temperature as a function of the coolant flowrate is shown in Fig. 23. To keep the maximum wall temperature at 1000 F would require a coolant flowrate of 18.5 lbm/sec. The maximum coolant mass velocity is 0.7 lbm/in.^{2}-sec. The maximum heat flux is 3.2 Btu/in.^{2}-sec and the nozzle heat load is 58,500 Btu/sec. For a coolant flowrate of 18.5 lbm/sec, the coolant temperature rise is 840 F and the pressure drop is only 7 psi. This analysis is preliminary and is only intended to show the feasibility of H_2 cooling of an extendible nozzle and the impact on the system. Considerable further analysis is required before a recommended design could be established.

Summary of Results

A summary of the mode 1 regenerative cooling system design points is shown in Table 5. The heat transfer/cooling analysis was conducted assuming the current SSME chamber and cooling channel geometry except in the one LOX cooled case where the channel dimensions were increased to reduce the ΔP. These design points were used in the mode 1 engine system mass/pressure balances.

TASK III - CYCLE AND POWER BALANCE

The objectives of this task were to define the candidate engine system cycles and perform cycle power balances to determine the required component flowrates, turbine inlet temperatures, and pump discharge pressures based on the pressure lossess of the various components. Power cycles examined included staged combustion and gas generator.

Selection of Engine System Candidates

Previous NASA funded studies (Ref. 4) indicate some possible advantage in operating a tripropellant engine in either a series burn or parallel burn

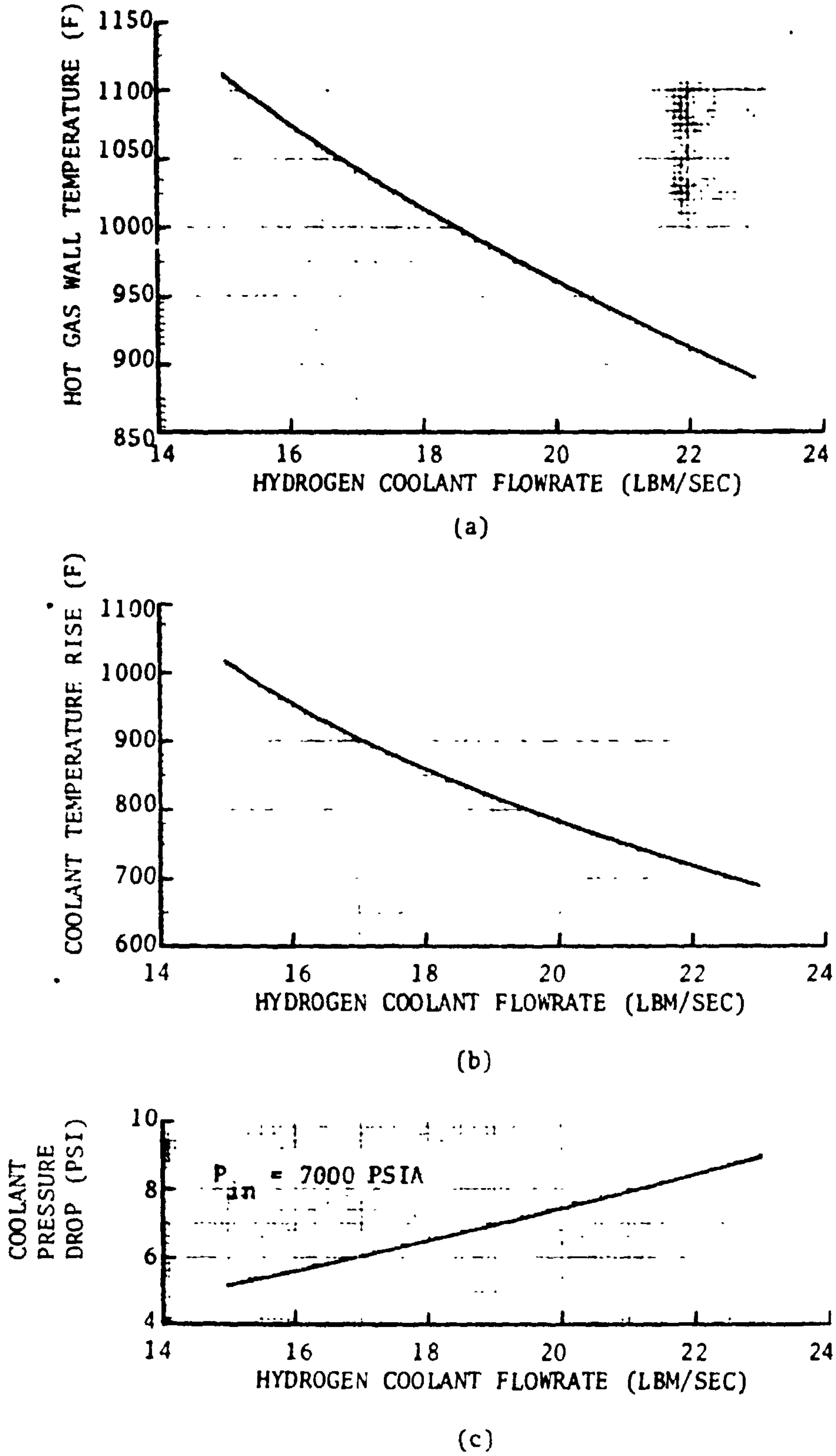


Figure 23. Wall Temperature, Coolant Temperature Rise and Coolant Pressure Drop for a P_c = 3237 psia O_2/H_2 Hydrogen-Cooled Downpass Extendible Nozzle

TABLE 5 . MODE 1 COMBUSTION CHAMBER AND NOZZLE COOLING DESIGN POINTS

($T_{wall\ max} \approx 1050$ F)

Combustion	Coolant	Chamber			Nozzle		
		$\dot{\omega}$, lb/sec	ΔT, F	ΔP, psi	$\dot{\omega}$, lb/sec	ΔT, F	ΔP, psi
O_2/RP-1	LH_2 (P_c = 3230)	15.5	600	500	18.5	750	75
O_2/CH_4	(P_c = 4000)	16.5	575	600	19.5	700	90
and O_2/C_3H_3	LO_2 (P_c = 3230)	225	450	4500	275	625	450
↓	(P_c = 3230)*	150	600	2100			
O_2/CH_4	CH_4 (P_c = 3230)	85	550	1600	107	690	200
	(P_c = 4000)	125	460	3500	150	620	340
O_2/C_3H_8	C_3H_8** (P_c = 3230)	160	320	2600	150	500	175
	(P_c = 4000)	200	270	5600	175	400	400

*Redesign of SSME coolant channels
**May have coolant side-wall coking problem

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configuration, as illustrated in Fig. 24. The series burn engine requires the sequential burning of a hydrocarbon fuel (mode 1) followed by H_2 in mode 2, and this is the engine classification of primary interest to this study. The tri-propellant engine used in the parallel burn configuration does not require the sequential burning of two fuels but may use the second fuel (hydrogen) as a coolant to avoid a possible inadequate cooling capability of the primary fuel. This later type of engine is actually a simplified version of the dual-mode engine and these single-mode engines can be derived from the dual-mode engines formulated in this study.

Ground rules were established for the dual-mode engine concept (Fig. 25) and a set of candidate engine cycle and propellant/coolant combinations was established.

Formulating these various engine system configurations provides a basis for evaluating engine system and component requirements for a dual-fuel engine and the subsequent compatibility of the SSME with these requirements. These candidate systems (Fig. 26) include all three specified hydrocarbon fuels and the full range of regenerative coolants. This group of candidate engine systems is considered to be representative of the power cycles, propellant combinations, and cooling techniques of interest. Other engine system variations could be derived from these ground rules but would not offer any greater potential for the adaptibility of SSME hardware. One additional important guideline in establishing the candidate engine systems was to minimize the use of H_2 in the mode 1 operation.

Since the SSME preburners operate fuel rich, it was particularly desirable to include several staged combustion engine cycles with all preburners fuel rich. As work was begun on the engine power balance calculations, it was found that even if all of the available fuel is used, turbine inlet temperatures exceeding 2000 R were required to achieve a power balance on the all-fuel-rich preburner systems. This is a result of the lower energy available per pound of turbine drive gas when comparing LOX/hydrocarbons with LOX/H_2. It is felt that these turbine inlet temperatures exceed the capability of the existing SSME hardware. Actually, some of the fuel must bypass the preburners to provide coolant for the injector face and hot-gas manifold. It is also desirable to maintain some engine power cycle

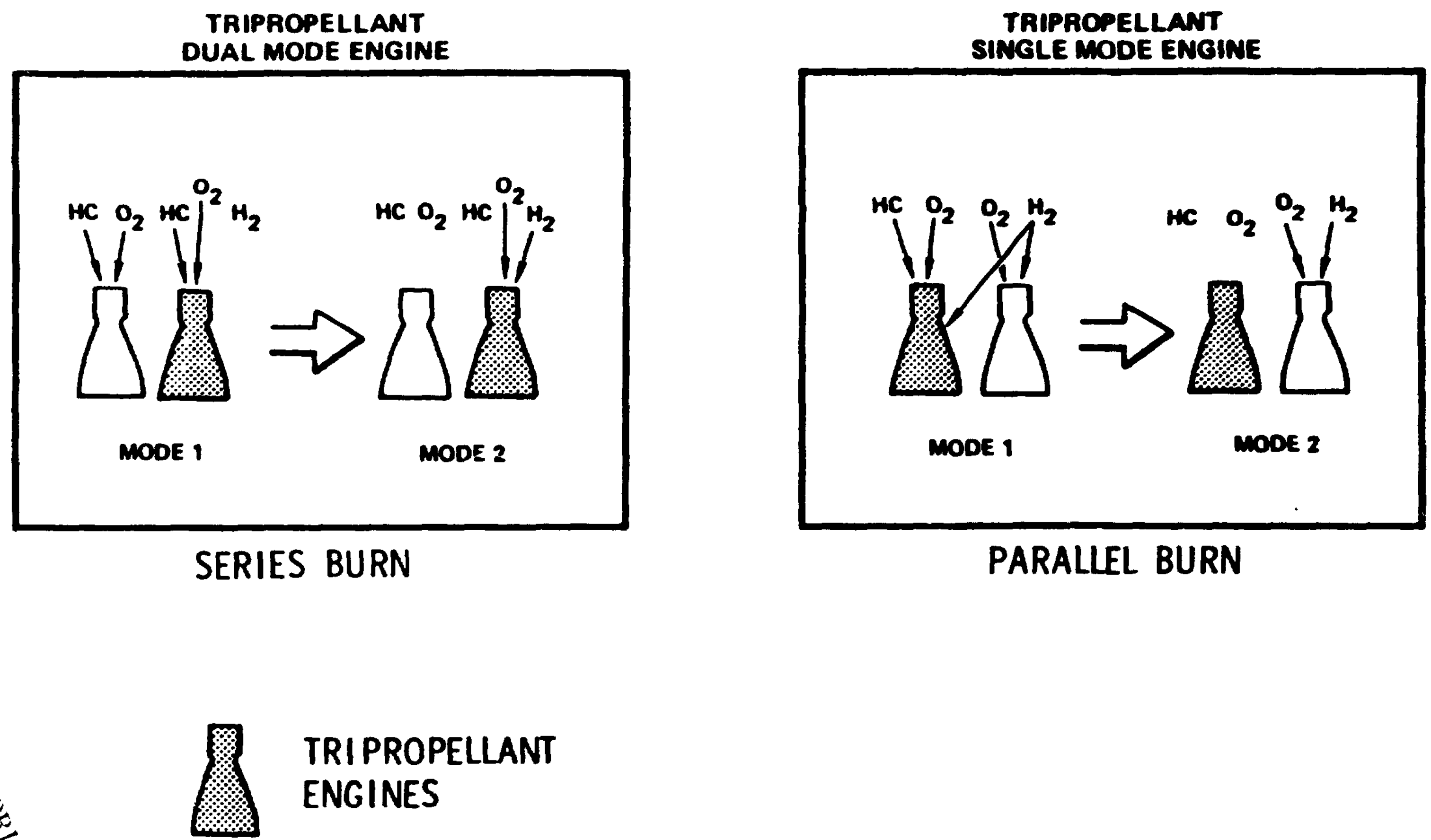

Figure 24. Tripropellant Engine Operating Modes

- POWER CYCLE TYPES:
  - STAGED COMBUSTION
  - GAS GENERATOR
- PROPELLANTS
  - MODE 1 LOX/RP-1, LOX/CH_4, LOX/C_3H_8
  - MODE 2 LOX/H_2
- ALL ENGINES CAPABLE OF OPERATING IN A SERIES BURN
- P_c = 3000 TO 4000 PSIA

  ϵ = 35:1 MODE 1

  = 150:1 MODE 2
- COOLANTS:
  - LH_2, CH_4, C_3H_8, LO_2
- TURBINE INLET TEMP TO 2000 R

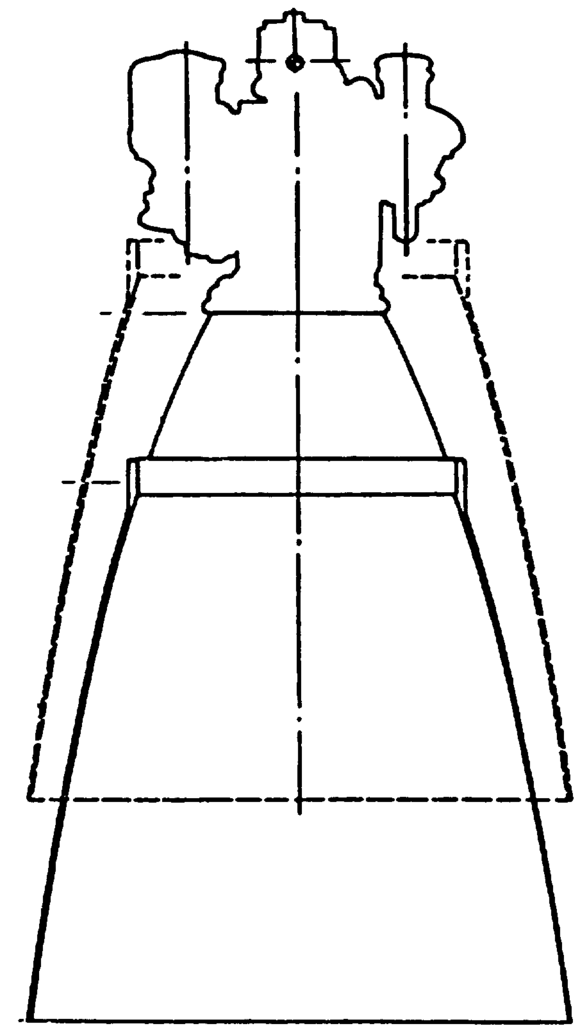

Figure 25. Ground Rules for Candidate Dual-Mode Cycle Selection

CONCEPT NO.	PROPELLANTS	CYCLE TYPE	COOLANT	
1	LOX/RP-1/H_2	S.C.	H_2	LOX PREBURNER LOX HIGH RICK LOX/RP-1 FUEL & H_2 PREBURNERS FUEL RICH LOX/RP-1
1A	LOX/RP-1/H_2	S.C.	H_2	ALL PREBURNERS FUEL RICH
2	LOX/RP-1/H_2	S.C.	H_2	ALL PREBURNERS OX RICH LOX/RP-1
3	LOX/CH_4/H_2	S.C.	H_2	LOX PREBURNER LOX RICH LOX/CH_4 FUEL & H_2 PREBURNERS FUEL RICH LOX/CH_4
3A	LOX/CH_4/H_2	S.C.	H_2	ALL FUEL RICH PREBURNERS
4	LOX/CH_4/H_2	S.C.	H_2	ALL PREBURNERS OX RICH LOX/CH_4
5	LOX/C_3H_8/H_2	S.C.	H_2	LOX PREBURNER LOX RICH LOX/C_3H_8 FUEL H_2 PREBURNERS FUEL RICH LOX/C_3H_8
5A	LOX/C_3H_8/H_2	S.C.	H_2	ALL PREBURNERS FUEL RICH
6	LOX/C_3H_8/H_2	S.C.	H_2	ALL PREBURNERS OX RICH LOX/C_3H_8
7	LOX/RP-1/H_2	G.G.	H_2	FUEL RICH G.G. LOX/H_2
8	LOX/CH_4/H_2	G.G.	H_2	FUEL RICH G.G. LOX/H_2
9	LOX/C_3H_8/H_2	G.G.	H_2	FUEL RICH G.G. LOX/H_2
10	LOX/RP-1/H_2	S.C.	O_2	LOX PREBURNER LOX RICH LOX/RP-1 FUEL PREBURNER FUEL RICH LOX/RP-1
10A	LOX/RP-1/H_2	S.C.	O_2	ALL PREBURNERS FUEL RICH
11	LOX/RP-1/H_2	S.C.	O_2	ALL PREBURNERS OX RICH LOX/RP-1
12	LOX/CH_4/H_2	S.C.	CH_4/H_2	ALL PREBURNER OX RICH LOX/CH_4
12A	LOX/CH_4/H_2	S.C.	CH_4/H_2	ALL PREBURNERS FUEL RICH
13	LOX/C_3H_8/H_2	S.C.	C_3H_8/H_2	ALL PREBURNER OX RICH LOX/C_3H_8
13A	LOX/C_3H_8/H_2	S.C.	C_3H_8/H_2	ALL PREBURNERS FUEL RICH
14	LOX/CH_4/H_2	G.G.	CH_4/H_2	FUEL RICH G.G. LOX/CH_4
15	LOX/C_3H_8/H_2	G.G.	C_3H_8/H_2	FUEL RICH G.G. LOX/C_3H_8

Figure 26. Candidate Engine Concepts

margin, thus requiring an additional preburner fuel bypass flow. Probably a more desirable means of incorporating this power margin capability is to use all of the fuel available, thus reducing the gas temperature and increasing the total gas flowrate. One other approach is to reduce the operating chamber pressure, thus reducing turbine power requirements and fuel flow to the preburners.

The required reduction in chamber pressure is excessive so the initial ground rules cannot be met and the resulting system is undesirable. However, because of the importance of these all-fuel-rich preburner systems in any future studies concerning dual-fuel engines, it was decided to carry these systems through the engine balance calculations to show the magnitude of the turbine inlet temperature increase or the chamber pressure reduction required to achieve an engine power balance. Therefore, candidate systems 2A, 3A, 5A, 10A, 12A, and 13A are shown in Fig. 26. Other staged combustion cycles incorporate either mixed fuel-rich and oxidizer-rich preburner or all LOX-rich preburners.

In all other staged combustion cases, a 2000 R turbine inlet temperature was incorporated in performing the engine balance calculation for either a fuel-rich or LOX-rich preburner. A turbine drive gas temperature of 2000 R may produce some areas of uncertainty with LOX-rich gases where the fuel-rich conditions would be more well known. A common temperature for both fuel-rich and LOX-rich preburners was selected for this study to maintain consistency for comparison purposes. It should be noted that further investigation is required pertaining to materials compatibility with 2000 R LOX-rich gases.

Engine Balances

The results of the mass/pressure/temperature balances for each of the candidate engine systems is presented in Table 6. All flowrates to the major engine system components are presented along with vacuum and sea level thrusts, specific impulse, and pump discharge pressure requirements. The LOX/RP-1 staged combustion systems with either LH_2 or LOX regenerative cooling are presented first. As previously mentioned, the all-fuel-rich preburner cases require turbine inlet temperatures above 2000 R. All of the available fuel is directed to the preburners and no power

TABLE 6. ENGINE BALANCES

O_2/RP-1/H_2 STAGED COMBUSTION

	Hydrogen-Cooled			Oxygen-Cooled		
	All PB's Fuel-Rich	Fuel & H_2 PB Fuel-Rich & O_2 PB O_2-Rich	All PB Ox-Rich	All PB's Fuel-Rich	Fuel PB Fuel-Rich & O_2 PB O_2-Rich	All PB Ox-Rich
CONCEPT NO.	1A	1	2	10A	10	11
Cycle Type	S.C.	S.C.	S.C.	S.C.	S.C.	S.C.
P_c	3230	3230	3230	3230	3230	3230
S.L. Thrust		460K	460K		470K	470K
Vac. Thrust		500K	500K		511.7K	511.7K
Propellants		O_2/RP-1	O_2/RP-1		O_2/RP-1	O_2/RP-1
Coolant		H_2	H_2		O_2	O_2
Turbine Drive Fluid		O_2/RP-1	O_2/RP-1		O_2/RP-1	O_2/RP-1
M.R.		2.8	2.8		2.8	2.8
I_S S.L., Sec		333.8	333.8		317.6	317.6
I_S Vac, sec		362.5	362.5		345.8	345.8
$T_{Turbine}$, R	2250	2000	2000	2100	2000	2000
Preburner/GG, lb/sec						
O_2		478.8	733.2		655.7	765.5
Fuel		240.3	22.2		144.5	23.2
H_2		-	-		-	-
$\dot{\omega}_{Turbine}$, lb/sec						
O_2		395.8	396.8		622.4	622.4
Fuel		174.7	187.3		177.9	166.2
H_2		147.6	171.3		-	-
$\dot{\omega}_{Coolant}$, lb/sec		34	34		225COMB 275NOZ	225COMB 275NOZ
$\dot{\omega}_{O_2}$ (Total), lb/sec		1045	1045		1090.4	1090.4
$\dot{\omega}_{Fuel}$ (Total), lb/sec		300.3	300.3		389.4	389.4
$\dot{\omega}_{H_2}$ (Total), lb/sec		34	34		-	-
$\dot{\omega}_{Total}$, lb/sec		1379	1379		1479.8	1479.8
$\dot{\omega}_{H_2}$ T.C., lb/sec		34	34		-	-
Pump Discharge Pressure						
LOX PB		7331	7331		7331	7331
Chamber		4123	4123		8600	8600
Fuel PB		7331	7331		7331	7331
Chamber		4123	4123		4123	4123
H_2		4000	4000		--	--

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TABLE 6. (Continued)

$O_2/CH_4/H_2$ STAGED COMBUSTION

	Hydrogen-Cooled			CH_4-Cooled	CH_4-Cooled	CH_4 Cooled
	Fuel-Rich	Fuel PB Fuel-Rich & O_2 PB O_2-Rich	All PB's O_2-Rich	All PB's O_2 Rich	All PB's Fuel-Rich	All PB's Fuel-Rich
CONCEPT NO.		3	4	12	12A	12B
Cycle Type	S.C.	S.C.	S.C.	S.C.	S.C.	S.C.
P_c	3230	3230	3230	3230	3230	2950*
S.L. Thrust		465K	465K	470K	470K	429.3
Vac. Thrust		508.6K	508.6K	516.2K	514.9K	473.3
Propellants		O_2/CH_4	O_2/CH_4	O_2/CH_4	O_2/CH_4	O_2/CH_4
Coolant		H_2	H_2	CH_4	CH_4	CH_4
Turbine Drive Fluid		O_2/CH_4	O_2/CH_4	O_2/CH_4	O_2/CH_4	O_2/CH_4
M.R.		3.5	3.5	3.5	3.5	3.5
I_S S.L., Sec		339.2	339.2	324	324	322
I_S Vac, sec		371	371	356	355	355
$T_{Turbine}$, R	2250	2000	2000	2000	2125*	2000
Preburner/GG, lb/sec						
O_2		533.8	945.9	659.2	162.2	126.1
Fuel		207.1	25.2	17.6	318.1	293.3
H_2		-	-	-	-	-
$\dot{\omega}_{Turbine}$, lb/sec						
O_2		459.9	459.9	378.6	190.7	165.7
Fuel		184.3	324.8	299	289.6	253.7
H_2		96.1	186.3	-	-	-
$\dot{\omega}_{Coolant}$, lb/sec		34	34	97.5 120	96.9 120.6	96.9 120.6
$\dot{\omega}_{O_2}$ (Total), lb/sec		1085	1085	1128	1127	1036.6
$\dot{\omega}_{Fuel}$ (Total), lb/sec		252	252	322	322	295.9
$\dot{\omega}_{H_2}$ (Total), lb/sec		34	34		-	-
$\dot{\omega}_{Total}$, lb/sec		1371	1371	1450	1449	1332
$\dot{\omega}_{H_2}$ T.C., lb/sec		34	34		-	-
Pump Discharge Pressure						
LOX PB		7331	7331	7331	7331	6470
Chamber		4123	4123	4123	4123	3760
Fuel PB		7331	7331	7331	7331	6470
Chamber		4123	4123	6064	-	-
H_2		4000	4000	--	-	-

*Minimum turbine inlet temperature or maximum chamber pressure that can be achieved if all available fuel is directed to preburners. In an actual engine some fuel must be reserved to cool the injector and hot gas manifold.

TABLE 6. (Continued)

$O_2/C_3H_8/H_2$ STAGED COMBUSTION

	Hydrogen-Cooled			C_3H_8 Cooled	
	Fuel-Rich	Fuel PB Fuel-Rich & O_2 PB O_2-Rich	All PB's O_2-Rich	All PB's Fuel Rich	All PB's O_2-Rich
CONCEPT NO.	5A	5	6	13A	13
Cycle Type	S.C.	S.C.	S.C.	S.C.	S.C.
P_c	3230	3230	3230	3230	3230
S.L. Thrust		464.6K	464.4K		470K
Vac. Thrust		507.6K	507.6K		515K
Propellants		O_2/C_3H_8	O_2/C_3H_8		O_2/C_3H_8
Coolant		H_2	H_2		C_3H_8
Turbine Drive Fluid		O_2/C_3H_8	O_2/C_3H_8		O_2/C_3H_8
M.R.		3.0	3.0		3.0
I_S S.L., Sec		335.7	335.7		320
I_S Vac, sec		366.7	366.7		35[illegible]
$T_{Turbine}$, R	2375	2000	2000	2275*	2000
Preburner/GG, lb/sec					
O_2		549.5	887.8		776
Fuel		247.6	26.1		22.8
H_2		-	-		-
$\dot{\omega}_{Turbine}$, lb/sec					
O_2		459.4	459.4		337.6
Fuel		206.7	267.2		461.2
H_2		131.0	187.2		-
$\dot{\omega}_{Coolant}$, lb/sec		34	34		140COMB 140NOZ
$\dot{\omega}_{O_2}$ (Total), lb/sec		1063.5	1063.5		1101
$\dot{\omega}_{Fuel}$ (Total), lb/sec		286.5	286.5		367
$\dot{\omega}_{H_2}$ (Total), lb/sec		34	34		
$\dot{\omega}_{Total}$, lb/sec		1384	1384		1468
$\dot{\omega}_{H_2}$ T.C., lb/sec		34	34		
Pump Discharge Pressure					
LOX PB		7331	7331		7331
Chamber		4123	4123		4123
Fuel PB		7331	7331		7331
Chamber		4123	4123		7064
H_2		4000	4000		--

TABLE 6. (Concluded)

GAS GENERATOR CYCLES

CONCEPT NO.	7	8	9	14	15	15A
Cycle Type	G.G.	G.G.	G.G.	G.G.	G.G.	G.G.
P_c	4000	4000	4000	4000	4000	3230
S.L. Thrust	470K	470K	470K	470K	470K	470K
Vac. Thrust	502K	504K	503.5K	505K	505.5K	505K
Propellants	O_2/RP-1	O_2/CH_4	O_2/C_3H_8	O_2/CH_4	O_2/C_3H_8	O_2/C_3H_8
Coolant	H_2	H_2	H_2	CH_4	C_3H_8	C_3H_8
Turbine Drive Fluid	O_2/H_2	O_2/H_2	O_2/H_2	O_2/CH_4	O_2/C_3H_8	O_2/C_3H_8
M.R.	2.8	3.5	3.0	3.5	3.0	3.0
I_S S.L., Sec	329.7	336.9	334	318.5	311.3	308.6
I_S Vac, sec	352.3	361.3	358	342.2	334.5	337.1
$T_{Turbine}$, R	2000	2000	2000	2000	2000	2000
Preburner/GG, lb/sec						
O_2	13.9	15.8	14.7	28.6	39.6	32.1
Fuel	-	-	-	66.5	90.1	72.9
H_2	17.4	19.7	18.3	-	-	
$\dot{\omega}_{Turbine}$, lb/sec						
O_2	18.5	18.9	18.5	47.8	61.6	51.1
Fuel	5.4	8.5	6.9	47.2	68.2	53.9
H_2	7.5	7.5	7.5	-	-	
$\dot{\omega}_{Coolant}$, lb/sec	36	36	36	125COMB 150NOZ	200 COMB 175 NOZ	160 COMB 175 NOZ
$\dot{\omega}_{O_2}$ (Total), lb/sec	1056	1084	1058	1102	1072	1094.8
$\dot{\omega}_{Fuel}$ (Total), lb/sec	335.7	277	314	373	434.6	427.2
$\dot{\omega}_{H_2}$ (Total), lb/sec	34	34	34			
$\dot{\omega}_{Total}$, lb/sec	1425	1395	1407	1476	1508	1523.1
$\dot{\omega}_{H_2}$ T.C., lb/sec	17	14.3	15.7		-	
Pump Discharge Pressure						
LOX PB Chamber	5106	5106	5106	5106	5106	4123
Fuel PB Chamber	5106	5106	5106	5466 8606	5107 11300	4123 6723
H_2	6084	6084	6084	-	-	

margin remains. Since the primary fuel (RP-1) is not used for regenerative cooling, some of the fuel does not have to bypass the preburner to provide cooling for the hot-gas manifold and injector face. In those cases where LH_2 is used as the coolant, the H_2 is directed to the main injector after exiting the cooling jacket, thus resulting in a higher specific impulse than if only the LOX/hydrocarbon were combusted.

For the mixed fuel and LOX-rich or all-LOX-rich preburner cases, there is always sufficient fuel available above the preburner requirement for auxiliary cooling of the injector face and hot-gas manifold.

Pump discharge pressures are generally in a range comparable with the SSME pump designs. The discharge pressures of the fuel and oxidizer flows to the preburners and directly to the main chambers are shown to aid in the SSME pump adaptibility studies.

The LOX/CH_4 staged combustion cases are shown in the second part of Table 6. Again, the all-fuel-rich preburners require turbine inlet temperatures exceeding current hardware capabilities. An additional variation of case 12A is presented where the chamber pressure is reduced to a level where an engine balance can be achieved with all-fuel-rich preburners and turbine inlet temperature of 2000 F. Cases 12A and 12B are only of theoretical interest in that they are limiting cases. In an actual engine system, some of the fuel flow must be reserved for cooling the injector face and hot-gas manifold. Also, no power margin capability exists unless the gas temperature is increased.

The maximum achievable chamber pressure for 2000 R turbine inlet temperature and the required turbine inlet temperature for 3230 psia chamber pressure are shown in Fig. 27 as a function of percent CH_4 reserved for auxiliary cooling. Estimates based on SSME design experience indicate that approximately 15 to 20% of the fuel must bypass the preburner for auxiliary cooling requirements. This results in an actual engine operating chamber pressure of 2300 to 2400 psia at 2000 R turbine inlet temperature, or a turbine inlet temperature of 2250 to 2300 R for a chamber pressure of 3230 psia.

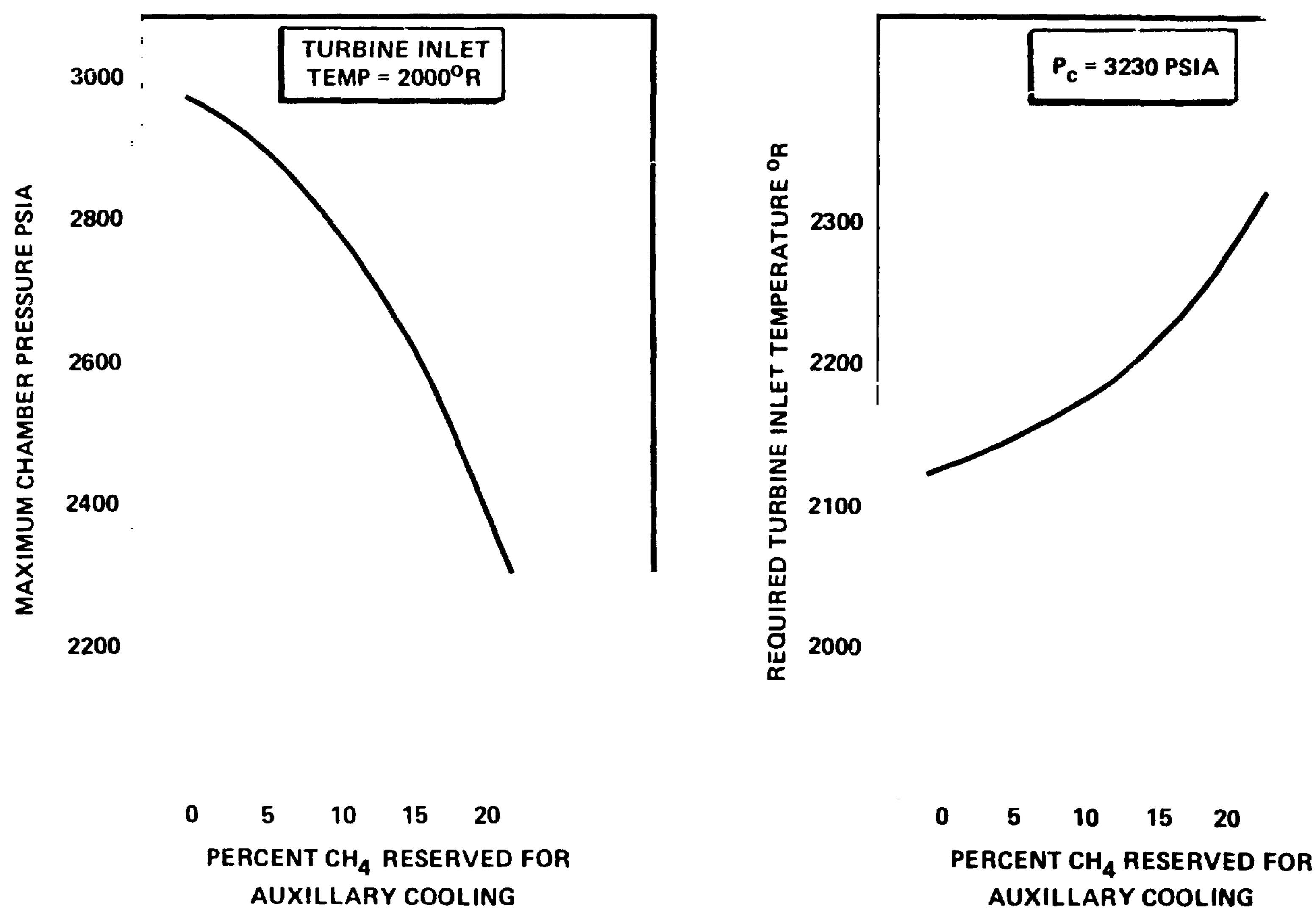

Figure 27. Maximum Chamber Pressure and Required Turbine Inlet Temperatures For LOX/CH_4 SSME Operating With Fuel-Rich Preburners

The engine balances for the staged combustion LOX/C_3H_8 systems exhibit the same trends as seen for the LOX/CH_4 systems except for slight differences in preburner flowrates resulting from the lower energy LOX/C_3H_8 preburner combustor products. Again, all-fuel-rich preburners are not feasible at 2000 R turbine gas temperature and 3230 psia chamber pressure.

The gas generator system balances are presented in the last part of Table 6. In general, these systems are not power limited as relatively small amounts of fuel are required for the gas generator feeding the higher pressure ratio turbines. It was also determined that gas generator engine system chamber pressures of 4000 psia could be achieved with pump discharge pressures appoximating those of a staged combustion cycle at 3230 psia chamber pressure. For this reason, all gas generator system balances were conducted at 4000 psia chamber pressure. The C_3H_8 cooling jacket ΔP at 4000 psia resulted in an excessively high fuel pump discharge pressure; therefore, a blanace also is shown for 3230 psia.

System Schematics

Preliminary engine flow schematics were generated for each of the engine system concept types. These schematics show the flow paths required for both mode 1 and 2 operation but all control components (valves, check valves, etc.) are not shown. A separate study conducted within Task IV established control requirements and necessary components. In some cases, the flow schematics were changed in the control studies to minimize the necessary control valves or simplify the system. It was found that in some cases, it was necessary to indicate isolation valves for the purpose of clarity. Boost pump drive methods were maintained as in the SSME where possible. In those cases where additional pumps are required or concepts do not permit the same boost pump drive technique, a logical alternative is selected, but is not necessarily the only method that might be considered.

A schematic representative of systems 1 through 6 is shown in Fig. 28. All three turbopumps must operate during mode 1 as the combustion chamber and nozzle are H_2 cooled in both mode 1 and mode 2. The H_2 bypass from the pump to the main chamber and the H_2 flow to the preburners is required only during mode 2 operation.

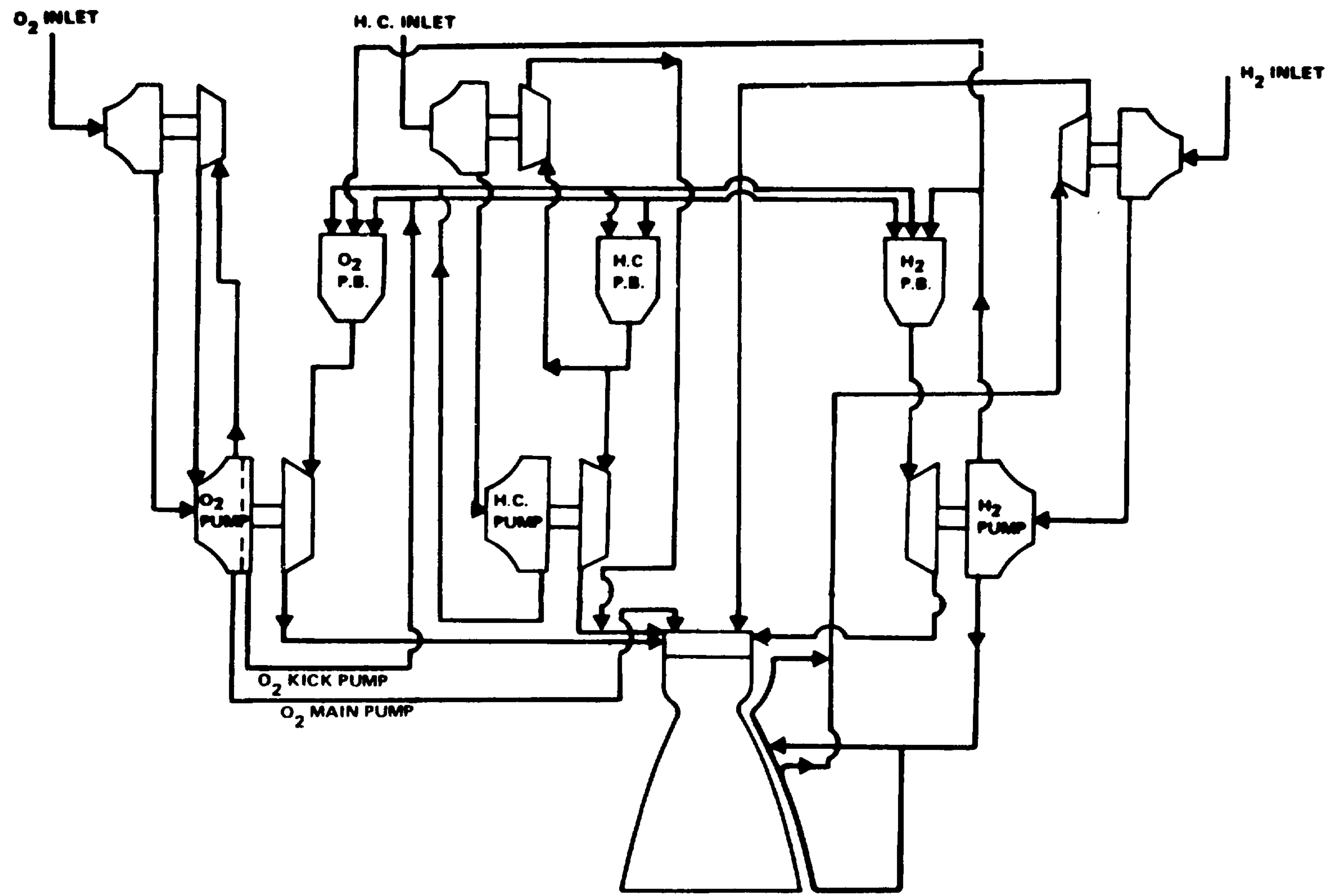

Figure 28. Dual-Mode, Hydrogen-Cooled, Staged Combustion Flow Schematic for Concepts No. 1 Through 6

Pump studies have shown that it is questionable whether one pump (SSME main H_2 pump) can satisfy both mode 1 and 2 H_2 head and flow requirements. The hydrocarbon (HC) pump operates only during mode 1. The LOX highpressure or kick pump stage is required to feed the preburners in both mode 1 and mode 2 operation. The HC boost pump is assumed to be driven in parallel with the HC main pump by preburner gases. The LOX and H_2 preburners must operate on either LOX/HC or LOX/H_2, as does the main chamber.

System concepts No. 10 and 11, as shown in Fig. 29, are quite similar to No. 1 and 2 except that they are LOX cooled instead of H_2 cooled. The lack of H_2 being injected into the main chamber explains the difference in performance. The LOX coolant requirements for the chamber and nozzle are lower than those required for the preburners and the remainder of the LOX flow is not sufficient for the preburner. Therefore, some of the nozzle coolant flow (GOX) is mixed with the LOX being fed to the preburners. This has the added advantage of heating up the LOX before it is injected into the chamber with the hydrocarbon and reducing the possibility of an explosive gel forming.

The schematic for systems 12 and 13 is shown in Fig. 30. These systems are similar to systems 1 through 6 except that No. 12 and 13 are hydrocarbon cooled during mode 1 and H_2 cooled during mode 2, while No. 1 through 6 are H_2 cooled in both modes. The fuel and the H_2 boost pumps are driven by the nozzle coolant discharge flow. Isolation valves are shown in the boost pump drive gas delivery lines since only the HC pump operates during mode 1 (with gaseous HC turbine drive gas) and only the H_2 pump operates during mode 2 (GH_2 turbine drive gases). This technique of driving the fuel and H_2 boost pump is only one of several possible; hydraulic turbines or driving them in parallel with the main pumps with the preburner flow also could be considered. The switch of chamber and nozzle cooling from a hydrocarbon in mode 1 to H_2 in mode 2 results in significant system and operational complexities.

The schematic for system concepts 7, 8, and 9 is shown in Fig. 31, and is typical of a low pump discharge pressure gas generator cycle. The chamber and nozzle are H_2 cooled in both modes 1 and 2 and the chamber coolant flowrate is adequate to

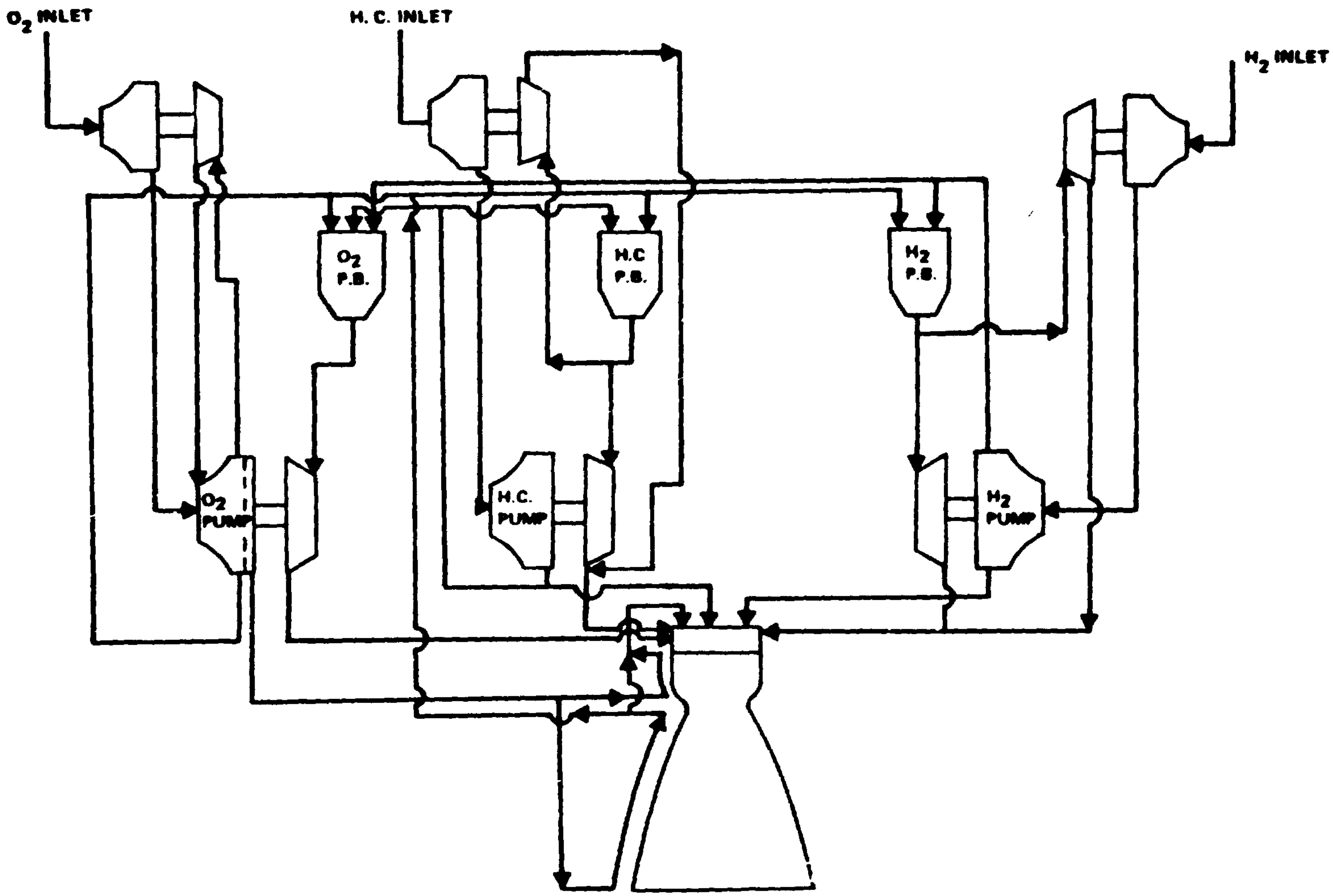

Figure 29. Dual-Mode, Oxygen-Cooled, Staged Combustion Flow Schematic for Concepts No. 10 and 11

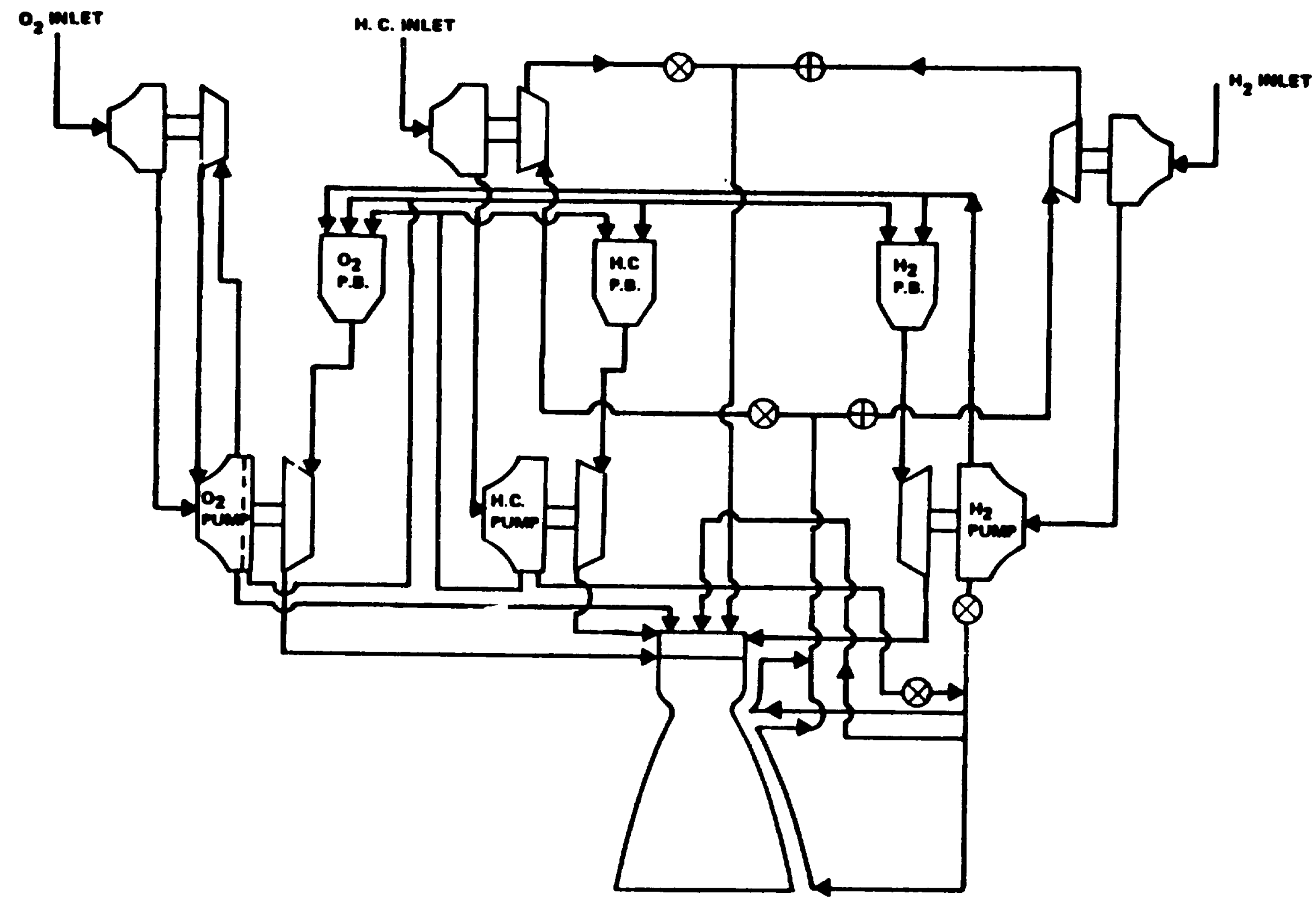

Figure 30. Dual-Mode, Hydrocarbon, H_2-Cooled, Staged Combustion Flow Schematic for Concepts No. 12 and 13

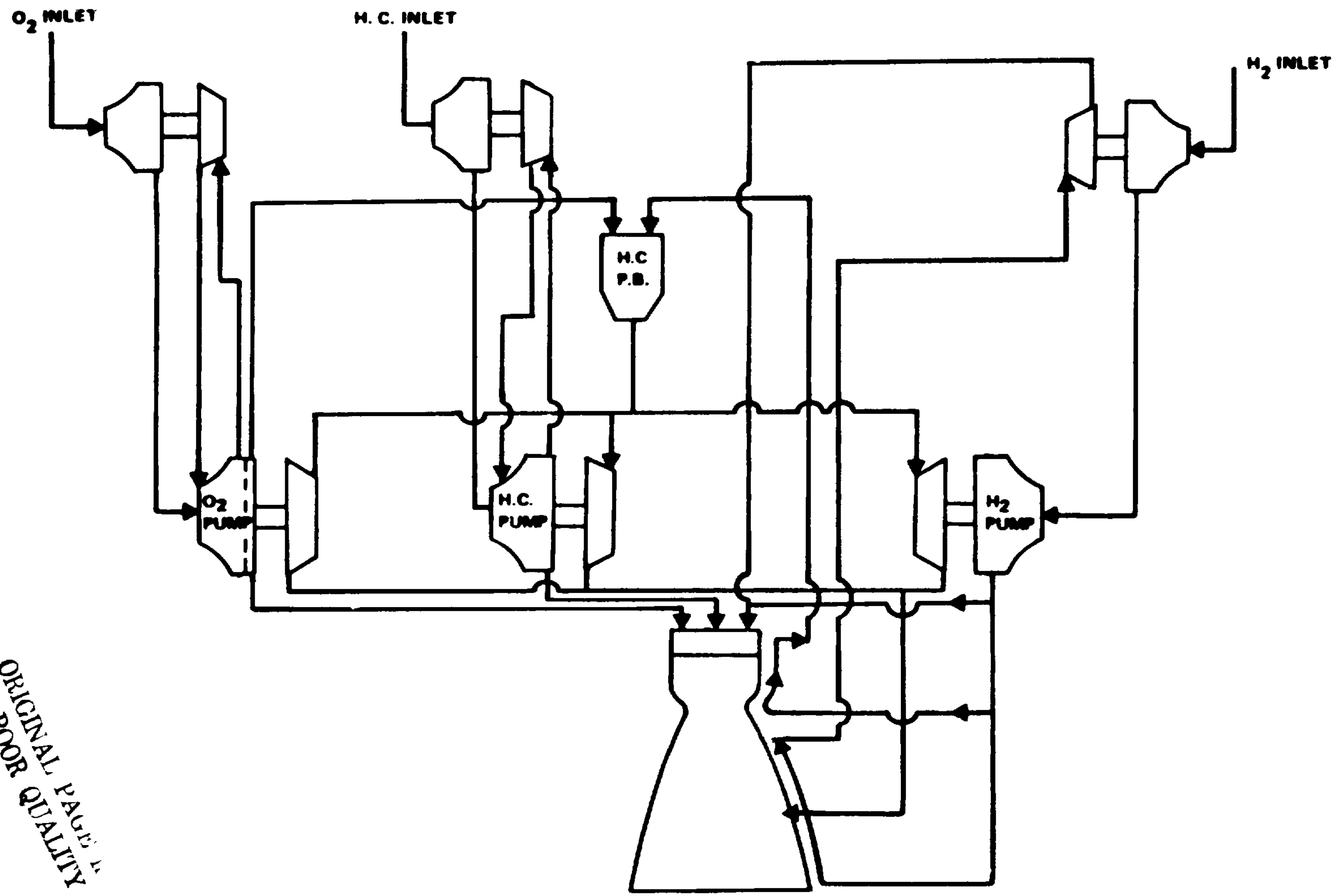

Figure 31. Dual-Mode Hydrogen-Cooled, Gas Generator Flow Schematic for Concepts No. 7, 8, and 9

satisfy the gas generator requirements. Again, in this case, the H_2 flowrates are greatly different between modes 1 and 2 and the H_2 bypass from the pump to the chamber is needed only during mode 2. It is questionable whether the SSME H_2 main pump can satisfy this requirement. All of the turbines are driven in parallel with a single gas generator and the turbine exhaust pressures are collected and ducted into the main nozzle where the expansion pressure matches the turbine exhaust pressure (approximately 200 psia). In this case, the HC boost pump is driven by the HC main pump through a hydraulic turbine, as is the case for the SSME LOX boost pump. The H_2 coolant flow through the nozzle supplies the hot-gas flow to drive the H_2 boost pump.

System concepts 14 and 15 (shown schematically in Fig. 32) are, again, gas generator cycles but differ from No. 7, 8, and 9 in that the chamber and nozzle are hydrocarbon cooled during mode 1 and H_2 cooled during mode 2. The gas generator fuel is supplied by the combustion chamber coolant flow; therefore, the gas generator fuel as well as the main chamber fuel changes between mode 1 and 2. The low-pressure booster pumps are driven by the respective nozzle coolants during mode 1 and mode 2.

System Variations

Mass flow balances shown in Table 6 for system concepts No. 1, 3, 4, 5, and 6 result in a large percentage of either the oxidizer or fuel being combusted in the preburner and the remaining smaller percentage being bypassed directly to the main combustor. This occurs as a result of fixing the turbine pressure ratio, turbine inlet gas temperature, and chamber pressure (pump discharge pressure). This results in additional main chamber injector complexity since a third fluid must then be injected into the chamber, and where this third flow is relatively small, it would simplify the system if the preburner flows could be increased so that either all of the available oxidizer or fuel could be fed through the preburner. Sufficient fuel must be reserved to satisfy the auxiliary cooling requirements.

In system concepts 4 and 6, all preburners are operated oxidizer rich with a 2000 R turbine inlet temperature and a turbine pressure ratio of 1.6:1. This temperature was selected for the oxidizer-rich preburners to maintain consistenty in the study.

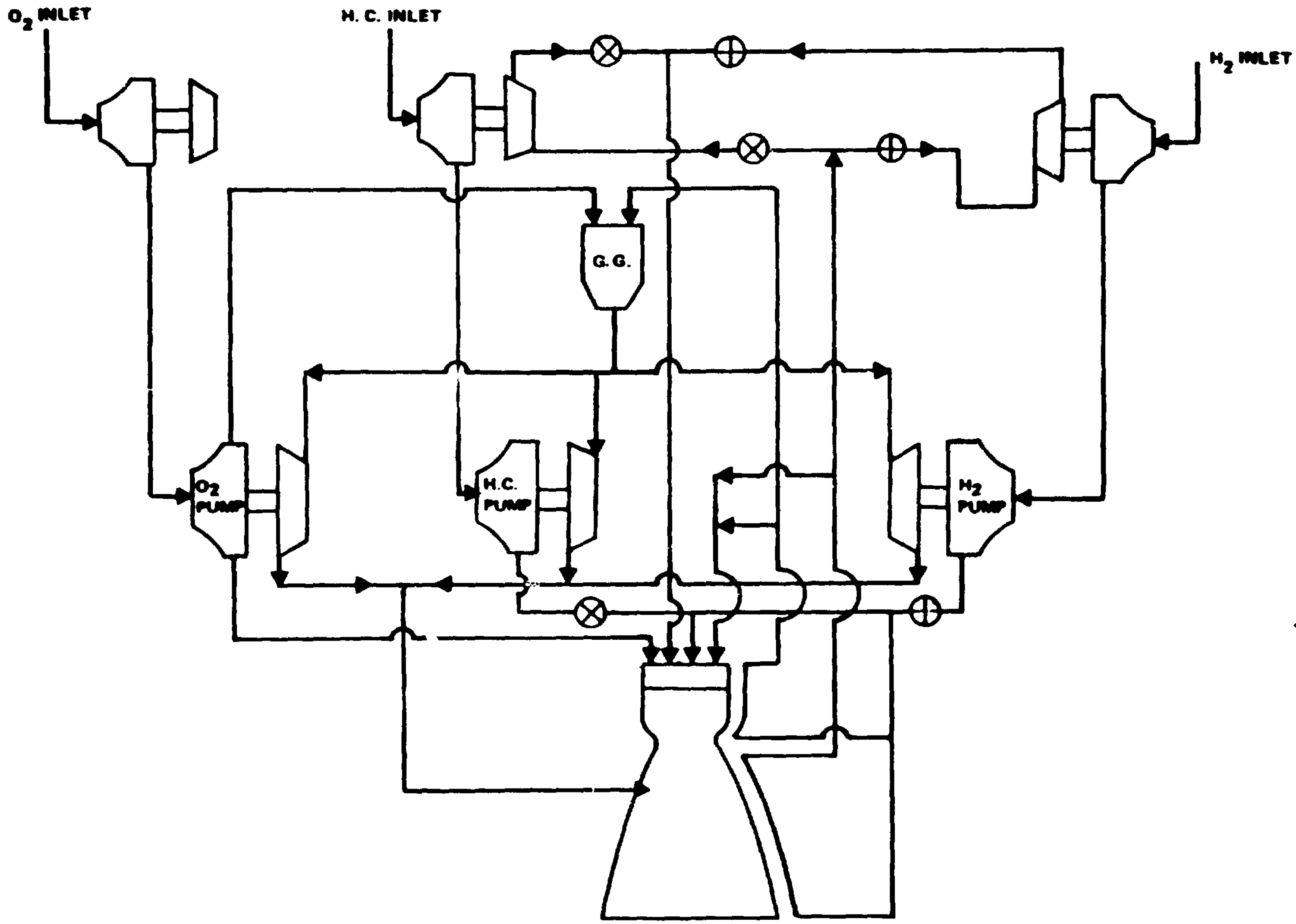

Figure 32. Dual-Mode Hydrocarbon Hydrogen-Cooled, Gas Generator Schematic for Concepts No. 14 and 15

However, materials compatibility with high-temperature LOX-rich gases could have some impact on the selection of this temperature. The resulting engine cycle balances require approximately 87% of the available O_2 flow to the preburners. Themain chamber injector adaptation would be easier if all of the oxidizer were routed through the preburner, thus eliminating the need for the liquid oxygen delivery to the main chamber. Several alternatives exist as to the possible utilization of this excess oxygen flow to the preburners.

1. The excess oxygen flow could be distributed to the three preburners, thus reducing turbine inlet temperature. This is probably the preferred method of utilizing the excess oxygen. An iterative flow balance would be required to determine how much this temperature could be reduced since pumping requirements are changing along with turbine drive gas properties.

2. The turbine flows could be increased at the same mixture ratio and temperature, thus producing more horsepower and permitting a higher chamber pressure.

3. The turbine flows could be increased, with the same mixture ratio and temperature, with a reduced turbine pressure ratio, thus producing the same horsepower at the higher flow.

Mode 2 LOX/H_2 Engine Balance

The mode 2 LOX/H_2 engine operating capabilities can be impacted by the mode 1 operating configuration. The ideal situation would be to have the mode 2 engine configuration and operation identical to the current SSME. This means LH_2 regenerative cooling with both preburners fuel rich. It is also necessary, from a practical standpoint, that the cycle configuration be maintained from mode 1 to mode 2. This, coupled with the fact that most of the candidate mode 1 engine systems have turbine drive cycle configurations different than those of the SSME, means that other cycle configurations must be examined for the LOX/H_2 mode 2 operation.

Engine system mass/pressure/temperature balances were calculated for several LOX/H_2 mode 2 operating configuration and the results are shown in Table 7. Both staged combustion and gas generator cycles are presented.

Two staged combustion cycles were considered. The case with both preburners fuel rich is taken directly from current SSME FPL performance predictions.

The other case incorporates a LOX-rich LOX preburner and a fuel-rich fuel preburner. This represents the mode 2 operating conditions required for those systems that incorporate mixed fuel and oxidizer-rich preburners in mode 1. This turbine drive configuration is feasible from a flowrate energy availability standpoint, but is not compatible with the SSME main chamber and preburner injectors.

An additional mode 2 staged combustion LOX/H_2 system was investigated that would incorporate LOX-rich preburners. This mode 2 system would correspond to those mode 1 systems that operate with LOX-rich preburners. This would eliminate the main chamber injector difficulties that occur when switching operation from mode 1 to mode 2. It was found that to operate the preburners LOX rich at 2000 R temperature, a mixture ratio of 86:1 is required and there is sufficient LOX flow in the system to achieve a power balance. Approximately 1840 lb/sec of LOX are required in the preburner with only 965 lb/sec available. Therefore, it was concluded that a LOX/H_2 staged combustion engine system at SSME conditions was not feasible with LOX-rich preburners.

An analysis also was conducted to establish an engine balance for mode 2, LOX/H_2 SSME with LOX cooling. However, with the limiting chamber pressure value of 2500 psia and the cooling ΔP established in the cooling analysis, the required LOX pump discharge pressure is excessively high and it was determined that operating the SSME with LOX cooling is not a practical alternative. This, in essence, eliminates LOX cooling for the dual-mode applications. The SSME hardware constraint plays a minor role in this conclusion and it is believed that LOX cooling would not be practical in any dual-mode engine system.

TABLE 7. LOX/H_2 ENGINE BALANCE FOR MODE 2

	ALL P.B.'S FUEL RICH STAGED COMBUSTION	FUEL RICH FUEL P.B. LOX RICH LOX P.B. STAGED COMBUSTION	GAS GENERATOR	GAS GENERATOR
P_c	3230	3230	4000	3230
S.L. Thrust, lb	379.5K	379.5K	373.6K	343.4K
Vac. Thrust	525.5K	525.5K	513.8K	517.5K
Propellants	O_2/H_2	O_2/H_2	O_2/H_2	O_2/H_2
Turbine Drive	O_2/H_2	O_2/H_2	O_2/H_2	O_2/H_2
M.R. (T.C.)	6:1	6:1	6:1	6:1
I_s S.L.	337	337	331.9	304.9
I_s Vac.	466.7	466.7	456.3	459.5
$T_{Turbine}$, R				
H.P. Ox	1860	2000	2000	2000
H.P. Fuel	1932	2000	2000	2000
Preburner, or Gas Generator Flowrates, lb/sec				
Oxid				
O_2	32.3	465.3	4.88	3.94
H_2	36.1	5.4	6.1	4.93
Fuel				
O_2	85.8	75.2	23.4	19.0
H_2	87.2	94.0	29.3	15.2
$\dot{W}_{Turbine}$				
H.P. Ox	68.4	470.7	10.98	8.87
H.P. Fuel	173.0	169.2	52.7	34.2
$\dot{W}_{Coolant}$				
Nozzle	54.5	54.5	77.9	54.5
Combustor	32.0	32.0	56.6	32.0
$\dot{W}_{O2}$, Lb/Sec	965	965	965	965
$\dot{W}_{H2}$, Lb/Sec	161	161	161	161
$\dot{W}_{Total}$, Lb/Sec	1126	1126	1126	1126
Pump Discharge Pressure, psia				
H.P. Lox	4972	4972	5106	4124
Lox Kick	3050	3050		
H.P. H_2	5939	5939	9032	5854

Power balances also are presented for two gas generator cycle versions of the LOX/H_2 mode 2 engines. The current SSME fuel and oxidizer flowrates were assumed for both the 3230 and 4000 psia chamber pressure cases. This implies that a different thrust chamber with a smaller throat diameter is employed for the 4000 psia chamber.

Summary of Results

The results of the engine system balance analysis have produced the following conclusions:

1. For the LOX/hydrocarbon mode 1 candidate systems, there is insufficient energy (fuel flowrate) available to obtain a stage combustion power balance with all preburners fuel rich at a chamber pressure of 3230 psia and a turbine inlet temperature of 2000 R.
2. LOX/hydrocarbon staged combustion cycle power balances can be achieved for all-fuel-rich preburners with sufficient fuel reserved for auxiliary cooling at reduced chamber pressures ($\sim$2300 psia) or increased turbine inlet temperatures ($\sim$ 2275 F).
3. LOX/hydrocarbon staged combustion power balance is achievable with both preburners LOX rich or LOX-rich LOX turbine and fuel-rich fuel turbine.
4. There is insufficient energy (LOX flowrate) available to obtain a staged combustion power balance with all preburners LOX rich for the mode 2 LOX/H_2 operation.
5. Mode 2, LOX/H_2 operation with LOX cooling is limited to a maximum chamber pressure of $\sim$2500 psia with an excessively high pump discharge pressure. For this reason, LOX cooling is not practical for dual-mode applications.
6. Gas generator cycles are not power limited.
7. Mode 1 or mode 2 hydrogen coolant from either the chamber or nozzle is adequate for fuel flow to the gas generator. This also provides minimum turbine drive performance loss.

TASK IV - CONTROL SYSTEM REQUIREMENTS

The start sequence and control system selection for the tripropellant engine are patterned after the Space Shuttle Engine (SSME). During normal power level, it takes the SSME (a closed-loop controlled stage combustion engine) approximately 3.6 seconds to attain 90% of rated thrust (Fig. 33). With closed loop control and proper selection of valve operating sequences, the start and cutoff transients of the tripropellant engine can be made to follow closely those of the SSME (Fig. 33 and 34).

It is intended that the tripropellant engine utilize SSME turbomachinery; therefore, valve opening schedules and utilization of open- and closed-loop control procedures are expected to be similar to those of the SSME. Engine start and shutdown criteria are indicated in Table 8.

Staged Combustion Cycle Concepts

The staged combustion cycle cooling options for the tripropellant engine are: hydrogen cooled in both modes 1 and 2, hydrocarbon cooled in mode 1 and hydrogen cooled in mode 2, and oxygen cooled. Oxygen cooling was ruled out from the standpoint of mode 2 operating limitations but was carried through the controls study to determine other aspects of the concept. There are 10 engine concepts (1-6 and 10-13) that fall within these three cooling categories. Schematics for these 10 concepts are shown respectively in Fig. 35 through 37. Required control valves are indicated in each schematic. Start and shutdown procedures and transients are similar for all three schematics.

Mode 1 Operation. The start sequence for the tripropellant LOX/hydrocarbon engine (concepts 1-6, 10-13) employs the open-loop control mode during early start phases and switches to closed-loop operation for buildup to rated thrust. Initial valve opening and sequencing provides ignition sequencing, engine priming, and initial turbine power buildup. Closed-loop control is then activated to achieve a start to the desired power level without transient overshoots or undershoots.

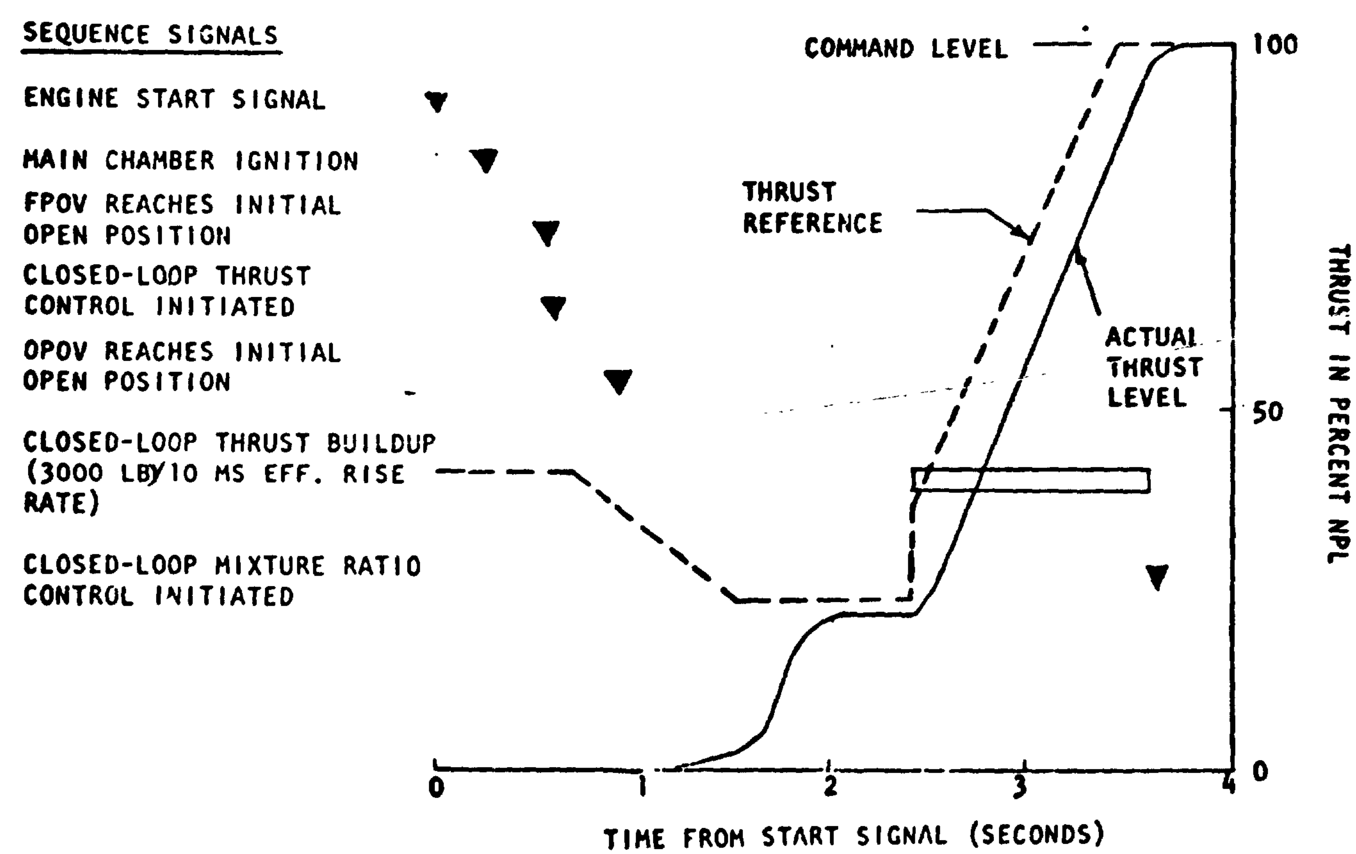

Figure 33. SSME Start Sequence to NPL

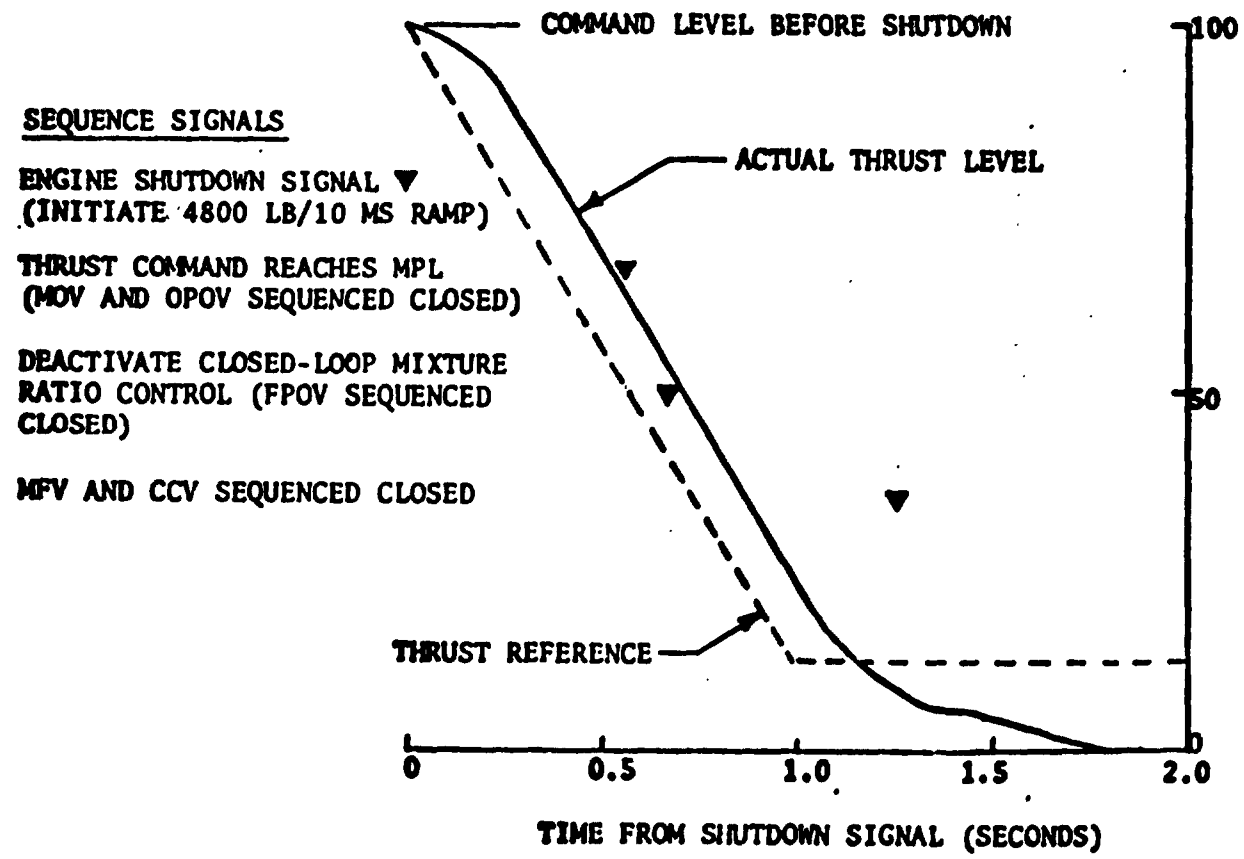

Figure 34. SSME Shutdown Sequence

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TABLE 8. ENGINE START AND SHUTDOWN CRITERIA

Prevent transient overshoots or undershoots.
Provide mixture ratio variations compatible with engine life and reliability.
Provide repeatable engine start characteristics to rated power levels.
Provide thrust accelerations within customer specification.
Provide thrust accelerations required to minimize side loads at sea level.
Provide start transients insensitive to vehicle and mission operation requirements.
Provide shutdowns without detrimental pump speed and turbine temperature transients.
Provide shutdowns with combustion of all fuel and oxidizer residuals without damaging mixture ratio transients.

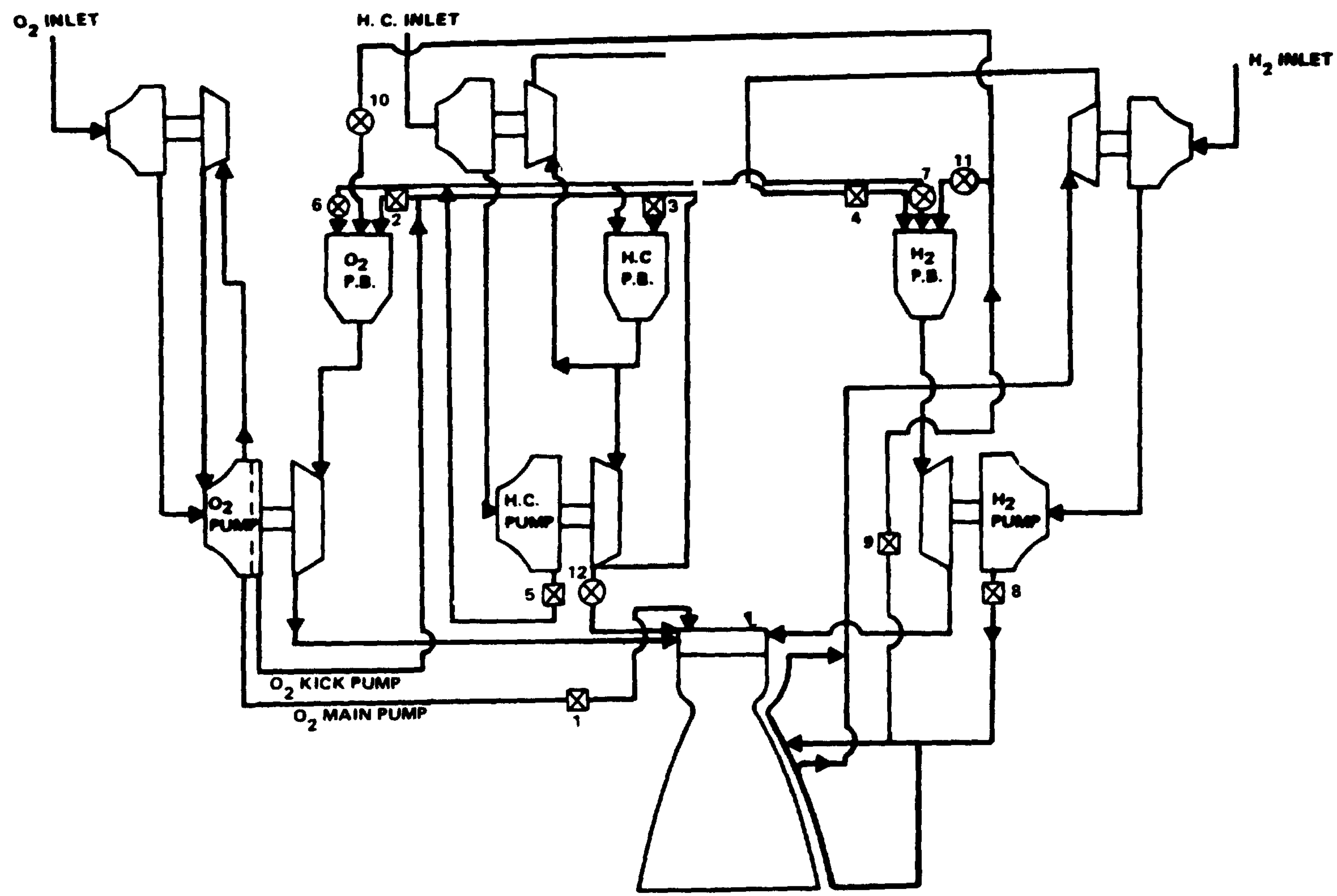


Figure 35. Dual-Mode, Hydrogen-Cooled, Staged Combustion Flow Schematic for Concepts No. 1 Through 6

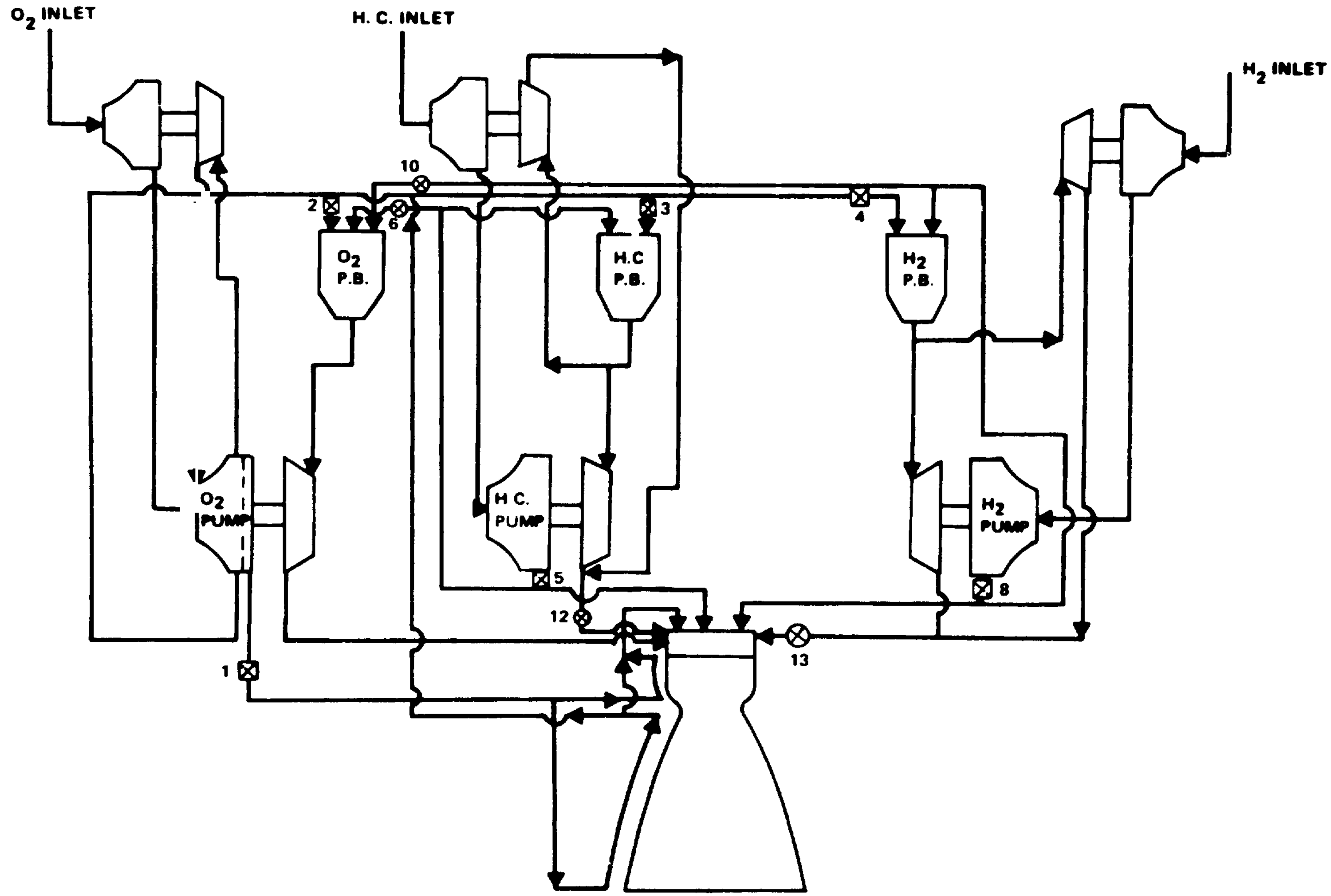


Figure 36. Dual-Mode, Oxygen-Cooled, Staged Combustion Flow Schematic for Concepts No. 10 and 11

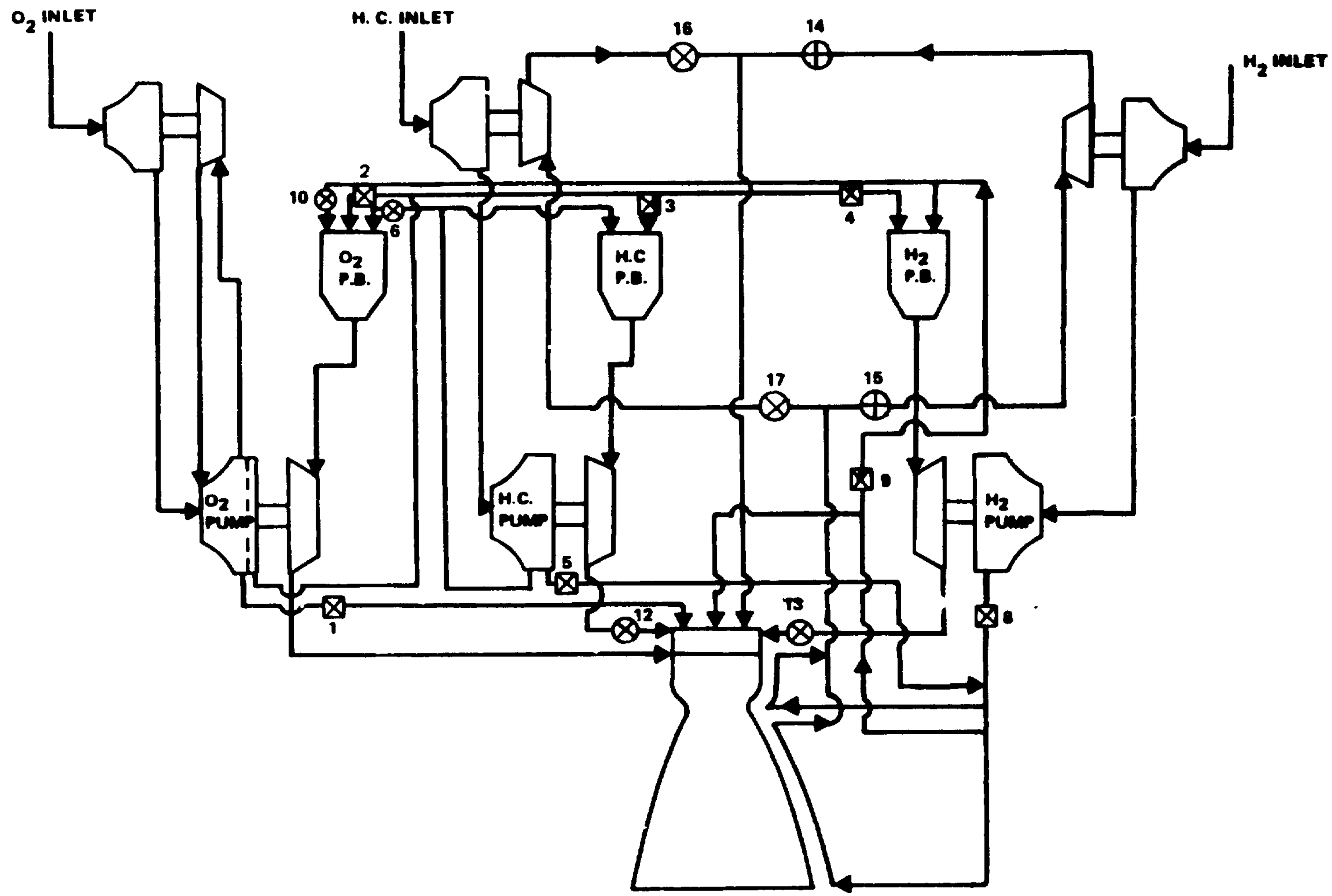


Figure 37. Dual-Mode, Hydrocarbon, H_2-Cooled, Staged Combustion Flow Schematic for Concepts No. 12 and 13

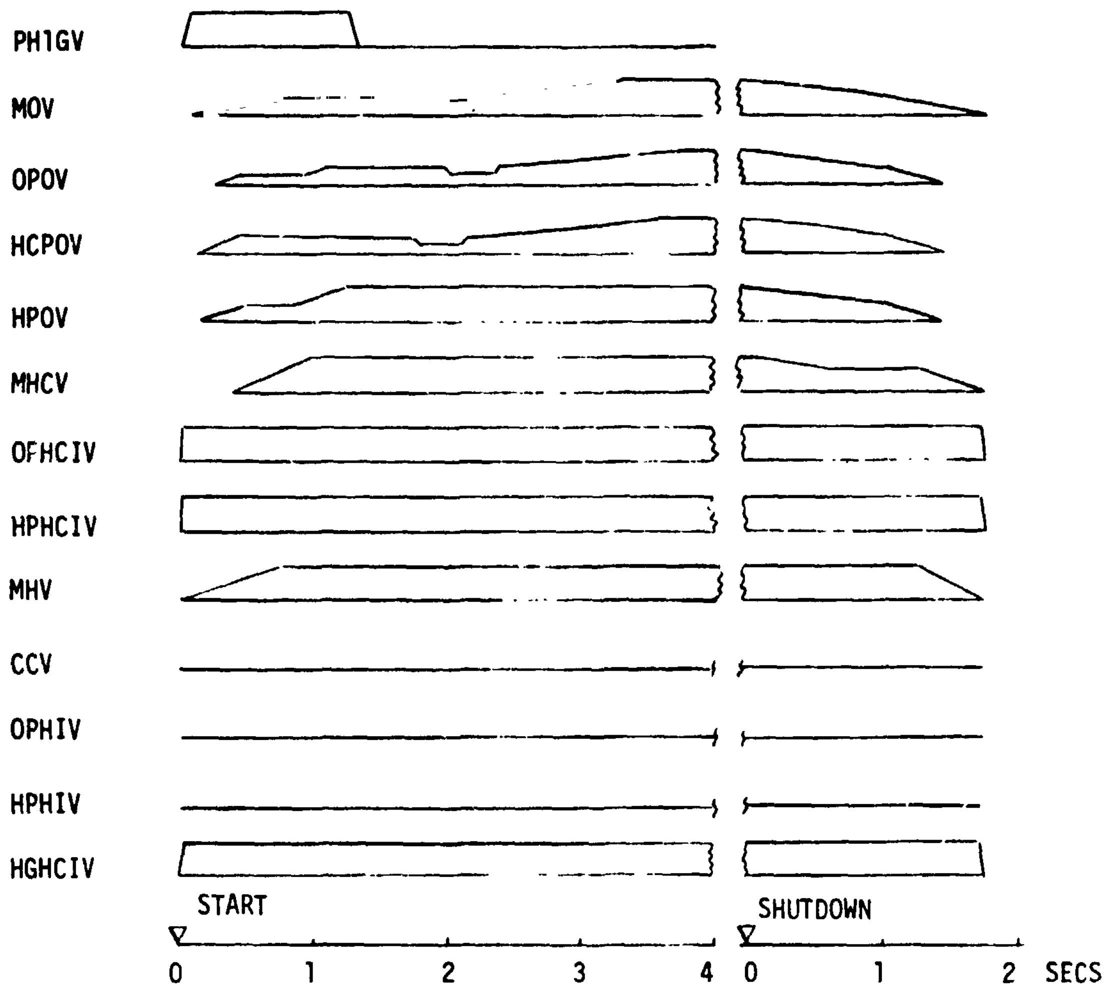
Open-Loop Control Mode. Start is initiated by a command from the vehicle. (Prestart procedures provide for removal of all vapor from engine passages above the main propellant valves and above the oxidizer preburner valves and inerting of propellant feed manifolds and coolant jackets.)

The start sequence (Fig. 38) starts with actuation of the main hydrogen valve (valve No. 8 in Fig. 35) to the full-open position and the preburner and main chamber hydrogen igniter valve. This establishes flow under tank pressure to systems downstream of the valve for priming, including the main combustors, preburners, and ignition system. Upon priming of the fuel systems, the main oxidizer valve (1)*, the oxidizer preburner oxidizer valve (2), the hydrocarbon fuel preburner oxidizer valve (3), and the hydrogen preburner oxidizer valve (7) begin to open, retracting the valve ball seats. Before main flow begins to build up from these valves, igniter element oxidizer flows past the valve ball seat and into the preburners, and main combustion chamber. Seal retraction of the oxidizer valves establishes hydrogen propellant flow in the igniter systems.

Propellant flow (hydrogen and oxygen) to the main chamber and preburner ignition units is ignited by a spark igniter unit at the main chamber and preburner, producing a hot-gas core for main (mode 1) propellant ignition at the injector approximately 300 milliseconds after the start signal is actuated. Initiation of LOX/H_2 pilot combustion early in the sequence ensures that the main hydrocarbon propellants of the main chamber and preburners ignite safely and that no raw propellants are dumped into the vehicle boattail during start.

Actuated a fraction of a second after the main hydrogen valve, the main oxidizer valve continues to open to approximately 60% of its full travel. Hydrocarbon, oxygen, and hydrogen preburner oxidizer valves are then ramped open to their intermediate 50% position. Immediately following, the hydrocarbon main valve (5) is ramped to the full-open position. This initiates preburner power buildup of the hydrocarbon turbomachinery, with the oxidizer turbomachinery power lagging slightly behind the hydrocarbon.

*Numbers in parentheses refer to valve number on appropriate schematic.



* Concepts 10 & 11: No HPHCIV, HPHIV, CCV } MHV Closed
Concepts 12 & 13: No HPHCIV, HPHIV

Figure 38. Start and Shutdown Sequence, Concepts 1-6, 10-13*, Mode 1

Hydrocarbon isolation valves on the oxidizer (6) and hydrogen (7) preburners are opened with the activation of the start signal and remain open during mode 1 operation, as well as the thrust chamber hot-gas hydrocarbon isolation valve (12). The oxidizer (10) and hydrogen (11) preburner isolation valves and the coolant control valves (5) remain closed during mode 1 operation. The first two valves prevent hot-gas hydrocarbon products from entering the hydrogen flow system during mode 1 operation.

Propellant flow to the main chamber and preburner ignition units is ignited by a spark igniter producing a hot-gas core for main propellant ignition at each injector 250 milliseconds after the engine start signal is actuated. Initiation of LOX/H_2 combustion early in the sequence provides assurance that no raw propellants are dumped into the vehicle boattail during start and that the main hydrocarbon propellants of the main chamber and preburners ignite safely.

Actuated at a fraction of a second after the main hydrogen valve, the main oxidizer valve continues to open to 62% of their travel. Shortly after, the hydrocarbon preburner oxidizer valve is ramped to the intermediate open position of 52%. Immediately after, the hydrocarbon main valve is ramped to the full-open position. This initiates preburner power to the intermediate open position of 52%. Immediately after, the hydrocarbon main valve is ramped to full-open position. This initiates preburner power buildup of the hydrocarbon turbomachinery.

The valve positions established by approximately 1 second set the engine power level at approximately 25% of rated power level. The transient to this thrust level provides preburner and main combustion chamber mixture ratio variations that do not degrade component life and reliability. The engine continues in this operating mode until 2.0 seconds. At this time, all engine start transients, including the slowest systems under the worst operating conditions, will have reached 25% of rated power level. When the thrust is increased from 25% to the final thrust level, all engine systems, regardless of environment, will respond in the same manner and with the same characteristics. The pre-established thrust acceleration rates conform with customer specifications and provide for minimization of side loads for sea level starts.

Closed-Loop Mode. Start buildup to the commanded thrust and mixture ratio levels is performed under closed-loop control. At approximately 0.75 second into the start transient, the oxidizer and fuel preburner oxidizer valve positioning controls are turned over to closed-loop thrust control. This procedure is selected to maintain the engine mixture ratio between the proper limits in the high-impulse range during the major portion of the thrust buildup. The commanded thrust level is achieved in approximately 4 seconds. This method achieves repeatable start characteristics with commanded thrust and mixture ratio achieved in the same time on every start.

Startup procedures are similar for concepts 1-6, 12, and 13 except for a few nonconsequential steps. The main hydrogen valve (7) remains closed during mode 1 operation since the engine is hydrocarbon cooled in mode 1. Hydrogen for the ignition system is obtained from upstream of the hydrogen valve. No hydrogen preburner hydrocarbon isolation valve (7) or hydrogen preburner hydrogen isolation valve (11) is required since the hydrogen preburner operates only once during the cycle.

During mode 1, the heated hydrogen isolation valves (14 and 15) remain closed while the heated hydrocarbon isolation valves (16 and 17) are open. With the main hydrogen valve closed, a portion of the hydrocarbon provides thrust chamber cooling and power for the low-pressure pump turbine before flowing into the thrust chamber injector.

Since concepts 10-11 utilize oxygen-cooled thrust chambers, no coolant control valves (9) have been included. As in concepts 12-13, the hydrogen preburner operates only once during the cycle and, therefore, does not require preburner isolation valves, hydrogen (7), and hydrocarbon (11). Also, no isolation valves (14-17) are required in the coolant circuits. Except for the above, the start procedures during mode 1 are identical to those of concepts 1-6.

Engine Shutdown. The engine achieves shutdown functions with the same elements used for start and mainstage control. The shutdown sequence (Fig. 38), by employing closed- and open-loop elements, provides repeatable shutdown transients that are insensitive to vehicle and mission operation requirements.

Mode 2 Operation. Mode 2 operation of the tripropellant engine is in the LOX/H_2 mode which is identical to SSME operation. Start and shutdown transients are as shown in Fig. 33 and 34. Valve sequencing and scheduling are as shown in Fig. 38. All the hydrocarbon valves (3, 5, 6, 7, 12, 16, 17) remain closed while all the hydrogen valves are operative.

Control Valve Requirements. Control valves required in the staged combustion cycles are summarized in Table 9 according to the three thrust chamber cooling concept groups. The least number of valves (10) is required by the oxygen cooled concepts 10 and 11 and the largest number (15) is required by concepts 12 and 13 which use both fuels sequentially for cooling the thrust chamber. This sequential use of fluids requires an increased number (3) of isolation valves over the all-hydrogen-cooled concepts (1-6). Isolation valves are used whenever a component such as the preburner, main thrust chamber, coolant jacket, or turbine is required to operate sequentially with two fluids. The respective isolation valve prevents the fluid in use from entering and contaminating the inactive circuit of the fluid not in use. In most cases, these isolation valves are simple one-way on-off valves while, in the case of isolation valves that handle hot gas, they can become large in size and intricate in design if nearly zero leakage is a requirement.

The principal system valves are used for coarse or fine control of fluid flow and are of design similar to the SSME valves. These are the main fuel and oxidizer valves, the preburners oxygen valves, and the coolant control valves. Though the SSME-type valve designs can be adopted in all cases for mode 1 tripropellant engine valves, the specific SSME hardware cannot be utilized in some cases because of differences in flowrate and pressure requirements between the SSME and the tripropellant engine (Table 10). The applicability of SSME control valves to the staged combustion tripropellant engine is indicated in Table 11. Because of flowrate restrictions (Table 10) the SSME OPOV cannot be used for the tripropellant engine OPOV (Table 11). Flowrate restrictions again preclude use of some of the SSME oxidizer valve candidates to the tripropellant engine HCPOV and HPOV (Table 11). There are no propellant isolation valves used in the SSME and, therefore, no candidates for the tripropellant engine isolation valves.

TABLE 9. CONTROL VALVE REQUIREMENTS, STAGED COMBUSTION CYCLES

VALVE NAME	SYMBOL	CONCEPTS 1-6 (H_2 Cooled)	CONCEPTS 12 & 13 (HC/H_2 Cooled)	CONCEPTS 10 & 11 (O_2 Cooled)
MAIN OXIDIZER VALVE	MOV	1	1	1
OXIDIZER PREBURNER OXIDIZER VALVE	OPOV	2	2	2
H.C. PREBURNER OXIDIZER VALVE	HCPOV	3	3	3
H_2 PREBURNER OXIDIZER VALVE	HPOV	4	4	4
MAIN H.C. COOLANT ISOLATION VALVE	MHC	5	5	5
OXIDIZER PREBURNER H.C. ISOLATION VALVE	OPHCIV	6	6	6
H_2-PREBURNER H.C. ISOLATION VALVE	HPHCIV	7	-	-
MAIN H_2 & COOLANT ISOLATION VALVE	MHV	8	8	8
COOLANT CONTROL VALVE	CCV	9	9	-
OXIDIZER PREBURNER H_2-ISOLATION VALVE	OPHIV	10	10	10
H_2-PREBURNER H_2 ISOLATION VALVE	HPHIV	11	-	-
T/C H.G. H.C. ISOLATION VALVE	HGHCIV	12	12	12
T/C H.G. H_2 ISOLATION VALVE	HGIV	-	13	13
HEATED H_2 ISOLATION VALVES	HH2IV	-	14,15	-
HEATED H.C. ISOLATION VALVES	HHCIV	-	16,17	

TABLE 10. FLOW AND PRESSURE REQUIREMENTS, STAGED COMBUSTION CYCLE SYSTEM CONTROL VALVES

VALVE FUNCTION	SSME	CONCEPT NUMBER									
		1	2	3	4	5	6	10	11	12	13
MOV, LB/SEC	965	1045	1045	1085	1085	1064	1064	1090	1090	1128	1101
PSI	4788	4123	4123	4123	4123	4123	4123	4123	4123	4123	4123
OPOV, LB/SEC	32.4	285.1	385.1	448	448	446	446	604	604	369	328
PSI	8038	7331	7331	7331	7331	7331	7331	7331	7331	7331	7331
HCPOV, LB/SEC	-	50.8	182	56.3	316	63.2	260	51.7	161	291	448
PSI	-	7331	7331	7331	7331	7331	7331	7331	7331	7331	7331
HPOV, LB/SEC	85.8	42.9	166	29.5	182	40	182	85.8	85.8	85.8	85.8
PSI	8038	7331	7331	7331	7331	7331	7331	8038	8038	8038	8038
MHV, LB/SEC	161	34	34	34	34	34	34	148	148	148	148
PSI	6831	4000	4000	4000	4000	4000	4000	6206	6206	6206	6206
CCV, LB/SEC	66.5	62	62	62	62	62	62	62	62	62	62
PSI	6534	5700	5700	5700	5700	5700	5700	5700	5700	5700	5700
MHCV, LB/SEC	-	300	300	252	252	287	287	389	389	322	367
PSI	-	4123	4123	4123	4123	4123	4123	4123	4123	6064	6064

TABLE 11. CONTROL VALVE AVAILABILITY, STAGED COMBUSTION CYCLES

VALVE NAME	SYMBOL	STAGED COMBUSTION CONCEPTS									
		1	2	3	4	5	6	10	11	12	13
Main Oxidizer Valve	MOV	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME
Oxidizer Preburner Oxidizer Valve	OPOV										
H. C. Preburner Oxidizer Valve	HCPOV	SSME HPOV		SSME HPOV		SSME HPOV		SSME HPOV			
H_2 Preburner Oxidizer Valve	HPOV	SSME		SSME		SSME		SSME	SSME	SSME	SSME
Main H.C. Coolant Isolation Valve	MHCV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV
Oxidizer Preburner H.C. Isolation Valve	OPHCIV										
H_2-Preburner H.C. Isolation Valve	HPHCIV										
Main H_2 and Coolant Isolation Valve	MHV	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME	SSME
Coolant Control Valve	CCV	SSME	SSME	SSME	SSME	SSME	SSME	-	-	SSME	SSME
Oxidizer Preburner H_2-Isolation Valve	OPHIV										
H_2-Preburner H_2 Isolation Valve	HPHIV										
T/C H.G. H.C. Isolation Valve	HGHCIV										
T/C H.G. H_2 Isolation Valve	HGIV										
Heated H_2 Isolation Valves	HH2IV										
Heated H.C. Isolation Valves	HHCIV										

Other Valve Requirements. In the case where components are operated with two propellants sequentially, purging of the component is required after use with the first propellant before use with the second can proceed. To minimize trajectory-performance losses, purging must occur in the shortest possible time interval. The propellants in question are methane, propane, and RP-1 used as coolant in the thrust chamber jacket, and as propellant in the main injector during mode 1 followed by hydrogen coolant during mode 2. Because the hydrogen enters the system at its normal boiling point of 37 R, the possibility exists that any of the hydrocarbon residuals may freeze. The lowest melting point is that of methane (154 R), the highest is that of RP-1 (405 R). Gaseous purging is required to reduce the concentration of these propellants and will be especially effective in the case of methane and propane. In the case of RP-1 (a liquid), purging effectiveness will depend on orientation of engine, location of vents, and geometry of the coolant passages. Experimental evaluation is required in this area. Purge valves and fluids are required, therefore, at the coolant jackets and at the injector manifolds for concepts 12 and 13. Concepts 1-6 require fuel system purge valves at the oxidizer preburner and at the hydrogen preburner. Concepts 10 and 11 require purge valves at the oxidizer preburner for the same reasons as stated above.

Gas Generator Cycle Concepts

The gas generator cycle cooling options are: all-hydrogen cooled, and hydrocarbon cooled in mode 1 with hydrogen cooling during mode 2. Only one gas generator is used in both concepts, thus necessitating an injector capable of burning LOX/hydrocarbon and LOX/hydrogen sequentially. Start and shutdown procedures are described below.

Mode 1 Operation. Criteria for start and shutdown are the same as outlined in Table 8. Schematics of the four engine concepts are categorized according to the two cooling options and are depicted in schematic form in Fig. 39 and 40.

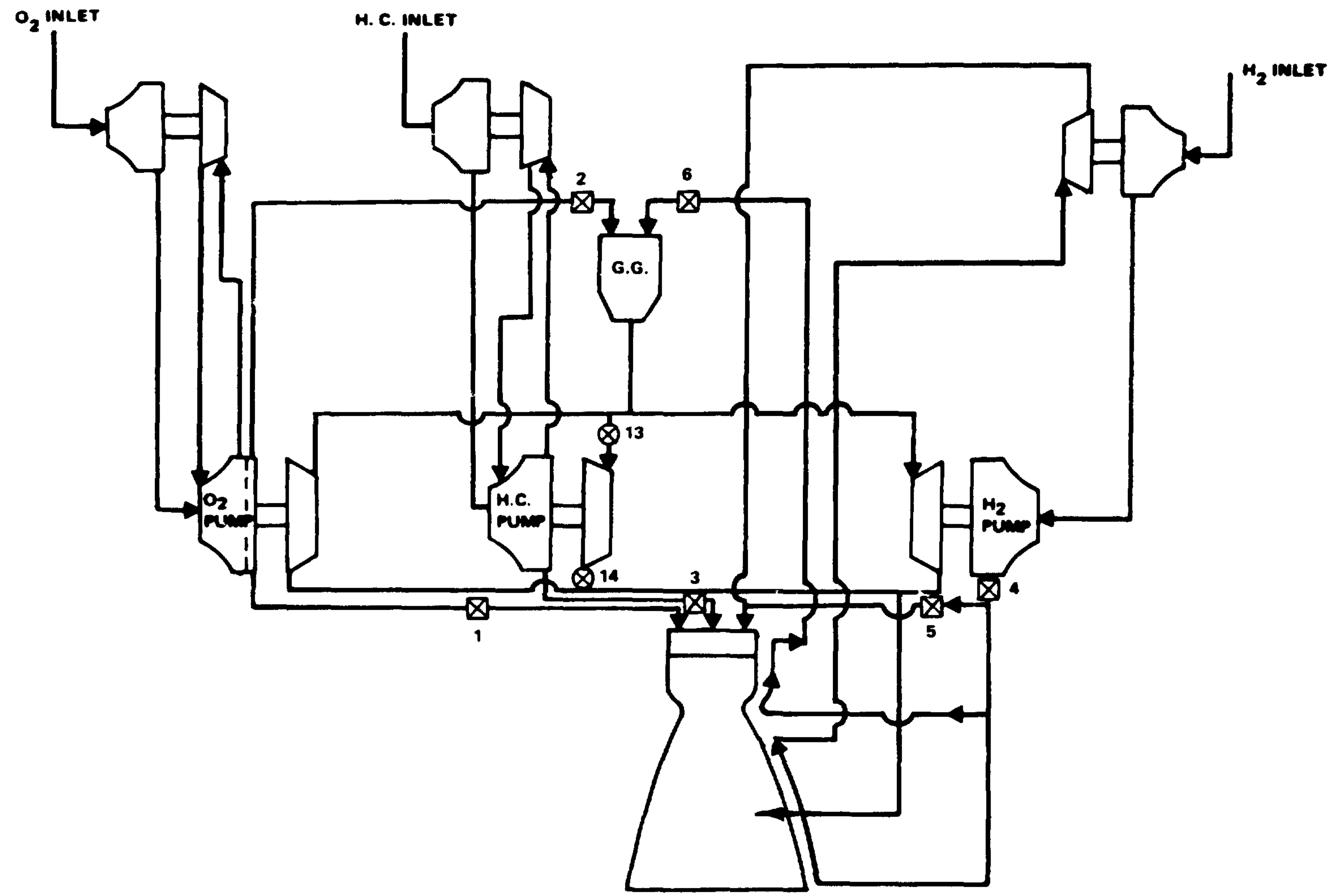


Figure 39. Dual-Mode Hydrogen-Cooled, Gas Generator Flow Schematic for Concepts No. 7, 8, and 9

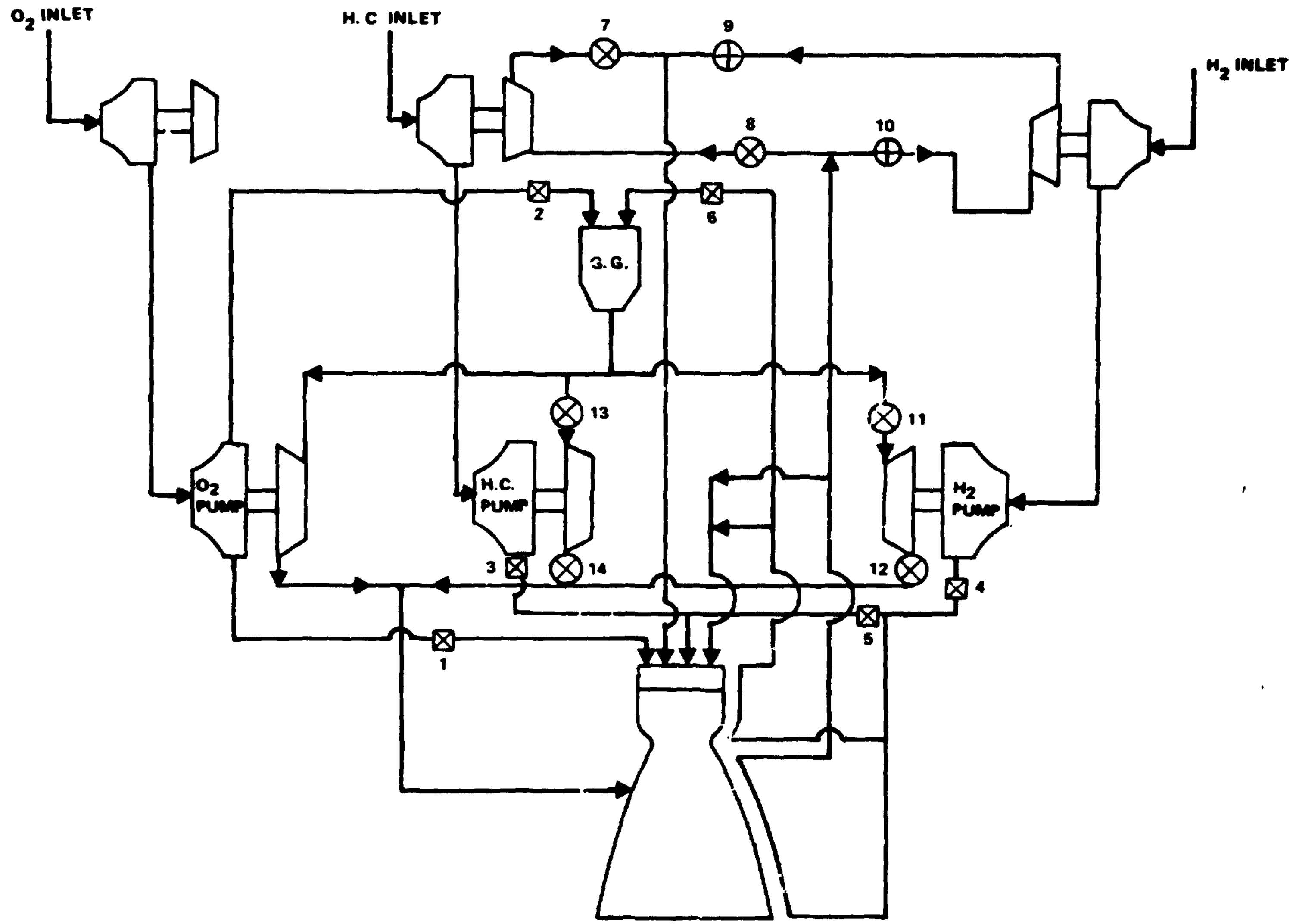


Figure 40. Dual-Mode Hydrocarbon H_2-Cooled, Gas Generator Schematic for Concepts No. 14 and 15

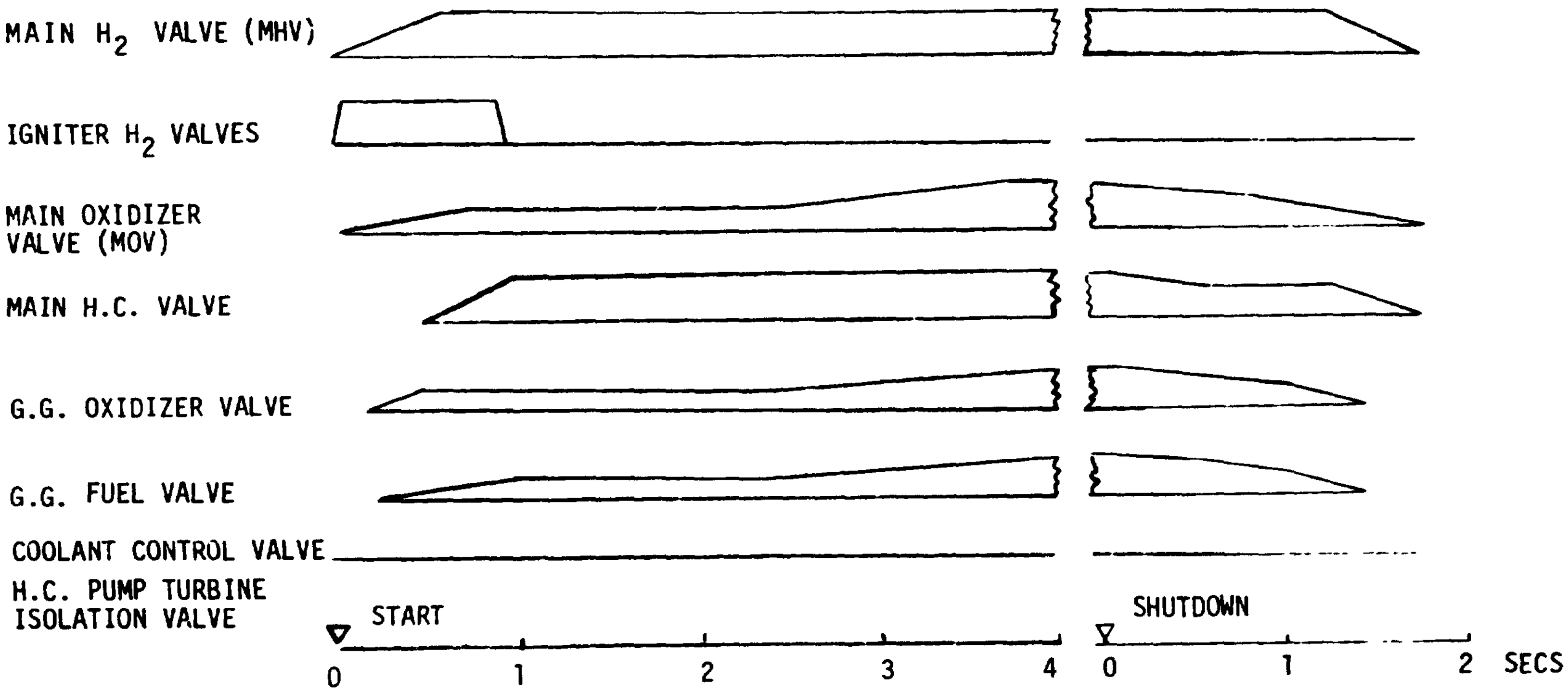
Start and shutdown procedures for both engine cooling categories are similar and will be discussed jointly.

Start valve sequencing is shown in Fig. 41. The start signal causes the main hydrogen valve (4) and the igniter hydrogen valves in the gas generator and thrust chamber to open, allowing priming of coolant jackets and hydrogen lines and initial igniter units hydrogen flow to start in the case of concepts 7, 8, and 9. Shortly after, the main oxidizer valve (1) and the gas generator oxidizer valve (2) are actuated allowing initial unseating of ball valve seals and allowing oxidizer flow to the igniter units. Ignition of LOX/H_2 propellants is then initiated in the gas generator and thrust chamber augmented spark igniters. In the case of concepts 14 and 15 (schematic in Fig. 40), the igniter hydrogen flow is obtained from upstream of the fuel valve which remains closed during mode 1 operation. The gas generator fuel valve (6) is then sequenced open (hydrogen in the case of concepts 7, 8, and 9). Main propellant ignition occurs then in the gas generator. Ignition is caused by the hot stream of combusting LOX/H_2 in the gas generator igniter. The main hydrocarbon valve (3) is then actuated which causes main propellant ignition to occur in the thrust chamber upon contact with the main chamber igniter LOX/H_2 combustion products.

The engine then enters a closed-loop control phase wherein the thrust is first increased to a 25% plateau with mixture ratio control and, after approximately 1/2 second, ramped to 100% rated thrust at prescribed ramp rates. This action produces start transients similar to those of the SSME (Fig. 33).

As in the staged combustion cycle concepts, closed-loop control prevents start transient overshoots or undershoots of any of the parameters that may affect engine life. It also provides for the uniformity of start transient behavior between engines.

Shutdown is effected with the same components and in a closed-loop control mode to minimize detrimental transients in turbine temperatures and pump speeds.



Concepts 14&15: HHCIV, HCPTIV, Valves Open; HH2IV, HPTIV Closed, MHV Closed

Figure 41. Start and Shutdown Sequence, Concepts 7, 8, 9, 14, 15, Mode 1

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Control Valve Requirements. The valve requirements for the gas generator cycles are shown in Table 12. Concepts 14 and 15 require 6 isolation valves more than concepts 7, 8, and 9. This is caused by the dual nature of the coolant fluid, i.e., hydrocarbon in mode 1 and hydrogen in mode 2. The hydrogen circuits need isolation during the hydrocarbon phase (valves 9 and 10) and vice-versa during mode 2 (valves 7 and 8). The hydrogen pump turbine requires isolation during mode 1 with valves 11 and 12. During mode 2, valves 13 and 14 isolate the hydrocarbon flow system from hot gases entering through the hydrocarbon pump turbine.

In addition to the six main control valves (1-6) concepts 7, 8, and 9 require hydrocarbon pump turbine isolation valves (13 and 14) during mode 2 operation.

Other Valve Requirements. For dual-fuel operation purging of the thrust chamber coolant jacket, injector manifolds and feed lines are required immediately after the hydrocarbon phase and before the hydrogen can be introduced in the circuits. Purging has to be performed to a degree so that no hydrocarbon residuals capable of freezing and obstructing flow passages or forming explosive mixtures remain. Other purge and inerting operations are as required by standard pre-launch or preactivation procedures.

Control Valve Availability. Flowrate and operating pressure requirements for the gas generator cycle main control valves are indicated in Table 13. Also shown are flows and pressures for applicable SSME main control valves. In Table 14, the applicability of SSME valve functions is indicated.

TABLE 12. CONTROL VALVE REQUIREMENTS, GAS GENERATOR CYCLES

VALVE NAME	SYMBOL	CONCEPTS 7, 8, 9	CONCEPTS 14 & 15
MAIN OXIDIZER VALVE	MOV	1	1
GAS GENERATOR OXIDIZER VALVE	GGOV	2	2
MAIN H.C. & COOLANT ISOLATION VALVE	MHCV	3	3
MAIN H_2 AND COOLANT ISOLATION VALVE	MHV	4	4
COOLANT CONTROL VALVE	CCV	5	5
GAS GENERATOR FUEL VALVE	GGFV	6	6
HEATED H.C. ISOLATION VALVES	HHCIV		7, 8
HEATED H_2 ISOLATION VALVES	HH2IV		9, 10
H_2 PUMP TURBINE ISOLATION VALVES	HPTIV		11, 12
H.C. PUMP TURBINE ISOLATION VALVES	HPTIV	13, 14	13, 14

TABLE 13. FLOW AND PRESSURE REQUIREMENTS, GAS GENERATOR CYCLE SYSTEM CONTROL VALVES

VALVE FUNCTION	SSME	CONCEPTS				
		7	8	9	14	15
MOV, LB/SEC	965	1056	1084	1058	1102	1074.5
PSI	4788	5106	5106	5106	5106	5106
GGOV, LB/SEC	(OPOV) 32.4	13.9	15.8	14.7	28.6	33.6
PSI	8038	5106	5106	5106	5106	5106
MHV, LB/SEC	161	34	34	34	-	-
PSI	6831	6084	6084	6084	-	-
GGFV, LB/SEC	(HPOV) 85.8	-	-	-	66.5	76.4
PSI	8038	-	-	-	5466	5107
MHCV, LB/SEC	-	335.7	277	314	373	423.3
PSI		5106	5106	5106	5106	5106

TABLE 14. CONTROL VALVE AVAILABILITY, GAS GENERATOR CYCLES

VALVE NAME	SYMBOL	CONCEPTS 7	8	9	14	15
Main Oxidizer Valve	MOV	SSME	SSME	SSME	SSME	SSME
Gas Generator Oxidizer Valve	GGOV	SSME OPOV	SSME OPOV	SSME OPOV	SSME OPOV	SSME OPOV
Main H.C. and Coolant Isolation Valve	MHCV	SSME MOV	SSME MOV	SSME MOV	SSME MOV	SSME MOV
Main H_2 and Coolant Isolation Valve	MHV					
Coolant Control Valve	CCV	SSME	SSME	SSME	SSME	SSME
Gas Generator Fuel Valve	GGFV				SSME HPOV	SSME HPOV
Heated H.C. Isolation Valves	HHCIV					
Heated H_2 Isolation Valves	HH2IV					
H_2 Pump Turbine Isolation Valves	HPTIV					
H.C. Pump Turbine Isolation Valves	HCPTIV					

TASK V - SSME COMPONENT ADAPTABILITY

This task is organized to interact with the efforts of Tasks I through IV to evaluate the possibility of adapting the already designed and proved SSME components to the candidate systems. A study has been conducted to evaluate the adaptability of existing oxidizer and fuel low-pressure and high-pressure pumps. A separate study was conducted to determine if the existing SSME turbines can satisfy the horsepower and hot-gas flow requirements established in the engine balance analysis. A third study was conducted to investigate the possibility of using the SSME preburners and main injectors in the tripropellant systems being studied.

SSME PUMP APPLICABILITY

The flow and head requirements for each of the candidate systems are presented in Table 15. The full-flow pumps supply the head rise for the total flow to the chamber; a part-flow pump then provides the additional head rise to that portion of the flow that goes to the preburner. Pump applicability was determined by plotting the design point head and flow requirements on the SSME pump performance maps (Fig. 42 through 46) that are based on a combination of analytical predictions and experimental test data. Both full-flow and part-flow pumping requirements are plotted. The data points that fell within the pump operating limits (defined in Table 16) were then denoted as being acceptable for that application. Due to the very low density of liquid hydrogen relative to the other propellants in Table 15, hydrogen pumps were not applicable to the other propellants. Therefore, hydrogen pumps were considered only for hydrogen and liquid oxygen pumps were used for all other propellants (LOX, RP-1, CH_4, and C_3H_8). For certain very low fuel flowrate cases (2, 6, 11, 12, and 13), the oxidizer pump for the ASE engine (the Mark 38 oxidizer pump) was found to be very applicable, as shown in Fig. 46. Therefore, that pump was included in the study.

The applicability of the SSME turbines was evaluated in a separate study; only the pumps are being considered in this discussion. However, the established pump horsepower requirements were used in the evaluation of the turbines. In

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TABLE 15. ENGINE REQUIREMENTS

Case	Propellant	Pump	Weight Flowrate, $\dot{w}$, lb/sec	Pressure Rise, ΔP, psi	Propellant Density, ρ, lb/ft^3	Volume Flowrates, Q, gpm	Head Rise, ΔH, ft
1	LOX	Full Flow	1045	4108	71.1	6600	8,320
	LOX	Part Flow	478.8	3208	71.1	3020	6,500
	RP-1	Full Flow	300.3	4108	50.5	2670	11,710
	RP-1	Part Flow	240.3	3208	50.5	2520	9,150
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
2	LOX	Full Flow	1045	4108	71.1	6600	8,320
	LOX	Part Flow	733.2	3208	71.1	4630	6,500
	RP-1	Full Flow	300.3	4108	50.5	2670	11,710
	RP-1	Part Flow	22.2	3208	50.5	197	9,150
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
3	LOX	Full Flow	1085	4108	71.1	6850	8,320
	LOX	Part Flow	533.8	3208	71.1	3370	6,500
	CH_4	Full Flow	252	4108	27.5	4110	21,500
	CH_4	Part Flow	207.1	3208	27.5	[illegible]	16,800
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
4	LOX	Full Flow	1085	4108	71.1	6850	8,320
	LOX	Part Flow	945.9	3208	71.1	5970	6,500
	CH_4	Full Flow	252	4108	27.5	4110	21,500
	CH_4	Part Flow	25.2	3208	27.5	411	16,800
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
5	LOX	Full Flow	1063.5	4108	71.1	[illegible]	8,320
	LOX	Part Flow	549.5	3208	71.1	3470	6,500
	C_3H_8	Full Flow	286.5	4108	36.4	3530	16,250
	C_3H_8	Part Flow	247.6	3208	36.4	3050	12,690
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
6	LOX	Full Flow	1063.5	4108	71.1	6720	8,320
	LOX	Part Flow	887.8	3208	71.1	5610	6,500
	C_3H_8	Full Flow	286.5	4108	36.4	3530	16,250
	C_3H_8	Part Flow	26.1	3208	36.4	322	12,690
	LH_2	TC Coolant	34	3985	4.42	3450	117,500
7	LOX	Full Flow	1056	5091	71.1	6670	10,310
	RP-1	Full Flow	335.7	5091	50.5	798.	14,320
	LH_2	TC Coolant + Turb.	34	6069	4.42	3450	172,700
8	LOX	Full Flow	1084	5091	71.1	6850	10,310
	CH_4	Full Flow	277	5091	27.5	4520	26,700
	LH_2	TC Coolant + Turb.	34	6069	4.42	3450	172,700

C-2

TABLE 15. (Concluded)

Case	Propellant	Pump	Flowrate, $\dot{w}$, lb/sec	Pressure Rise, ΔP, psi	Propellant Density, ρ, lb/ft^3	Volume Flowrates, Q, gpm	Head Rise, ΔH, ft
9	LOX	Full Flow	1058	5091	71.1	6680	10,310
	C_3H_8	Full Flow	314	5091	36.4	3870	20,100
	LH_2	TC Coolant + Turb.	34	6069	4.42	3450	172,700
10	LOX	Full Flow	1090.4	7316	71.1	6890	14,820
	LOX	Part Flow	511.9	1269	71.1	3230	2,570
	RP-1	Full Flow	389.4	4108	50.5	3460	11,710
	RP-1	Part Flow	140.1	3208	50.5	1246	9,150
11	LOX	Full Flow	1090.4	7316	71.1	6890	14,820
	LOX	Part Flow	621.5	1269	71.1	3920	2,570
	RP-1	Full Flow	389.4	4108	50.5	3460	11,710
	RP-1	Part Flow	18.8	3208	50.5	167.2	9,150
12	LOX	Full Flow	1128	4108	71.1	7120	8,320
	LOX	Part Flow	659.2	3208	71.1	4160	6,500
	CH_4	Full Flow	322	6049	27.5	5260	31,700
	CH_4	Part Flow	17.6	1267	27.5	287	6,630
13	LOX	Full Flow	1101	4108	71.1	6950	8,320
	LOX	Part Flow	776	3208	71.1	4900	6,500
	C_3H_8	Full Flow	367	6049	36.4	4530	23,900
	C_3H_8	Part Flow	22.8	1267	36.4	281	5,010
14	LOX	Full Flow	1102	5091	71.1	6960	10,310
	CH_4	Full Flow	373	5451	27.5	6090	28,500
	CH_4	Part Flow	66.5	3140	27.5	1086	16,440
15	LOX	Full Flow	1074.5	5091	71.1	6790	10,310
	C_3H_8	Full Flow	423.3	5091	36.4	5220	20,100
	C_3H_8	Part Flow	76.4	2605	36.4	942	10,310

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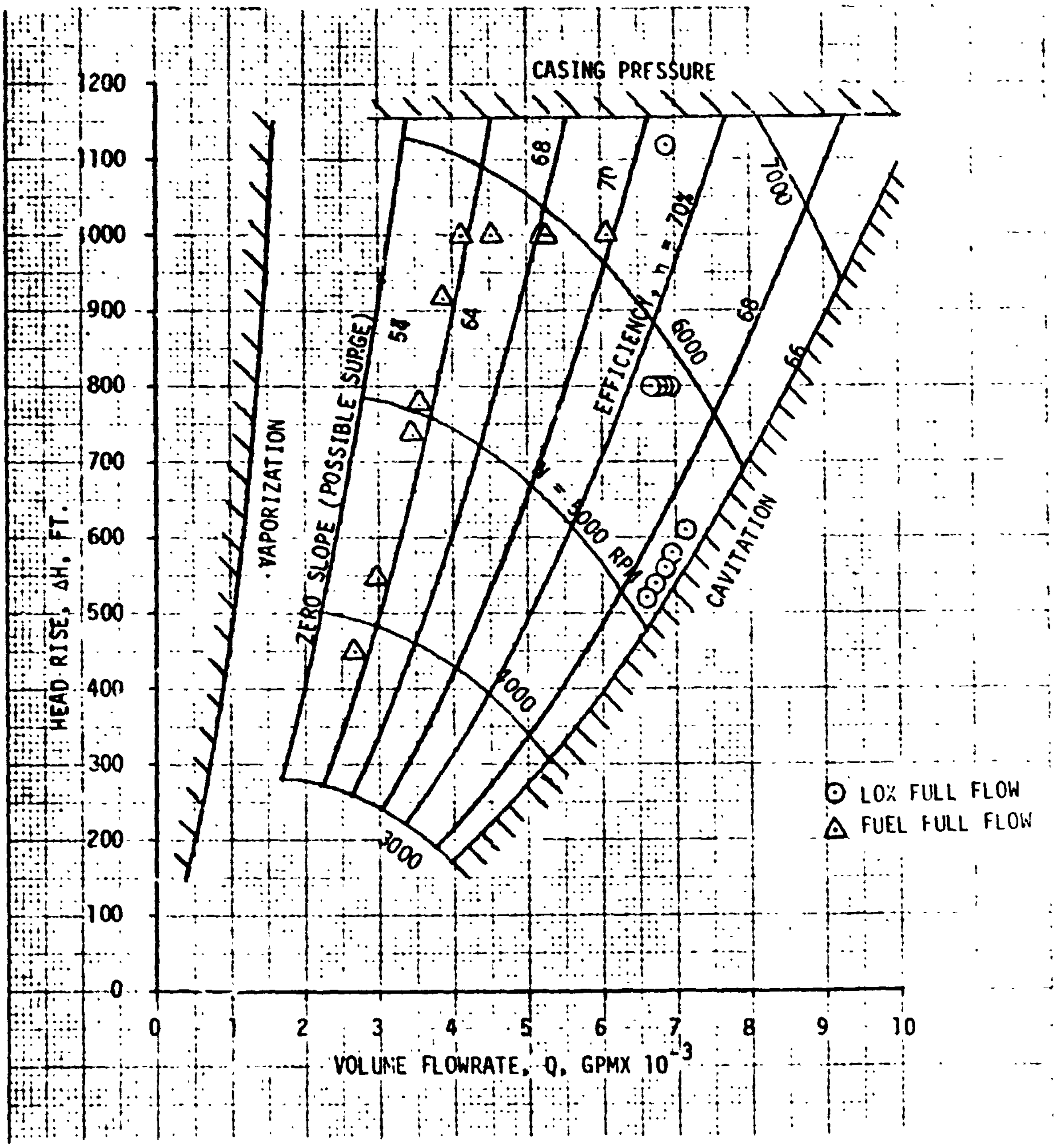

Figure 42. SSME Low-Pressure Oxidizer Pump Performance Map and Limits

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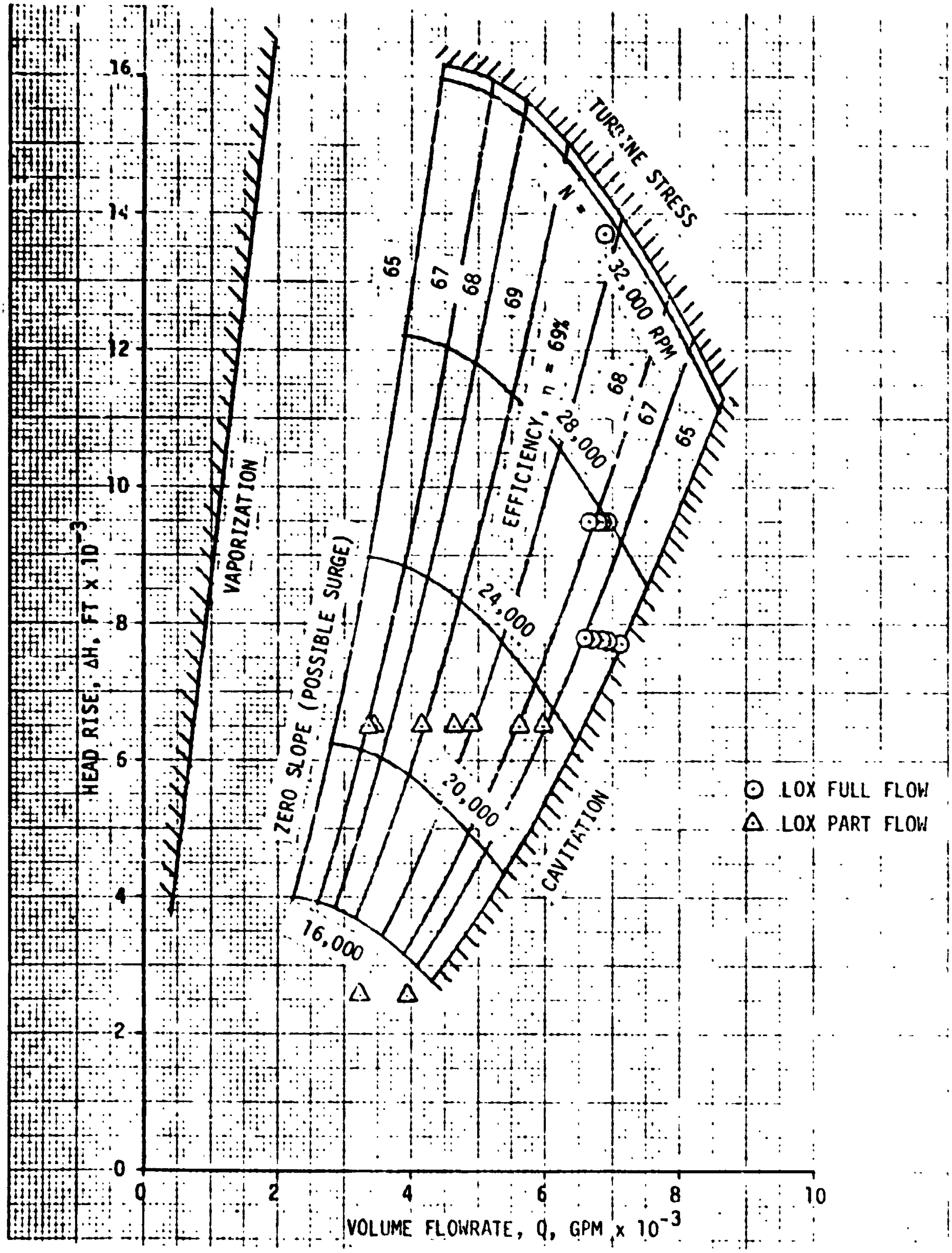

Figure 43. SSME High-Pressure Oxidizer Pump
Performance Map and Limits

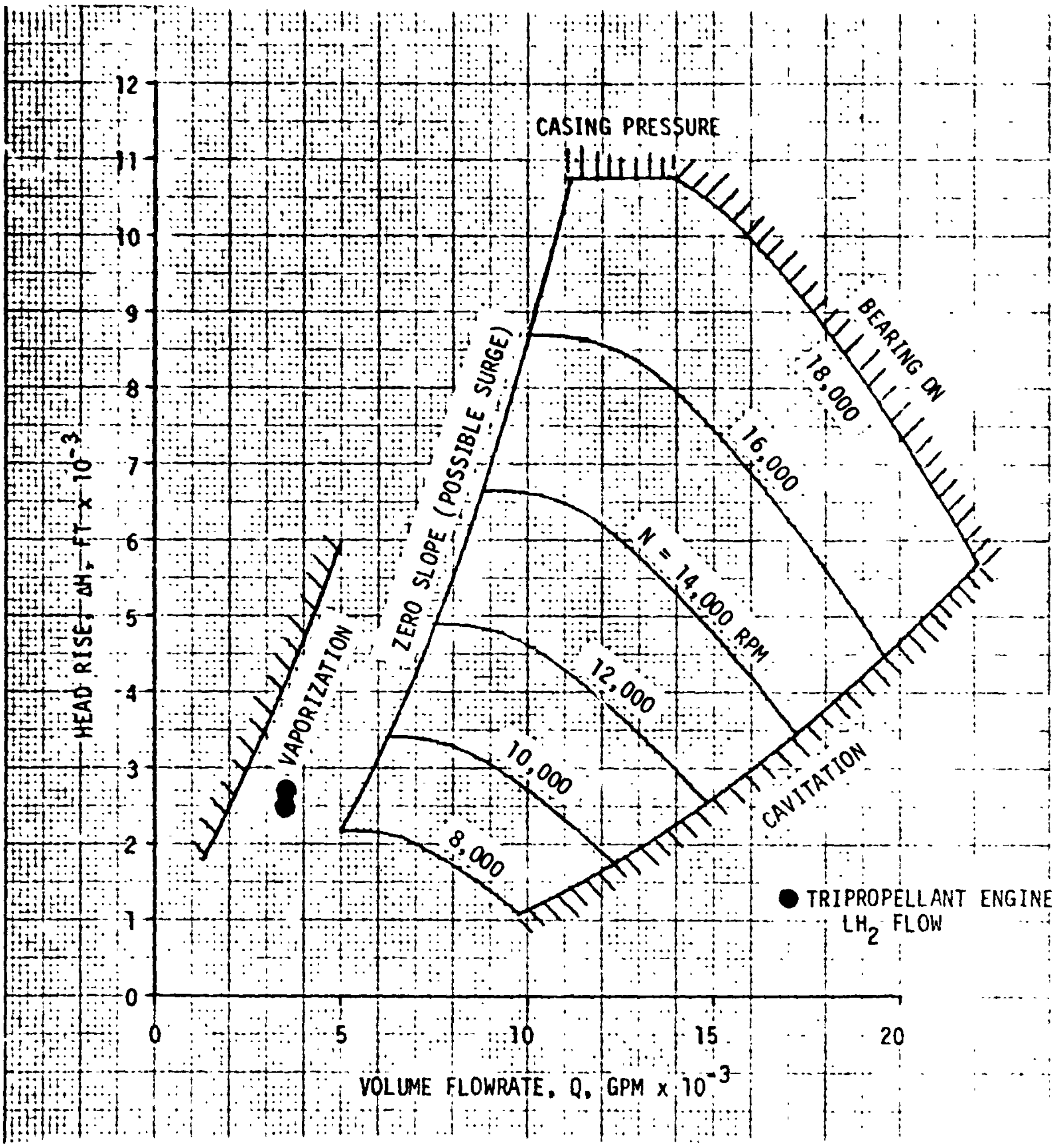

Figure 44. SSME Low-Pressure Fuel Pump Performance Map and Limits

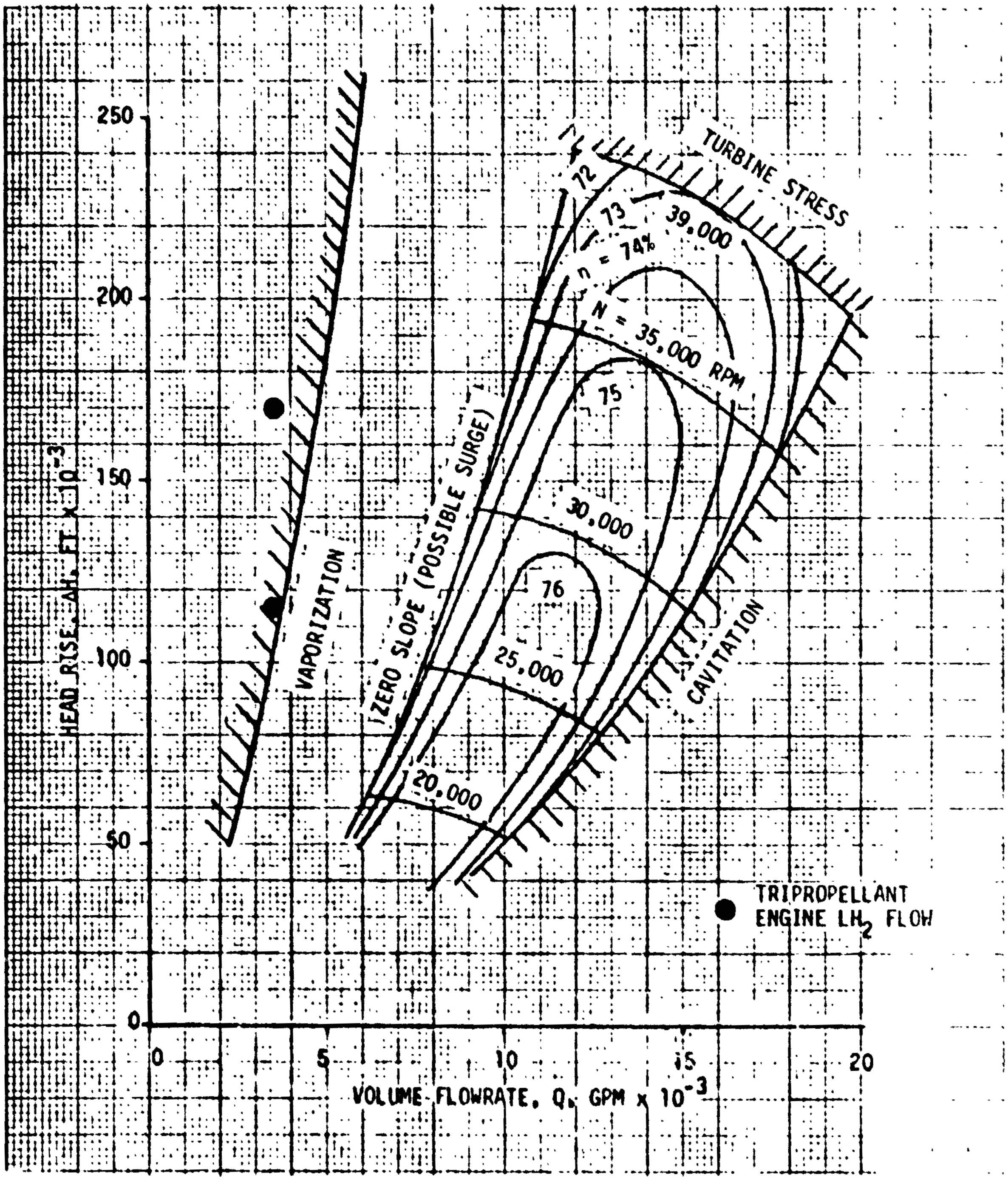

Figure 45. SSME High-Pressure Fuel Pump Performance Map and Limits

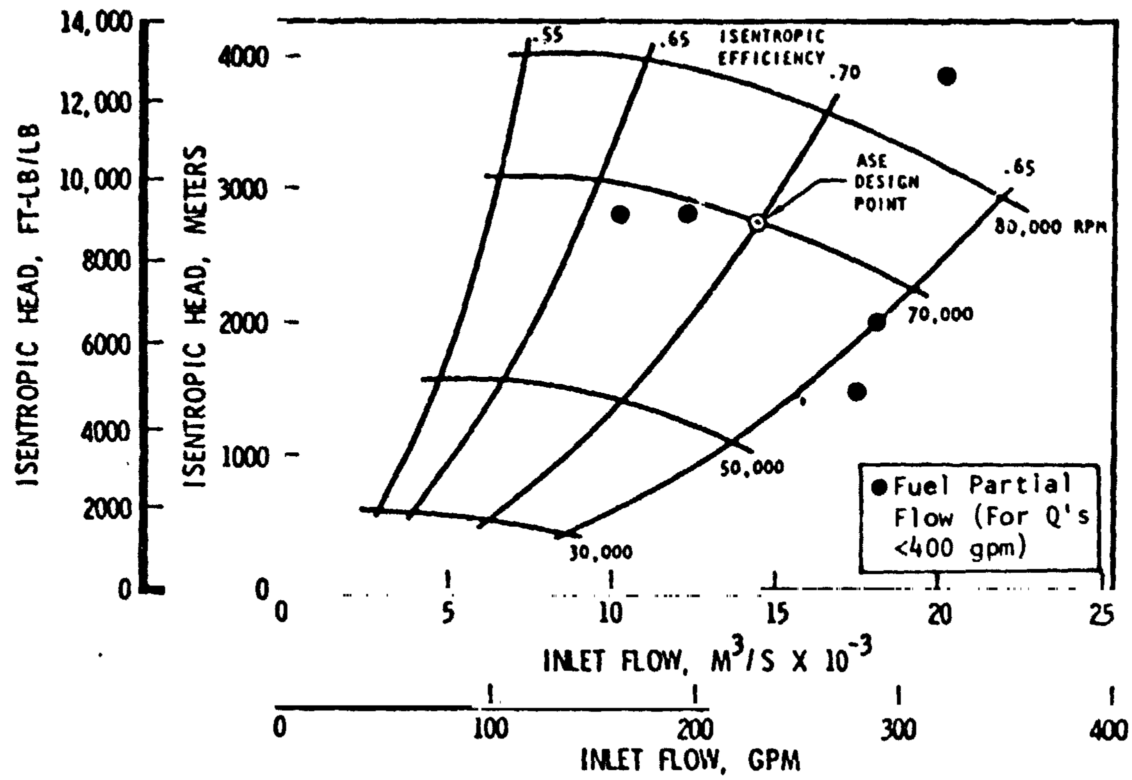

Figure 46. Mark 48 Oxidizer Pump Performance

TABLE 16. SSME TURBOPUMP LIMITS

Limit	Description
TURBINE STRESS	TURBINE WHEEL FACTOR OF SAFETY ON ULTIMATE = 1.2 ON ROTATIONAL SPEED (WHEEL BURSTS @ N/N_{DES} = 1.2).
CASING PRESSURE	CASING FACTOR OF SAFETY ON ULTIMATE = 1.5 (CASING BURSTS @ $\Delta P/\Delta P_{DES}$ = 1.5).
VAPORIZATION	HIGH TEMPERATURE RISE AT LOW FLOW CAUSES VAPORIZATION AND CONSEQUENT PRESSURE DROP IN PUMP.
CAVITATION	HIGH FLOW COEFFICIENT (Q/N) OPERATION CAUSES DROP IN SUCTION PERFORMANCE CAPABILITY.
BEARING DN	AXIAL THRUST LOADS ARE TOO HIGH FOR THE DN AT WHICH THE BEARING IS OPERATING.
ZERO SLOPE	OPERATION TO THE LEFT OF ZERO SLOPE CAN CAUSE SURGING IN THE PUMP.

some cases, if it is found that the turbine is unsatisfactory for a particular application, this will also disqualify the pump for a direct substitution as the SSME turbopumps are an integral unit and it is considered a major modification to separate the two and mate them to a new pump or turbine. The results are summarized in Table 17. As shown, a new pump was indicated if no applicable unit could be found.

As shown in Table 17, the SSME low- and high-pressure oxidizer turbopumps (LPOTP and HPOTP) were found to be satisfactory for all full-flow LOX pumping applications and the HPOTP was found to be satisfactory for all part-flow LOX pumping applications. All of the oxidizer part flows (preburner flow) are higher flowrates than the SSME preburner oxidizer turbopump (PBOTP) can handle; therefore, a second HPOTP must be used in series to provide the oxidizer preburner flow. The PBOTP will still be required on the full-flow pump for mode 2 operation but the PBOTP could be removed from the second pump in the series arrangement.

As far as the full-flow fuel applications are concerned, the LPOTP is satisfactory as a boost pump and a new pump is required for all main pumps. If the SSME HPOTP is capable of being throttled to the left of the zero slope line on the performance map, the HPOTP may be applicable to the lower pressure, full-flow fuel cases (1, 2, 5, 6, 7, 10 and 11). However, this would be at the expense of pump efficiency. It is also of interest to note that two HPOTP stages (in series) would do the job for all full-flow fuel pumps and most the the part-flow fuel pumps. However, such a design would require a new housing, a new bearing and seal arrangement, a new shaft, and the addition of interstage ducting. It would be considered a new design and, therefore, such a candidate was not considered practical.

The ASE oxidizer pump (ASEOTP) was found to be applicable to the part-flow fuel pumping applications for cases 1, 6, 11, 12, and 13 (again, the turbine was not considered). For all other part-flow fuel cases, a new pump was found to be necessary.

TABLE 17. PUMP APPLICABILITY

Case	Propellant	Application	Pump Candidates	
			Boost	Main
1	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	RP-1	Full Flow	LPOTP	New Pump
	RP-1	Part Flow	None	New Pump
	LH_2	TC Coolant	LPFTP(Marginal)	New Pump
2	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	RP-1	Full Flow	LPOTP	New Pump
	RP-1	Part Flow	None	ASEOTP
	LH_2	TC Coolant	LPFTP (Marginal)	New Pump
3	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	CH_4	Full Flow	LPOTP	New Pump
	CH_4	Part Flow	None	New Pump
	LH_2	TC Coolant	LPFTP (Marginal)	New Pump
4	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	CH_4	Full Flow	LPOTP	New Pump
	CH_4	Part Flow	None	New Pump
	LH_2	TC Coolant	LPFTP (Marginal)	New Pump
5	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	C_3H_8	Full Flow	LPOTP	New Pump
	C_3H_8	Part Flow	None	New Pump
	LH_2	TC Coolant	LPFTP (Marginal)	New Pump
6	LOX	Full Flow	LPOTP	HPTOP
	LOX	Part Flow	None	HPOTP
	C_3H_8	Full Flow	LPOTP	New Pump
	C_3H_8	Part Flow	None	ASEOTP
	LH_2	TC Coolant	LPFTP (Marginal)	New Pump
7	LOX	Full Flow	LPOTP	HPOTP
	RP-1	Full Flow	LPOTP	New Pump
	LH_2	TC Coolant + Turbine	LPFTP (Marginal)	New Pump

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TABLE 17. (Concluded)

Case	Propellant	Application	Pump Candidates	
			Boost	Main
8	LOX	Full Flow	LPOTP	HPOTP
	CH_4	Full Flow	LPOTP	New Pump
	LH_2	TC Coolant + Turbine	LPFTP (Marginal)	New Pump
9	LOX	Full Flow	LPOTP	HPOTP
	C_3H_8	Full Flow	LPOTP	New Pump
	LH_2	TC Coolant + Turbine	LPFTP (Marginal)	New Pump
10	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	RP-1	Full Flow	LPOTP	New Pump
	RP-1	Part Flow	None	New Pump
11	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	RP-1	Full Flow	LPOTP	New Pump
	RP-1	Part Flow	None	ASEOTP
12	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	CH_4	Full Flow	LPOTP	New Pump
	CH_4	Part Flow	None	ASEOTP
13	LOX	Full Flow	LPOTP	HPOTP
	LOX	Part Flow	None	HPOTP
	C_3H_8	Full Flow	LPOTP	New Pump
	C_3H_8	Part Flow	None	ASEOTP
14	LOX	Full Flow	LPOTP	HPOTP
	CH_4	Full Flow	LPOTP	New Pump
	CH_4	Part Flow	None	New Pump
15	LOX	Full Flow	LPOTP	HPOTP
	C_3H_8	Full Flow	LPOTP	New Pump
	C_3H_8	Part Flow	None	New Pump

As far as liquid hydrogen is concerned, the flows are low enough to possibly cause surging in the SSME LPFTP (Fig. 44) and both surging and vaporization in the SSME HPFTP (Fig. 45). As a result, the LPFTP was deemed marginal and the HPFTP was deemed unacceptable. If future SSME engine throttling studies and modifications are successful in throttling this pump down to these flows, the applicability should be reassessed due to the dual-mode engine simplification that could be obtained (for cases 1 through 9) if the mode 2 pumps could be used. Another possibility is the use of four ASE liquid hydrogen pumps (ASEFTP). This would probably be too complex. However, a redesign possibility would be an ASEFTP scaled up to twice size (twice the diameter and half the speed) so that it would match with the higher flow.

As far as new pump designs are concerned, only one new design is required for cases 11, 12, and 13. This is for the high-pressure fuel pumps that require more head than can be delivered by the HPOTP because the possible surge limit is exceeded in all three cases and the turbine stress limit is exceeded in cases 12 and 13. However, all three cases require the addition of four pumps to the SSME system to get the dual-mode capability. Cases 14 and 15 require the minimum number of additional pumps, which is three. However, two of them have to be new designs.

SSME TURBINE APPLICABILITY

The turbomachinery study phase of this task of the tripropellant engine investigation is concerned with the utilization of existing SSME and ASE turbomachinery in the propellant feed systems of the candidate engine concepts. The turbine analyses were conducted to establish a relationship between the required operating condition for the tripropellant feed systems being evaluated and the operational capability of the turbines.

Those designs that could be adaptable to this application would have to be used either as built or require redesign of the gas path elements only; this includes the nozzles and rotor blades only. Any additional modifications to the turbine assemblies are not practical because of the complexity of the turbomachinery. The development of new designs would be more cost-effective on the basis of development time, performance characteristics, and modification cost. The criteria used to evaluate the respective high-pressure fuel and oxidizer turbines to the 15 candidate concepts are associated with: (1) the engine cycle, (2) turbine working fluid properties and available energy, (3) operating conditions and required turbine horsepower, and (4) size of the existing gas paths to handle the required turbine flows.

The high-pressure SSME turbopumps are driven by two-stage, reaction turbine designs; the respective pitch diameters of the fuel and oxidizer turbines are 10.19 and 10.09 inches. The principal turbine operating parameters are as follows:

	Turbine	HPOTP	HPFT
1.	Working Fluid	LO_2/LH_2	LO_2/LH_2
2.	Speed, N, rpm	31,204	38,000
3.	Total Inlet Pressure, P_{t1}, psia	5,848	5,916
4.	Turbine Pressure Ratio, PR_t, T-T	1.57	1.58
5.	Mass Flowrate, W_t, lb/sec	64.24	162.7
6.	Horsepower, HP_t	28,658	76,698
7.	Total Inlet Temp, t_{t1}, R	1,567	1,928

A major consideration is the engine cycle in which these low-pressure ratio turbines, which were designed for the staged combustion SSME, shall be required to operate.

The gas turbine analyses utilized the working fluid available energy data and the operating parameters. Turbine velocity ratios (U/C_o) were established, and predictions of turbine performance were subsequently calculated. The required turbine mass flowrates, based on oxidizer and fuel propellant pump horsepower(s) and speed(s), were evolved. If the required turbine powers could be developed with the propellant feed system operating conditions, the required turbine gas path flow areas were calculated. This determined whether the existing turbine hardware could be used for the application or the limiting parameters could be pinpointed and gas path modifications could be considered. A summary of the study conclusions is presented in Table 18.

Candidate engine 1 and 2 utilize LOX/RP-1 turbine working fluid in a staged combustion cycle installation. The 28,660 design horsepower of the HPOT turbine is not exceeded by the required 22,100 horsepower of these candidate engines. The required 25,800 rpm turbine speed can be achieved. The analysis indicates the LO_2/RP-1 velocity ratio (U/C_o) is 0.624; this is in an unfavorable off-design operating region; the HPOT turbine design U/C_o is 0.296. The oxidizer turbine required turbine gas path area is larger than the physical areas existing in the turbine nozzles and blading. The area difference is too large, and modification of the existing gas path elements is not practical. The required flow area(s) is approximately three times larger than available in the existing turbine. Use of the HPOT turbine in those applications is not recommended.

Candidate engines 3 and 4 use LO_2/CH_4 turbine working fluid in a staged combustion cycle configuration. The 23,200 required turbine horsepower in these candidate engines does not exceed the HPOT turbine design power, and turbine speed can be achieved for these candidate designs. Turbine velocity ratio U/C_o is 0.684; this is in the off-design operating range of the turbine. In addition, a large difference exists between the gas path area(s) required for these candidate applications and the flow area(s) available in the HPOT turbine. This is exemplified by the 12.57 sq in. area required in the first-stage nozzle for the LO_2/CH_4 working fluid; the current design area for this gas path element is 2.94 sq in. The difference between these turbine gas path areas is too large and it is impractical to consider modifying the existing turbine design to accommodate operations for the LO_2/CH_4 staged combustion condidates.

The LO_2/C_3H_8 turbine performance and flow constraints for No. 5 and 6 candidate engines are approximately the same as found in the staged combustion candidates

TABLE 18. TURBINE EVALUATION

T/M COMPONENT			FIRST STAGE NOZZLE		COMMENTS
CANDIDATE	TURBINE	APPLICATION	FLUID SPECIFIC VOLUME ft^3/lb	FLOW AREA REQUIRED IN^2	LIMITING PARAMETER
1	HPØT	Full Flow Lox	0.198	9.17	Turbine Flow Area Not Adequate.
2	HPØT	Full Flow Lox	0.198	9.17	
3	HPØT		0.203	12.57	
4	HPØT		0.203	12.57	
5	HPØT		0.200	12.20	
6	HPØT		0.200	12.20	
7	HPØT		4.038	2.23	New Nozzles and Blading May Satisfy Requirements.
8	HPØT		4.038	2.33	
9	HPØT		4.038	2.28	
10	HPØT		-	-	Turbine Horsepower Requirements Exceeds Design.
11	HPØT		-	-	HP design = 28,658 HP required = 39,300
12	HPØT		0.203	12.57	Turbine Flow Area Not Adequate.
13	HPØT		0.201	13.26	" " " " "
14	HPØT		-	-	Requires at Least One Additional Turbine Stage
15	HPØT		-	-	" " " " "

Design Nozzle Flow Area = 2.94 Sq. In.
Design Nozzle (First Stage) Specific Volume = 1.066 ft^3/lb

1 through 4. The 0.683 off-design velocity ratio (U/C_o) in these candidate designs reduces the turbine efficiency to 55.7%. The 22,600 horsepower can be achieved at a speed of 26,100 rpm; this requires a turbine mass flowrate of 508 lb/sec.

Candidate engines 7, 8, and 9 utilize LO_2/H_2 turbine working fluid in a gas generator engine cycle. The study results indicate the required turbine powers and speeds can be achieved. To accomplish this, 29 lb/sec turbine mass flowrate is required with the designated turbine pressure ratio of 20:1. The gas path of the HPOT turbine was designed for a pressure ratio of 1.57 for a staged combustion engine cycle. Therefore, to satisfy the required power requirements, the analysis indicates the turbine should be modified with new nozzle(s) and rotor blading designs in these candidate engines. A typical redesigned turbine gas path will contain two stages, with a pressure ratio of 5 across the first stage. The resultant 55% stage efficiency is influenced principally by the low 0.155 velocity ratio (U/C_o) in which the turbine will operate. The turbine performance can be improved with adjustments in the design speed, pressure ratio, and turbine inlet temperature. The required first-stage nozzle areas for the existing reaction turbine design and for the redesigned gas generator cycle turbine nozzle are approximately equal.

The horsepower requirements in the staged combustion candidate engines No. 10 and No. 11 exceed the design power of the HPOT turbine and, therefore, eliminate its use in these applications. The required turbine power is 39,300 horsepower whereas the existing turbine was designed to develop a maximum of 28,658 horsepower. A redesign of the turbine to accommodate the increased power requirement is not practical. The complexity of modifying the existing configuration, coupled with the cost and time required to achieve this type of change, eliminates use of the HPOT turbine in these candidate engines.

LO_2/CH_4 and LO_2/C_3H_8 turbine working fluids are respectively used in the staged combustion candidate engines 12 and 13. The power and speed required in these applications are within the design limits of the HPOT turbine. The turbine velocity ratio(s) (U/C_o) are 0.69 at the 24,000 rpm speed range and 1.6 turbine pressure ratio. This places the turbine in an off-design operating range and, therefore, the performance is penalized; the predicted turbine efficiency is 55% in each of these candidate engine systems. The turbine required mass flow is 555 lb/sec and 553 lb/sec, respectively, in engines 12 and 13. The initial sizing of the gas path details indicates the existing turbine nozzle area is too small to accommodate flow for the new application. The 2.94 sq in. nozzle design area is approximately one-fourth the area required for the 555 lb/sec turbine mass flowrate in candidates 12 and 13. The use of the HPOT turbine is eliminated on the basis of low turbine performance and too large a mismatch in gas path area to effectively implement a gas path modification.

Candidate engines 14 and 15 require turbine designs, which respectively operate with O_2/CH_4 and O_2/C_3H_8 working fluids in gas generator installations. The HPOT turbine design speed and horsepower are within the design requirements for these candidate engines. Matching the gas path conditions, at the 20:1 turbine pressure ratio to the 1.57 HPOT design pressure ratio configuration, reduced the velocity ratio range in which the turbines operate. The respective single-stage velocity ratios (U/C_o) for these candidates are 0.188 and 0.133; these were calculated with a 5:1 pressure ratio in the first stage. The data indicated the use of a two-stage HPOT turbine was pressure-ratio limited and, therefore, new nozzles and blading were considered. The performance of a typical redesigned two-stage configuration is penalized because of the velocity ratio range in which it will operate. The proper design for these candidate engines would contain three turbine stages, or perhaps a three-row design could be developed to efficiently utilize the working fluid available energy. For these reasons, the use of the HPOT turbine was determined not suitable for these candidate applications. A redesign to a three-turbine rotor configuration for the HPOT turbopump is too complex and costly. A new turbopump design is recommended.

SSME COMBUSTION COMPONENTS ADAPTABILITY

The purpose of this task is to evaluate the SSME preburner and main combustion chamber injectors to determine if they could be used in any of the 15 candidate tripropellant engine systems. This means that these injectors must provide stable, high performance when operating in any of the 15 mode 1, hydrocarbon fuel configurations and then be able to switch to LOX/H_2 operation in mode 2.

The SSME turbopumps are powered by two preburners providing fuel-rich gases.

The two preburner flows expand through the turbines and are then combined and ducted to the main injector. Both the preburners and the main chambers employ coaxial-type injectors. The preburners have liquid oxygen injected through the center post and gaseous H_2 injected from the annulus. In the main injector, the fuel-rich turbine exhaust gases are injected through the annulus and liquid oxygen in the center post. SSME injector flow areas are presented in Table 19.

TABLE 19. SSME INJECTORS TOTAL FLOW AREA*

	Center, ft^2	Annulus, ft^2
Fuel Preburner	0.0114	0.025
Oxidizer Preburner	0.00388	0.01113
Main Chamber	0.1012	0.1979

*Excluding baffle elements

The coaxial injector relies on a large velocity ratio between the two streams to enhance the turbulent mixing. If one fluid is in liquid form, atomization can be achieved only by the shearing force between the two streams. Hence, a large velocity differential is promoted to ensure good atomization and subsequently good vaporization and high performance. To determine whether hydrocarbon fuels can be used in the SSME combustion devices, the injection velocities must be estimated for each case based on the fixed injector element flow areas. The calculated injection velocities for the preburners and main chamber are presented in Tables 20, 21, and 22 for the staged combustion cycle engine systems and Table 23 for the gas generator cycles. The SSME conditions also are shown for reference. A velocity ratio of 10 or higher is desirable.

In cases 1, 3, 5, and 10, the oxidizer preburner operates oxidizer rich and the fuel and H_2 preburners operate fuel rich. These two gas streams would either have to be mixed prior to injection into the main chamber or injected separately. The latter would require a completely redesigned injector since three streams must be accommodated. If the oxidizer- and fuel-rich turbine exhaust streams are mixed prior to injection, it would be extremely difficult to maintain the mixture nonreactive and avoid a detonation hazard. If they are allowed to react further in another chamber, the cooling would be a substantial engineering problem. The injection velocities shown for these cases in Table 20 were based on the assumption that the two streams are mixed prior to injection and somehow maintained nonreactive. Based on these factors, the main chamber injector cannot be used directly in cases 1, 3, 5, and 10. In the case of the preburners, the injection velocity ratios are quite high for the oxidizer preburner but the injection pressure drop will be very high on the oxidizer side. The orifices at the entrance to the injector posts could be enlarged to reduce this pressure loss but this would adversely affect the mode operation. In some of the cases where the preburner fuel injection velocities are low, the pressure drops also are low and combustion stability could be a problem. For the fuel preburner, injection velocity ratios are low in most cases, and in those cases where the oxidizer injection velocity is high, the pressure loss also will be high. In general, the coaxial injector is not considered a good configuration for liquid-liquid injection, which is the

TABLE 20. MAIN INJECTOR FLOW CHARACTERISTICS

Case	Fuel	$\dot{W}$ (lb/sec)	V_{inj} (ft/sec)	Oxidizer	$\dot{W}$ (lb/sec)	V_{inj} (ft/sec)
1	Fuel Rich* Comb Gas	798.1	560	LOX	577.2	82
2	RP-1	278.1	28	GOX	1067.2	1388
3	Fuel Rich* Comb Gas	785.8	551	LOX	551.2	78
4	CH_4	226.8	54	GOX	1110.2	1443
5	Fuel Rich* Comb Gas	836.0	587	LOX	514	73
6	C_3H_8	260.4	37	GOX	1089.6	1417
10	Fuel Rich* Comb Gas	1045.1	754	GOX	434.7	263
11	RP-1	366.2	37	GOX	1113.6	1448
12	CH_4	304.4	252	GOX	1145.6	740
13	C_3H_8	344.2	124	GOX	1078.2	1045
SSME	H_2	241.4	1506	LOX	846.9	120

* Assumes all the fuel mixes with all the combustion gases at 1600F.

TABLE 21. FUEL PREBURNER INJECTION VELOCITIES

Case	Fuel $\dot{W}$ (lb/sec)	Fuel V_{inj} (ft/sec)*	Oxidizer $\dot{W}$ (lb/sec)	Oxidizer V_{inj} (ft/sec)*
1	239.6	192	82.7	104
2	10.5	8	348.2	436
3	195.2	332	85.8	108
4	13.3	23	497.9	624
5	234.5	269	103.2	259
6	13.0	15	441.5	553
10**	126.2	227	51.6	493
11**	4.9	9	161.4	1541
12	7.8	36	290.4	364
13	13.2	21	448.0	561
SSME	87.2	898	85.8	107

* Flow Area = 0.025 ft^2 (F), 0.0114 ft^2 (Ox)

** Flow Area = 0.0111 ft^2 (F), 0.0039 ft^2 (Ox)

TABLE 22. OXIDIZER PREBURNERS INJECTION VELOCITIES

	Fuel		Oxidizer	
Case	$\dot{W}$ (lb-sec)	V_{inj} (ft/sec)	$\dot{W}$ (lb/sec)	V_{inj} (ft/sec)
1	11.7	21	385.1	1418
2	11.7	21	385.1	1418
3	11.9	45	448	1649
4	11.9	45	448	1649
5	13.1	34	446.3	1643
6	13.1	34	446.3	1643
10	18.3	15	604.1	1963
11	18.3	15	604.1	1963
12	9.8	101	368.8	1358
13	9.6	35	328.0	1207
SSME	36.1	830	32.3	117

TABLE 23. GAS GENERATOR CYCLES

Case	Fuel	T_{inj} (R)	ρ_{inj} (lb/ft^3)	$\dot{W}_{inj}$ (lb/sec)	V_{inj}* (ft/sec)	Oxidizer (1)	W_{inj} (lb/sec)	V_{inj}** (ft/sec)
7	RP-1	540	50	335.7	34	LOX	1042.1	147
8	CH_4	330	22.8	277	61	LOX	1068.2	151
9	C_3H_8	540	34.7	314	46	LOX	1043.3	147
14	CH_4	830	8.5	306.5	182	LOX	1073.4	152
15	C_3H_8	840	24.9	346.9	70	LOX	1040.9	147
O_2/H_2 SSME	H_2	300	2.35	122.3	263	LOX	9[illegible]4.1	132

* $A_{inj} = 0.1979\ ft^2$

** $A_{inj} = 0.1012\ ft^2$

(1) $T_{inj} = 190F$, $\rho_{inj} = 70\ lb/ft^3$

condition for these preburners. Based on this rather general analysis, it appears that the preburners cannot be directly substituted into these candidate cases.

In the case of systems No. 2, 4, 6, and 11 through 13, all preburners are oxidizer rich and all of the oxidizer enters the main combustion chamber through this source. There are two alternatives as to how this hot, oxidizer-rich flow can be introduced into the main chamber. It could be injected through the hot-gas manifold and through the annulus in the injector element. The main problem with this approach is that in mode 2, the flow through this side of the injector would suddenly become fuel rich, as this is the normal mode of operation for the SSME. A mode 2 cycle balance cannot be achieved with both preburners fuel rich. There would also be a switch from fuel to oxidizer on the other side of the injector. This presents a detonation problem that is unacceptable.

The other approach is to inject the hot oxidizer-rich gases through the oxidizer dome of the SSME. Some means of cooling the dome must be provided. Based on the injector flow areas, injection velocity calculations show large velocity differential between the fuel and oxidizer. The present injector should provide adequate velocity ratio for good atomization and mixing. It should be noted that, in these cases, the main injector has liquid fuel through the annulus and gases through the oxidizer post. Due to the low liquid flowrate and large annular flow area, liquid injection velocities and pressure drops appear to be too low and will be prone to low-order feed system-coupled instability.

To correct the instability problem, orifices can be installed in the fuel annulus of the injector element as in the SSME oxidizer injection elements. However, this will create excessive pressure drop in mode 2 and the H_2 is injected as a gas. A solution to this problem would be adjustable orifices for either the annulus or the post. However, this obviously requires considerable development and a new injector.

Cases 10 and 11 are both oxygen cooled. Due to the possible detonation problem with direct contact of cold liquid oxygen and RP-1, warm oxygen obtained by mixing the cooling circuit flow with the remaining liquid oxygen is necessary. Case No. 10, with both oxidizer-rich and fuel-rich preburners, will have the same problem as th t described for cases 1, 3, and 5. Case 11 is similar to cases 2, 4, and 6 with respect to the preburner and main injector problems. It is also necessary to use the SSME oxidizer preburner as the fuel preburner in these two cases since there is a large amount of oxygen to be pumped at the high pressure and a considerably high turbine flow is required. The rearrangement of these components may present some hardware interface and packaging problems.

The main injector fuel and oxidizer velocities are presented in Table 23 for the gas generator cycles defined in cases No. 7, 8, 9, 14, and 15. The SSME main injector cannot provide a large velocity differential because of the low fuel injection temperature and, hence, high density. The higher densities of the hydrocarbons further reduce their injection velocities relative to H_2. Even in cases 14 and 15, where the hydrocarbon fuel is heated in the cooling circuit, the injection velocity on the fuel side is too low. The mode 2 O_2/H_2 (gas generator cycle SSME) case shown in Table 23 is for a gas generator cycle and it is shown that the velocity ratio for this case also is low. This suggests the possibility of resizing the elements in the SSME main injector to provide acceptable pressure drops and velocities in both modes 1 and 2 operation for cases 14 and 15. In general, the coaxial injector is not suitable for liquid-liquid injection; therefore, cases 14 and 15 have the greatest potential for adaptation of the resized element SSME injector. The same situation occurs in the use of either of the SSME preburners as a gas generator. In general, they are sized for considerably higher flows and a gaseous H_2 fuel. There is a possibility of resizing the elements in one of these preburners to adapt it to one of these gas generator cycles.

Several other factors should be considered in determining the adaptability of these injectors to the candidate tripropellant engines. Little experience is available in the operation of a LOX-rich precombustor. It has been suggested that a flame holder may be required to maintain a lower mixture ratio in the center and then provide rapid mixing of the hot combustion gas and the excess oxygen. This remains to be demonstrated.

In those cases where a turbine drive gas is suddenly switched from oxidizer-rich to fuel-rich hot gases in transitioning from mode 1 to mode 2, the effect of alternately exposing materials to an oxidizing and fuel-rich environment should be investigated.

Another significant factor in injector designs for liquid oxygen is the potential formation of detonatable gel. The cold liquid oxygen mixes with and solidifies most hydrocarbon fuels if directly mixed. The coaxial injectors should be considered not suitable for LOX/RP-1 unless gaseous oxygen can be assured. Methane and propane have melting points above the LOX injection temperature (190 R) as shown in Table 24. It is hopeful that detonatable gel would not occur. However, experiments have to be performed to verify that.

In the SSME, the main injector plate is cooled by hydrogen transpiration through the rigimesh faces. More analyses should be performed if either methane or propane are used as the coolant. Oxygen is not recommended nor is RP-1. Hence, cases 10 and 11 will require hydrogen cooling for the main injector.

TABLE 24. FUEL PHASE CHANGE CONDITIONS

	Normal Boiling Point, R	Normal Melting Point, R
RP-1	882	405
C_3H_8	201	163
CH_4	416	154

TASK VI - TEST PLANS

The objective of this task was to identify critical areas for experimental component evaluation based on the results of Tasks I through V. Based on this information, test plans were generated for additional testing to complement the current NASA test plans for 40K hardware with LOX/RP-1. Since the tripropellant engine studies have not identified any SSME components that would have direct applicability to a tripropellant engine, these test plans will not be directed toward verifying component adaptability but will be geared toward more general technology questions that arose during the course of these studies. The results of the proposed testing, therefore, would have a more general usefulness in that they would answer questions pertaining to the design of an all new dual-mode tripropellant engine or a single mode LOX/hydrocarbon booster engine.

NASA has already planned a comprehensive test program using the 40K SSME subscale hardware with LOX/RP-1 propellants and the test plans developed in this study are to be in addition to or complement the current NASA plans. The results of this study have shown that CH_4 offers some significant advantages for a dual-mode tripropellant engine or in any LOX/hydrogen booster engine system. Therefore, the test plans to be studied in this task will be primarily for LOX/CH_4 propellants. However, some of the tests would be of equal importance with any hydrocarbon fuel being considered. A list of test plan objectives and reasons why the technology demonstrations are needed is presented in Fig. 47.

Low Mixture Ratio Gas Properties

Previous experience in the F-1 and H-1 engine programs has shown that considerable difference exists between the low mixture, low temperature hot combustion (LOX/RP-1) gas properties observed experimentally and those predicted with current free-energy performance codes. This is believed to be primarily due to the high amount of carbon formed in the very fuel-rich combustion process. This comparison has been demonstrated only at low combustion pressures (<1000 psia); the effect is unknown at higher pressures and, therefore, is a subject for an experimental test program. It is anticipated that LOX/C_3H_8 will

TEST PLAN NO.	TECHNOLOGY TEST PLAN SUBJECT	REASONS WHY TECHNOLOGY DEMONSTRATIONS ARE NEEDED
1	LOW MIXTURE RATIO, HIGH P_c, LOX/HYDROCARBON COMBUSTION GAS PROPERTIES (T, C_p, γ, Mwt VS M R.)	NEEDED FOR GAS GENERATOR OR PREBURNER AND ENGINE SYSTEM DESIGN AND ANALYSIS. NO COMBUSTION GAS PROPERTY DATA AVAILABLE FOR LOX/CH_4, LOX/C_3H_8. ONLY LOW P_c DATA AVAILABLE FOR LOX/RP-1 AND THIS DOES NOT AGREE WITH THEORETICAL PREDICTIONS. P.B. INJECTOR PERFORMANCE UNKNOWN. CARBON DEPOSITION MAY BE CONCERN.
2	LOX RICH PREBURNER DEMONSTRATION	ONE LOX RICH P.B. WILL BE REQUIRED FOR LOX/HYDROCARBON HIGH P_c STAGED COMBUSTION ENGINES UNLESS TURBINE INLET TEMPERATURES ABOVE 2000°R ARE PERMITTED. THERE IS NO EXPERIENCE WITH HIGH MIXTURE RATIO P.B. OPERATION AND EXPERIMENTAL GAS PROPERTIES AND IGNITION CHARACTERISTICS ARE UNKNOWN.
3	REGENERATIVE COOLING WITH HYDROCARBON FUEL	NO HIGH P_c REGENERATIVE CH_4 COOLING EXPERIENCE. DESIGN ANALYSIS MUST BE VERIFIED
4	DUAL FUEL OPERATIONAL TRANSITION	A DUAL FUEL ENGINE REQUIRES THIS CAPABILITY. NO EXPERIENCE WITH TRANSITION FROM HYDROCARBON FUEL TO H_2. SOME QUESTIONS MUST BE EXPERIMENTALLY VERIFIED.
5	INCREASED TURBINE DRIVE GAS TEMPERATURES >2000°R	HIGHER TURBINE DRIVE GAS TEMPERATURES WILL ALLOW INCREASED ENGINE SYSTEM P_c AND ALL FUEL RICH PREBURNERS. HIGH TEMP P.B. OPERATING EXPERIENCE CAN BE GAINED ALONG WITH TESTING OF TURBINE NOZZLE AND BLADE MATERIALS IN HOT GAS ENVIRONMENT.
6	STAGED COMBUSTION WITH LOX/CH_4	NO EXPERIENCE WITH LOX/CH_4 STAGED COMBUSTION, NEED PREBURNER, MAIN CHAMBER, AND REGEN COOLING DEMONSTRATION. DESIGN ANALYSIS MUST BE VERIFIED
7	COMBINED FUEL AND OXIDIZER RICH PREBURNERS IN STAGED COMBUSTION DEMONSTRATION	LOX-RICH AND FUEL RICH PREBURNER IS REQUIRED TO ACHIEVE A HIGH P_c STAGED COMBUSTION LOX/HYDROCARBON ENGINE BALANCE UNLESS TURBINE INLET TEMPERATURES ABOVE 2000°R ARE PERMITTED

Figure 47. Recommended Technology Test Plans

behave much like LOX/RP-1 but that the LOX/CH_4 system may not exhibit this discrepancy between experimental and theoretical predictions of mixture ratio versus temperature.

Approach. This experimental hot-firing program would be conducted using a combustion chamber and injector capable of operating at high chamber pressures and at mixture ratios typical of the LOX/hydrocarbon preburner or gas generator low mixture ratio conditions. The combustion gas properties can be measured from pressure, temperature, and flowrate measurements collected during the test. The details of the parameters to be recorded and the necessary calculations are discussed in a later section. An alternate means of estimating combustion gas properties is through chemical analysis of combustion gas samples collected during the hot-firing test. Additional test objectives such as preburner injector performance and the investigation of potential carbon deposition problems can be evaluated during these test series by including the necessary test parameter measurement techniques.

Two approaches can be taken regarding the hardware to be used in this program. The existing SSME subscale 40K preburner chamber and injector could be used or a new sub-subscale preburner could be fabricated for this technology program. The existing coaxial preburner injector element was analyzed to determine if this injector could be used with LOX/CH_4.

It was assumed that liquid oxygen and ambient temperature gaseous methane would be supplied to the injector at a mixture ratio of approximately 0.44 (combustion temperature of 2000 R). It was found the total flowrates of from 20 to 45 lb/sec would result in reasonable fuel and oxidizer injection velocities. These flows correspond to chamber pressures of 2500 to 3500 for the current throat area. However, the injection velocity ratio is somewhat low and mixing efficiency may not be as good as with the H_2 fuel.

Increasing the chamber length would compensate for this reduced mixing efficiency. This could be accomplished by adding a spool to the existing chamber after first testing with the existing 12-inch chamber length. Performance would be more questionable with higher density fuels such as C_3h_8 and RP-1. A plate-type nozzle with multiple small holes or slots could be used in the preliminary tests to evaluate the carbon deposition at the low mixture ratios associated with a gas generator or preburner.

The fabrication of a new subscale preburner would provide more versatility in the test program in that test costs can be reduced by the smaller flowrates, other test facilities can be considered, multiple injectors could be fabricated for other fuels, and the hardware can be designed to specifically accomplish the test objectives and additional injectors can be provided to investigate high mixture ratio operation. A typical sub-subscale preburner design would incorporate injector elements much like those in the existing 40K preburner except resized to provide the proper injection pressure drops and velocities. A 2- to 3-inch-diameter injector face would incorporate from 19 to 41 injector elements, thus providing high performance.

Property Measurement. The goal of this experimental procedure is to measure combustion product properties, including:

T_c - chamber temperature
M_w - hot-gas molecular weight
C_p - hot-gas specific heat at constant pressure
γ - ratio of specific heats

A schematic diagram of the proposed experimental setup is shown in Fig. 48. It consists of the combustion chamber and a converging-diverging nozzle. The throat and exit areas of the nozzle are known, as well as the total mass flow (in) during a particular test.

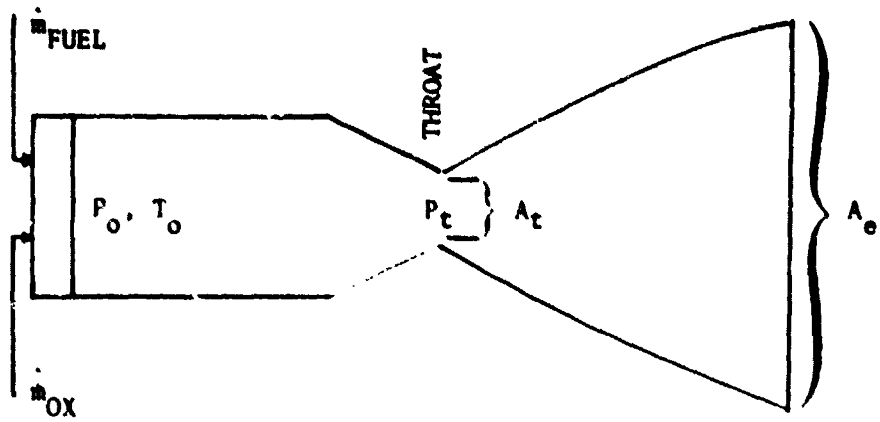


Figure 48. Proposed Experimental Setup

The combustor and nozzle are insulated to prevent heat loss from the gas. The diverging section is not necessary for performance of the experiment, since all that is required is to bring the gas to sonic velocity (M=1) at a throat, so that P_o* and T_o* may be measured.

Two alternate methods for determining the properties of hot gas are detailed below.

1. Assuming 1-D Isentropic Flow With no Heat Loss (Q = 0)

In this case, the equations of flow in a 1-D converging-diverging channel apply directly. Chamber pressure (P_o) and temperature (T_o) are measured directly. To measure gas γ, note that

$$\frac{dP_o}{P_o} = \frac{-\gamma M^2}{2} \frac{dT_o}{T_o} \tag{1}$$

which upon integration gives

$$\frac{P_{o1}}{P_{o2}} = \left(\frac{T_{o1}}{T_{o2}}\right)^{\frac{-\gamma M^2}{2}} \tag{1^1}$$

In particular, at the throat,

$$\left(\frac{P_{o_1}}{P_o^*}\right) = \left(\frac{T_{o_1}}{T_o^*}\right)^{-\gamma/2} \tag{2}$$

Solving for γ:

$$\gamma = -2 \frac{\ell_n \left(P_{o_1}/P_o^*\right)}{\ell_n \, T_{o_1}/T_o^*} \tag{3}$$

The assumption of ideal gas behavior gives

$$\boxed{C_p = \left(\frac{\gamma}{\gamma - 1}\right) R} \tag{4}$$

with $R = 1.987 \ \frac{\text{cal}}{\text{g mole}} \ K$

Molecular weight is determined by noting that the nozzle is choked at the throat, so that:

$$\dot{M} = \sqrt{\frac{g_c \, \gamma \, M_w}{R \, T_o}} \ P_o \, A_t \left[1 + \frac{\gamma - 1}{2}\right]^{-\frac{(\gamma+1)}{2(\gamma-1)}} \tag{5}$$

Solving for M_w,

$$\boxed{M_w = \frac{\dot{M}^2 \, R \, T_o}{g_c \, \gamma \, P_o^2 \, A_t^2 \left[1 + \frac{\gamma - 1}{2}\right]^{-\left(\frac{\gamma+1}{\gamma-1}\right)}}} \tag{6}$$

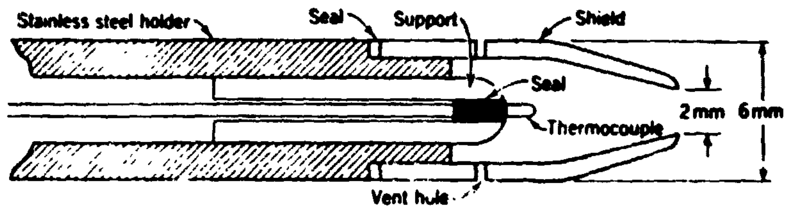
where

$g_c = 32.174 \ \frac{\text{lbm - ft}}{\text{lbf - sec}^2}$

$R = 1545 \ \frac{\text{ft-lbf}}{\text{lb mole R}}$

T_o = stagnation (chamber) temperature, R

Throat stagnation pressure and temperature may be measured with total temperature and total pressure probles like that shown below:



Design of total temperature probe. From Eva M. Winkler, *J. Appl. Phys.*, 25 (1954), p. 231.

II. Gas Sampling and Analysis

An alternate approach to obtaining hot-gas γ, C_p, and M_w is through chemical analysis of a sample of gas. Mass spectroscopy, gas chronatography, or both can be used to determine the constituents of the combustion products. Gas properties can then be inferred from the mole fractions of the constituents:

$$\boxed{\overline{M_w} = \frac{\sum_{i=1}^{N} X_i M_{w_i}}{\sum_{i=1}^{N} X_i}} \tag{7}$$

where

X_i = mole fraction of $i^{\underline{th}}$ constituent

M_{w_i} = molecular weight of $i^{\underline{th}}$ constituent

N = No. of constituents

$$\boxed{\overline{C_p} = \frac{\sum_{i=1}^{N} X_i C_{p_i}}{\sum_{i=1}^{N} X_i}} \tag{8}$$

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where

$$C_{p_i} = C_p \text{ of } i^{\underline{th}} \text{ constituent } \left(\text{molar } C_p \ \frac{\text{Btu}}{\text{Mole-R}}\right)$$

and

$$\boxed{\bar{\gamma} = \left(\frac{1}{1 - R/\bar{C}_p}\right)} \tag{9}$$

Note that $\bar{\gamma}$ cannot be mole-weighted.

This method provides an alternate and independent determination of gas properties. It is recommended that both gas sample analysis and flow property measurement methods be used. Note that chamber temperature must be measured directly, as sampling cannot be used to determine T_c.

Oxidizer-Rich Preburner

Staged combustion LOX/hydrocarbon engine system power balances at chamber pressure levels of interest have shown that insufficient energy (fuel flowrate) is available to drive the turbines with all fuel-rich preburners. One alternative is to operate one or more preburners oxidizer rich since there is considerably more LOX available. This brings up numerous questions concerning the design and operation of a preburner capable of operating in a very high mixture ratio, low combustion temperature mode. Little experience is available and a test program would provide the much needed information in this area.

The lack of experience along with the need for design and operational information for LOX-rich combustion gases makes this test objective of major importance to the overall LOX/Hydrocarbon Engine Technology Program. To achieve LOX-rich combustion gases at temperatures near 2000 R, a LOX/CH_4 mixture ratio of 34:1 is required. This results in large LOX flows and very small fuel flowrates relative to current preburner injector designs. The possibility of operating the existing coaxial injector with reversed flows (oxidizer through the annulus and fuel through the center post) was investigated briefly for injection velocities and pressure drop. The LOX flow through the annulus results in reasonable pressure drops and injection velocities. However, the fuel flow in the center posts would have a very low pressure drop. A means could be devised to reduce the size of the existing orifices in the center posts, but the resulting nonoptimum injector would probably result in low performance. It is recommended that a new, optimum design, LOX-rich preburner injector be fabricated to satisfy this test objective.

A new injector could be fabricated for the existing preburner or a new sub-subscale LOX-rich preburner could be fabricated and these test run in conjunction with the fuel-rich preburner testing. Preliminary injector configuration studies have shown that a pentad (4 oxidizer - 1 fuel) injector element might be attractive for this application because of the low fuel stream energy. The four oxidizer streams provide near-equal orifice sizing and the impinging oxidizer streams would provide the necessary atomization, and the small fuel stream is introduced into the middle of the impingement zone. Also, because of the relative fuel and oxidizer flowrates, this injector will probably use the oxidizer for face cooling.

Hydrocarbon Cooling

Little information is available pertaining to experience with any of the hydrocarbon fuels as regenerative coolants. Analytical predictions indicate that the RP-1 is a poor coolant and there is less interest in demonstrating its chamber cooling capabilities. However, CH_4 appears to be an attractive candidate for future LOX/hydrocarbon booster engine systems and a hot-firing cooling demonstration would provide valuable information in the further study and comparison of the candidate systems and in the actual design of a chamber. It is expected that this regenerative cooling demonstration could be conducted with the 40K hardware in conjunction with an injector/combustion process demonstration. The approach would be first to perform calorimeter chamber tests to determine the heat flux profile in the main chamber with LOX/CH_4 combustion. With this information, predictions for wall temperatures and coolant temperatures for the regeneratively cooled chamber could be improved. A subsequent test with the channel wall regeneratively cooled chamber would finally demonstrate the CH_4 cooling capabilities and provide valuable design information necessary to design a full-scale engine.

The regenrative cooling testing can be conducted using the existing preburner and main chamber injectors in a staged combustion mode or a new liquid/gas LOX/CH_4 injector and the existing 40K channel wall main chamber can be used in a gas generator mode. Preliminary calculations indicate that the existing 40K preburner and main chamber injectors will provide satisfactory performance without any modifications when operated in a staged combustion configuration and using gaseous CH_4. However, the CH_4 density results in a very low injection pressure drop and velocity if gaseous CH_4 is introduced as in a gas generator cycle. The fuel annulus gap must be reduced to near 0.010 inch to maintain a reasonable pressure drop and injection velocity. It is anticipated that this very thin annulus stream and the relatively large core stream will result in poor performance. A new injector is recommended for operation with LOX and gaseous CH_4 in a gas generator configuration.

As part of this effort to consider using the 40K hardware to experimentally investigate an O_2/CH_4 combustion with CH_4 cooling, a thermal analysis was conducted for the calorimeter and CH_4 cooled channel wall chambers.

Heat Transfer Analysis. Ratioing of the combustion gas properties and flowrates, as in the case of the full-scale SSME heat transfer analysis conducted in Task II of this study effort, gives a heat transfer coefficient profile for the O_2/CH_4 propellants of 63% of the O_2/H_2 propellants at the same chamber pressure. The O_2/H_2 heat transfer coefficient profile utilized is 95% of the experimentally determined profile corrected to 3000 psia. For this analysis, it has been assumed that there is no carbon coating on the hot-gas wall.

For the calorimeter chamber, the water burnout heat flux is calculated from the following equation:

$$Q/A_{BO} = 0.00908\ V^{0.5}\ (T_{SAT} - T_c)$$

The methane coolant heat transfer coefficients are calculated using the following equation:

$$H_c = 0.023 \frac{k}{D_h} Re^{0.8} Pr^{0.4} \phi_r$$

The roughness enhancement (ϕ_r) is calculated by taking the ratio of the actual friction factor to the friction factor assuming a smooth surface. An average roughness enhancement value of 1.4 is used. A coolant inlet temperature of -240 F is used. A coolant inlet pressure of 4500 psia is assumed for operation at a chamber pressure of 3000 psia.

Calorimeter Chamber Results. The water coolant maximum log pressure available now at the MSFC Test Facility is 1500 psia. For this pressure, the optimum velocity in the small throat region channels is approximately 170 ft/sec. For the large combustion zone channels, the optimum velocity is 120 ft/sec. For 40K calorimeter chamber No. 1, the burnout heat flux profile using these water velocities is shown in Fig. 49. Since the heat flux in the combustion zone and expansion section is much lower than in the throat region, the optimum velocity (120 ft/sec) need not be used in these regions.

The throat heat flux and the throat heat flux relative to the throat maximum burnout heat flux (for a water log pressure of 1500 psia) are shown in Fig. 50 as functions of chamber pressure. For a burnout margin of 10% the maximum chamber pressure at which the calorimeter chamber can be operated is 1800 psia. The throat heat flux at this chamber pressure is 41 Btu/in^2-sec.

Methane-Cooled Chamber Results. The wall temperature in the throat region (-1") and the combustion zone (-9") at a chamber pressure of 3000 psia is shown in Fig. 51a as a function of the methane coolant flowrate. This figure is for an uppass cooling circuit. The maximum wall temperature occurs in the combustion zone and the required coolant flowrate to keep the maximum wall temperature at 1000 F is 30 lbm/sec.

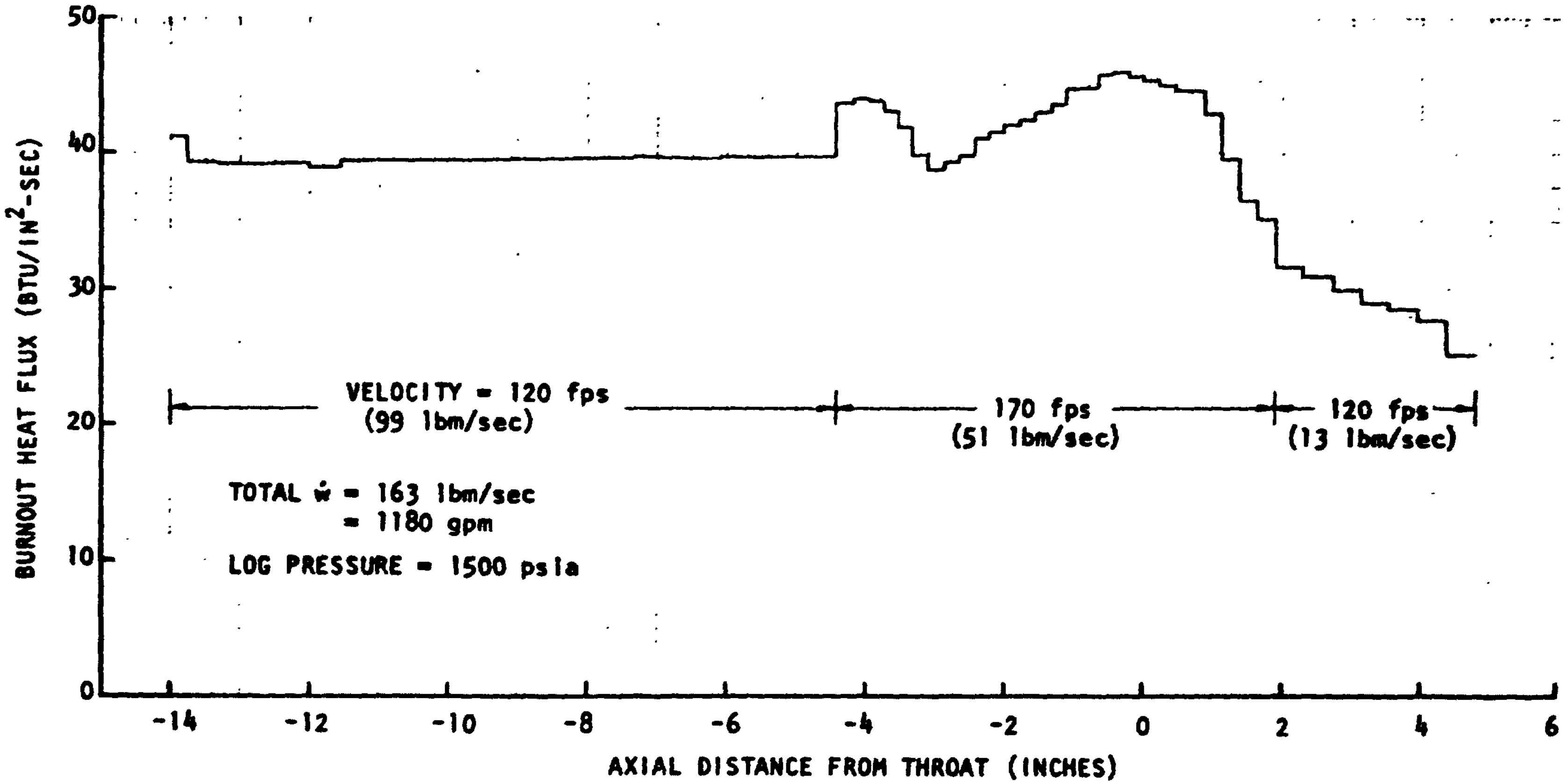

Figure 49. Burnout Heat Flux Profile for 40K Calorimeter Chamber No. 1

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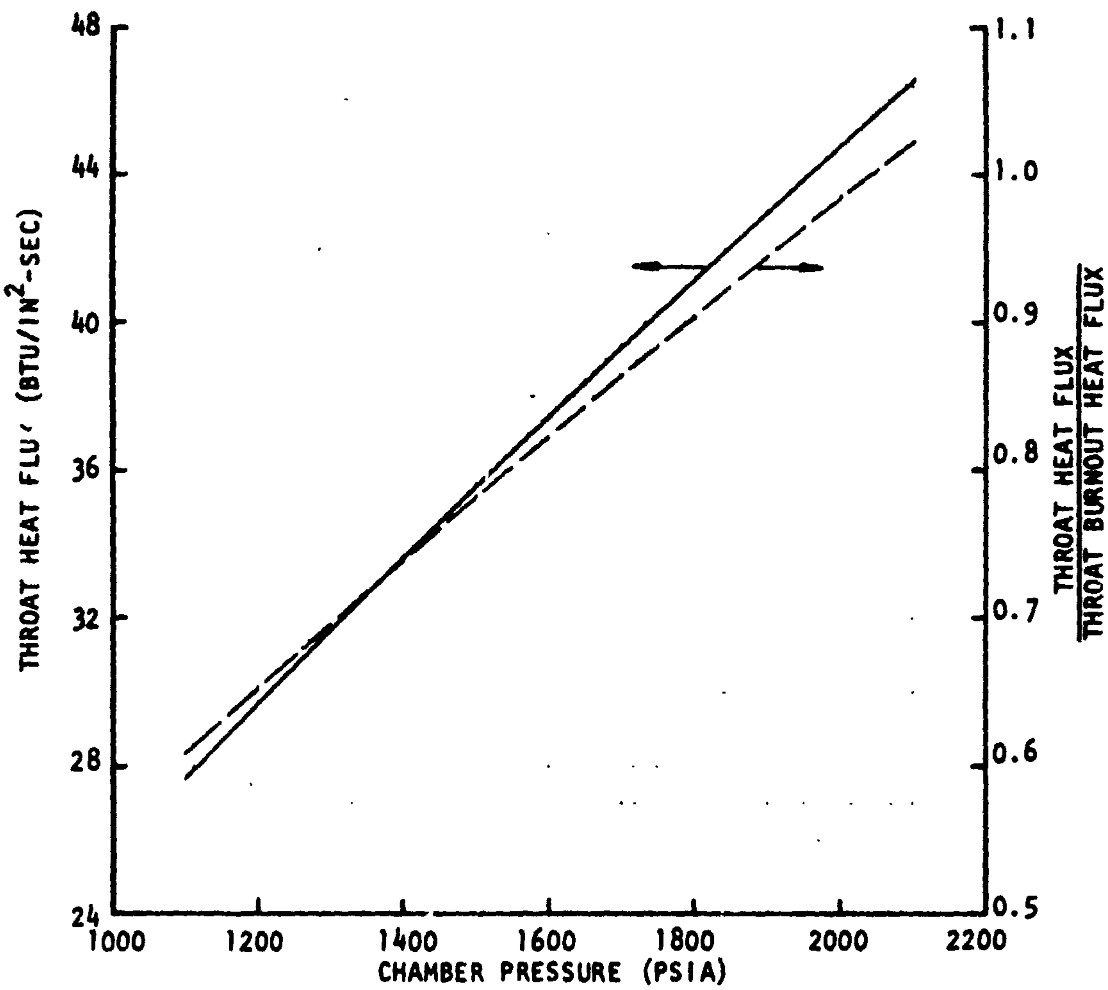

Figure 50. 40K Calorimeter Chamber Throat Heat Flux for O_2/CH_4 Propellants

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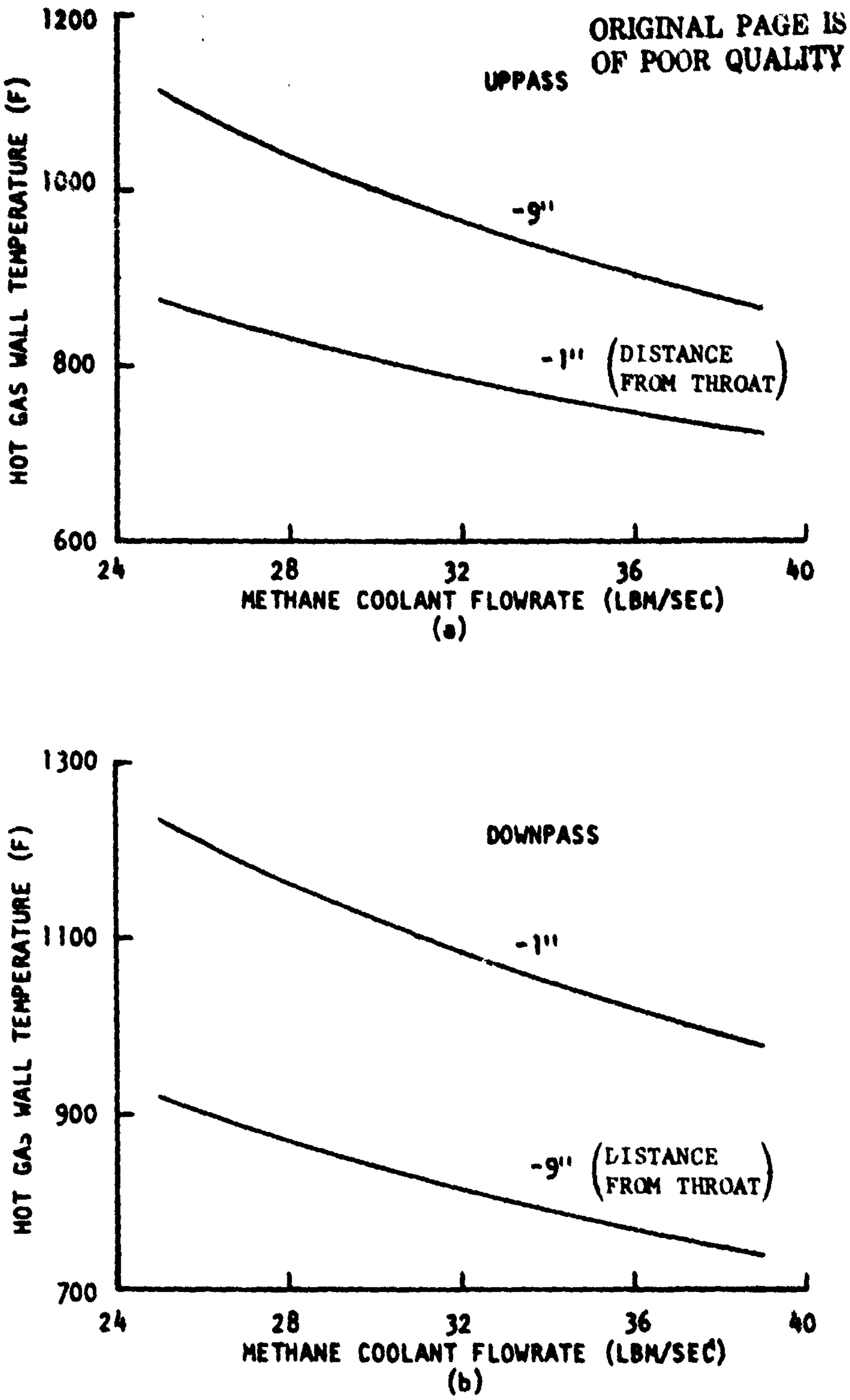

Figure 51. Coolant Flowrate Requirements for the 40K Combustor at P_c=3000 psia (O_2/CH_4 Propellants)

In an effort to decrease the required coolant flowrate, a downpass cooling circuit was considered (Fig. 51b). The maximum wall temperature now occurs in the throat region. However, the required flowrate to maintain a 1000 F wall temperature has been increased to 38 lbm/sec (due to increased bulk temperature and lack of curvature enhancement) so that there is no coolant flowrate benefit to be gained by going to a downpass cooling circuit.

Hot-gas wall temperature versus axial position curves for chamber pressures of 2000 and 3000 psia are shown in Fig. 52. The curves are for an uppass cooling circuit with the coolant flowrates which give a maximum wall temperature of 1000 F. No change has been made to the combustor coolant channel geometry. For both chamber pressures, the coolant flowrate requirement is low enough so that the chamber could be run in the regenerative cooling mode if desired. For the chamber pressure equal to 3000 psia case, the coolant pressure drop is 515 psia and the coolant temperature rise is 360 F. For an inlet temperature of -240 F, this results in an outlet temperature of 120 F. The chamber heat load is 9000 Btu/sec with a throat (-1") heat flux of 65 Btu/in.^{2}-sec.

Dual-Fuel Operational Transition

One of the biggest questions that arises in the tripropellant engine concept concerns the transition from a hydrocarbon fuel during mode 1 to H_2 in mode 2. It is not known whether some intermediate purging of the injector, manifold, and cooling circuit will be required to prevent freezing of the residual hydrocarbon by the entering LH_2. Based on hardware thermal response experience on the SSME, there is a good possibility that the thrust chamber residual heat will provide sufficient heat during the transition from CH_4 to LH_2 fuels to avoid the need for the intermediate purge. A detailed transient heat transfer analysis must be conducted to verify this for the demonstration hardware. The demonstration testing could be conducted using the existing 40K calorimeter and regenerative main chambers. The initial pressure-fed injector transitional operational testing would be conducted with the calorimeter chamber and the pressure-fed regenerative cooling transition with the regenerative chamber. Injector flowrates, pressure drops, and injection velocities were calculated for operation with both

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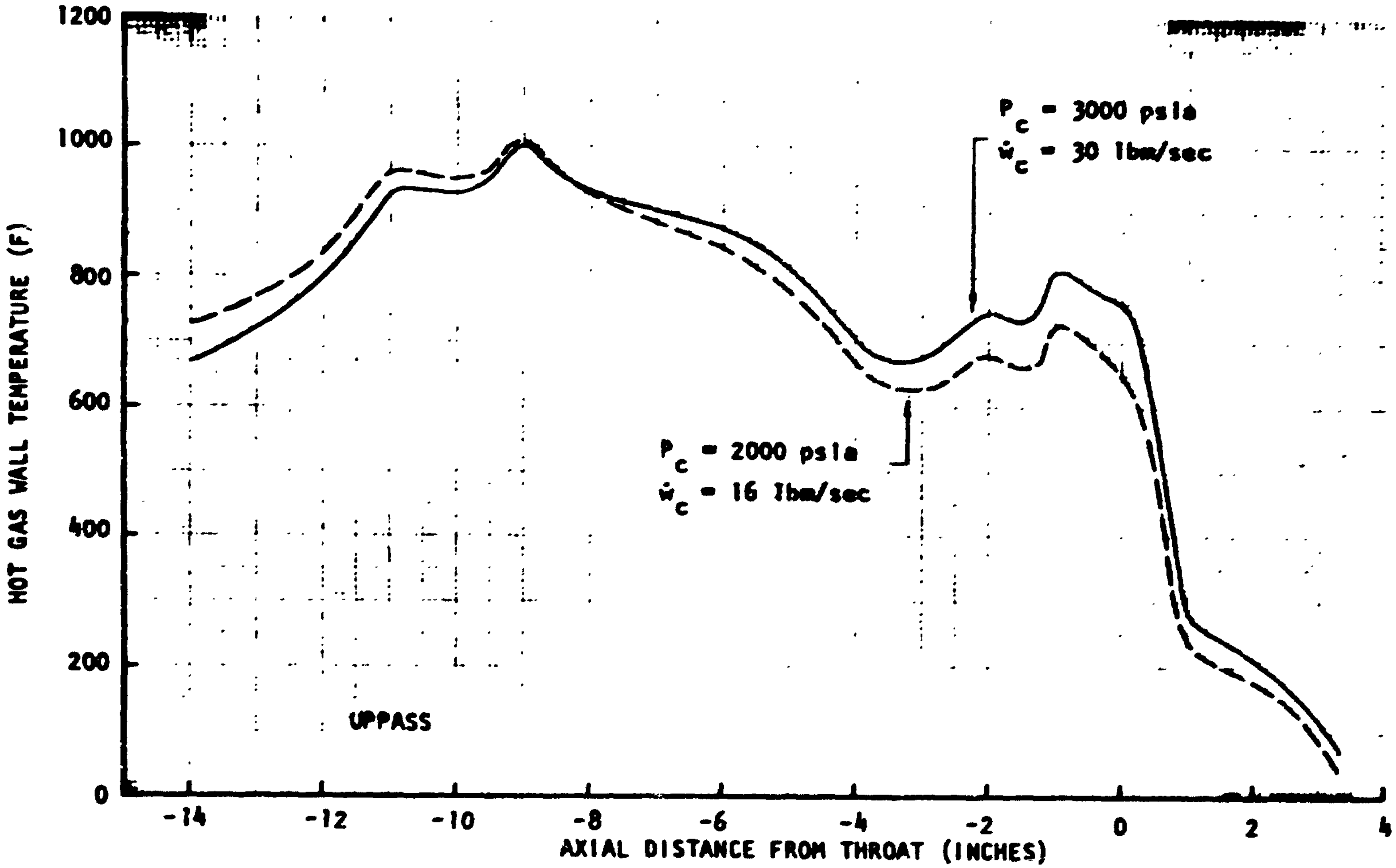

Figure 52. Wall Temperature Profiles for the 40K Combustor with Methane Cooling (O_2/CH_4 Propellants)

CH_4 and H_2 fuels. The results indicate that the existing preburner and main chamber injectors are capable of high-performance operation in the staged combustion mode. A new liquid/gas, LOX/CH_4 injector that NASA-MSFC is currently in the process of procuring is also capable of the dual-mode operation in a gas generator cycle mode. Important injector operating parameters for mode 1 (CH_4 fuel) and mode 2 (H_2 fuel) are shown in Table 25. Regenerative cooling capabilities are adequate since the chamber was designed to be cooled with H_2 and the analysis shown in the previous test plan shows that the chamber also can be cooled with methane.

Higher Turbine Drive Gas Temperatures

The engine cycle balances conducted during this study for the staged combustion cycle have shown that to achieve an engine balance with all preburners fuel rich, turbine inlet gas temperatures exceeding 2000 R are required. This requires that the preburner operate at higher mixture ratios than the current SSME design and that it must be able to withstand the additional heat load. The higher temperature turbine inlet gases also will have a significant impact on the turbine operational limits. A demonstration test could be conducted with the existing 40K preburner operating at combustion temperatures of 2000 to 2500 R with instrumented ceramic turbine blade simulators in the hot-gas stream. A brief analysis was conducted to evaluate the capability of the existing preburner operating at these elevated temperatures. It was found that the injector flow parameters with LOX/CH_4 will provide adequate performance. Some injector face cooling is provided by the fuel flow and this should be adequate for temperatures up to 2500 R. The preburner body is heat sink cooled and the inside surface temperature transient as function of the hot-gas temperature is shown in Fig. 53. The melting temperature of the material is approximately 2900 R. Some stress calculations must be performed to determine the limiting hot-gas temperature and hot-firing durations.

Staged Combustion with LOX/Hydrocarbon

A 40K demonstration of a staged combustion system using LOX/hydrocarbon propellants would provide information concerning ignition, carbon formation,

TABLE 25. NEW GAS GENERATOR CYCLE CH_4/LOX INJECTOR

Methane Operation	Hydrogen Operation
P_c = 3000 psia	P_c = 3000 psia
MR = 3.5	MR = 6.0
c* = 6131 ft/sec	c* = 7750 ft/sec
ρ_{CH_4} = 9.1 lb/ft^3 T = 580 R, P = 3000 psia	ρ_{H_2} = 1.4 lb/ft^3 T = 350 R, P = 3000 psia
w_o = 104 lb/sec	w_o = 91 lb/sec
w_{CH_4} = 30 lb/sec	w_{H_2} = 15 lb/sec
V_o = 120 ft/sec	V_o = 105 ft/sec
V_{CH_4} = 530 ft/sec	V_{H_2} = 1725 ft/sec
ΔP_o = 600 psi	ΔP_o = 460 psi
ΔP_{CH_4} = 360 psi	ΔP_{H_2} = 580 psi

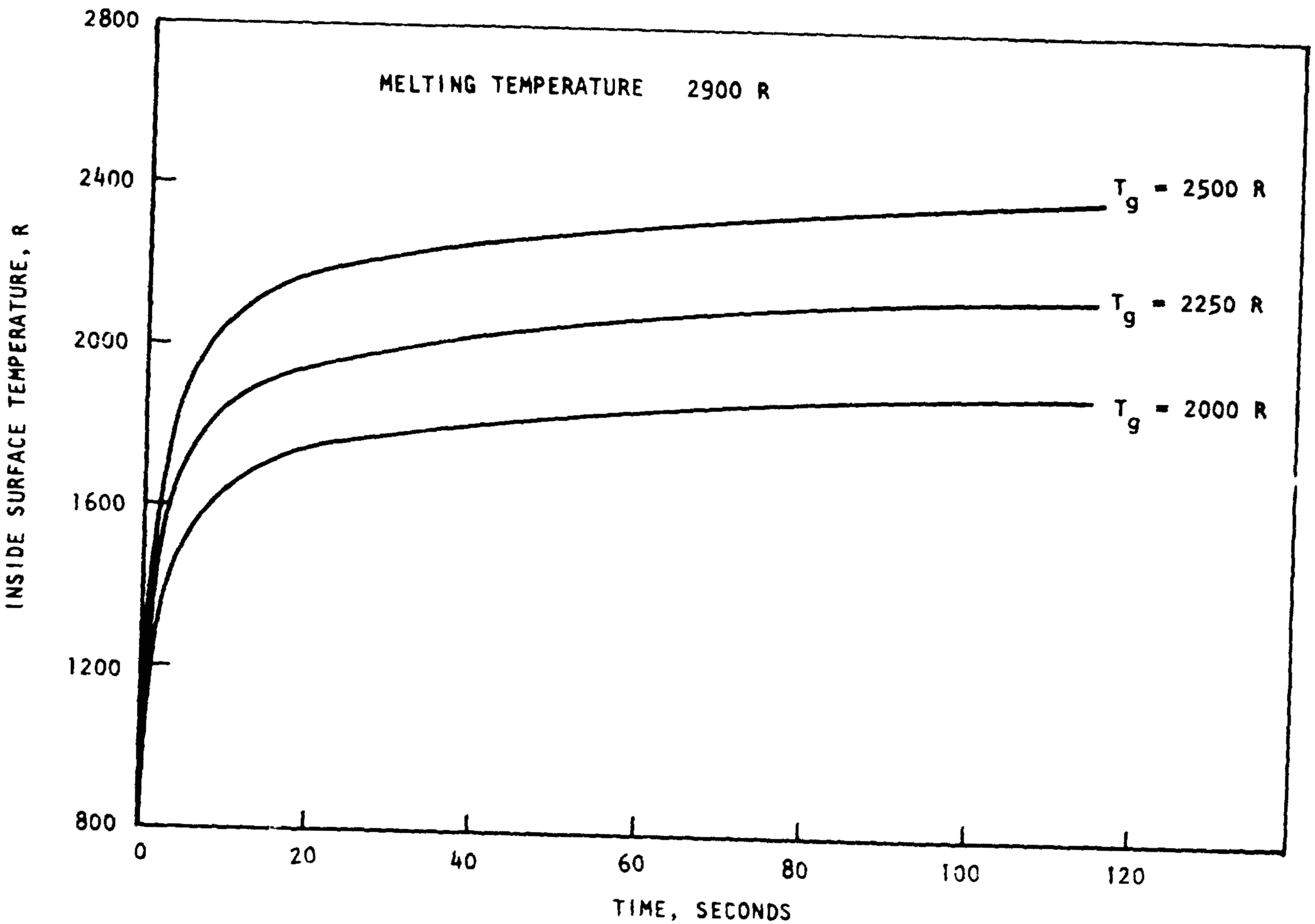

Figure 53. 40K Preburner Body Temperature Profiles

combustion heat transfer characteristics, and stability of this system. These test objectives could possibly be achieved in conjunction with the hydrocarbon cooling and injector testing. As stated in previous test plan discussions, it has been found that the existing 40K preburner and main chamber injector are capable of staged combustion operation with LOX/CH_4 propellants. Initial testing would be conducted with the calorimeter chamber to establish heat flux profiles and preburner and main injector performance. These data would then be used to predict the regenerative chamber operating conditions. The final tests would be conducted with the regeneratively cooled chamber to demonstrate regeneratively cooled, staged combustion operation with LOX/CH_4. The allowable operating conditions and cooling capabilities of these two chambers are discussed in detail in the regenerative cooling demonstration test plan discussion.

Combined Fuel and Oxidizer Rich Preburners

One of the alternatives for achieving adequate turbine drive gas energy for the LOX/hydrocarbon staged combustion systems is to operate the fuel preburner fuel rich and the oxidizer preburner oxidizer rich. This concept requires a new main injector to accommodate the two hot-gas streams. The current NASA plans already call for the fabrication of new fuel-rich and oxidizer-rich preburners of the 40K size. These preburners could be used in conjunction with the available 40K main chamber and a new main chamber injector to provide all of the hardware necessary for this demonstration. The existing calorimeter main chamber would be used to establish the heat flux profile and demonstrate injector performance. The heat transfer data obtained in these tests would be used to predict the regenerative cooling characteristics and subsequent testing would be conducted with the regeneratively cooled chamber. The fuel-rich and oxidizer-rich preburner injector requirements were discussed in previous test plans.

The new item that is required for this test demonstration is a main injector that will accept an oxidizer-rich hot-gas stream and a fuel-rich hot-gas stream. There is no previous experience with this type of injector and it is expected to be a complex design problem. The concept offers the potential for high performance due to the elimination of vaporization problems and the gas/gas propellant injection is considered to be less susceptible to combustion instability. The gas/gas injection also should result in less streaking of the chamber wall. Candidate injector element types to be considered include coaxial, impinging (doublet or triplet), and showerhead. Some of the major design problems that will be encountered in this gas/gas injector design will be in the areas of injector and injector-to-thrust chamber thermal stress, injector face cooling, hot-gas manifold cooling, and stability aid cooling.

SYMBOL NO. NCLATURE

P_0 = combustion pressure

T_0 = combustion gas temperature

$\dot{W}_g$ = combustion gas flowrate

MR = combustion gas mixture ratio (o/f)

MW = molecular weight of combustion gases

hg = hot gas heat transfer coefficient

K = gas thermal conductivity

μ = gas viscosity

X = distance

G = mass velocity

$$Q/A)_{BO} = 0.00908\, V^{0.5}\, (T_{SAT} - T_c)$$

$Q/A)_{BO}$ = burnout heat flux (Btu/in.^{2}-sec)

V = water velocity (ft/sec)

T_{SAT} = water saturation temperature (F)

T_c = water temperature (F)

$$h_c = 0.023\, \frac{k}{D_h}\, R_e^{0.8}\, P_r^{0.4}\, \phi_r$$

h_c = coolant coefficient (Btu/in.^{2}-sec-F)

k = thermal conductivity (Btu/in.-sec-F)

D_h = hydraulic diameter (in.)

R_e = Reynolds Number = G D_h/μ (dimensionless)

- G = mass velocity (lbm/in.^{2}-sec)
- D_h = hydraulic d meter (in.)
- μ = dynamic visoscity (lbm/in.-sec)

P_r = Prandtl Number = $c_p\, \mu/k$ (dimensionless)

- C_p = specific heat (Btu/lbm-F)
- μ = dynamic viscosity (lbm/in.-sec)
- k = thermal conductivity (Btu/in.-sec-F)

ϕ_r = roughness enhancement = f_r/f_s

- f_r = friction factor for rough surface
- f_s = friction factor for smooth surface

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