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TRIPROPELLANT ENGINE STUDY

BIMONTHLY TECHNICAL PROGRESS REPORT

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PREPARED FOR

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# INTRODUCTION

The advanced vehicle studies that have been conducted for the 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.

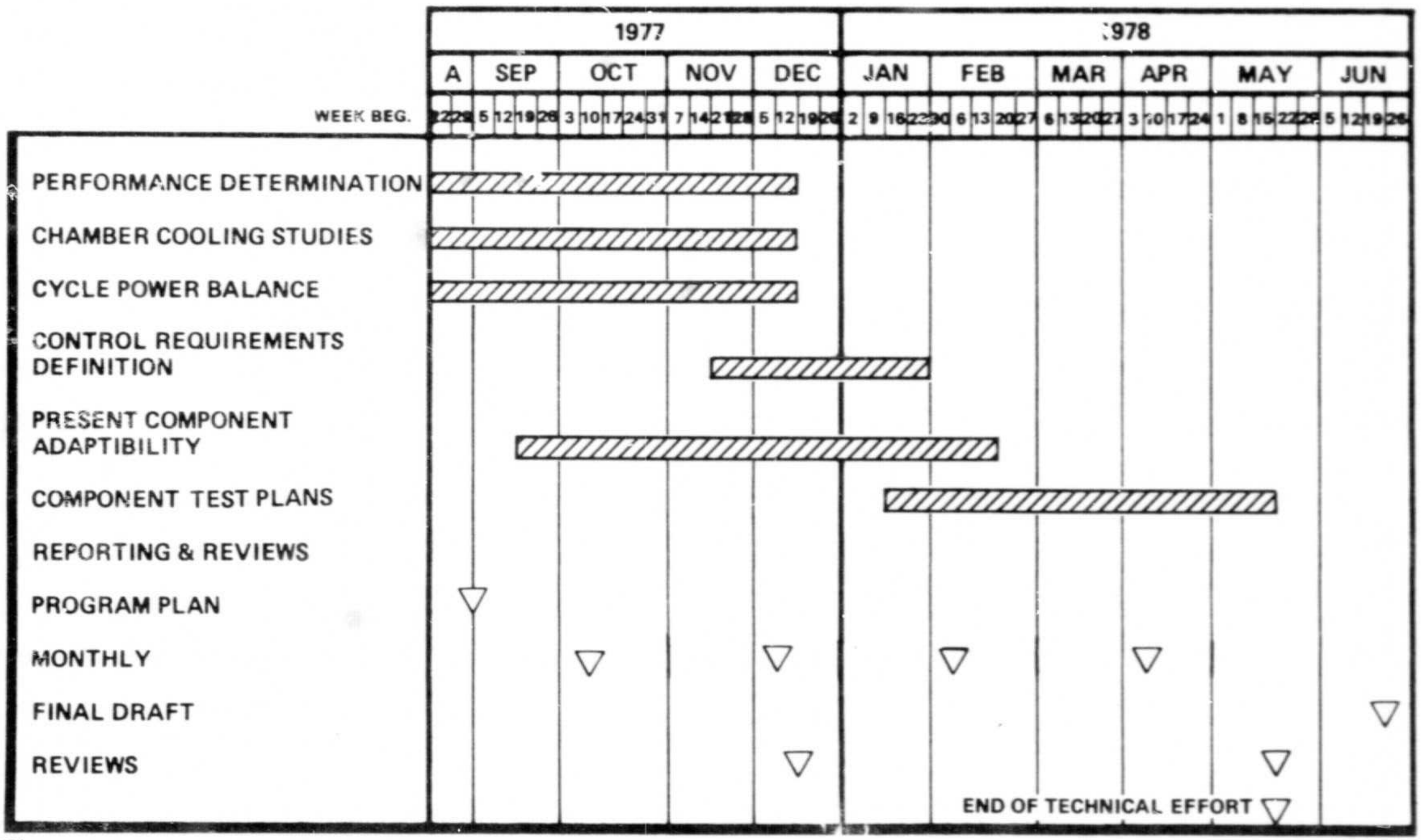
The Space Shuttle Main Engine (SSME) is a high-pressure oxygen/hydrogen engine that potentially could be modified for a dual-mode operation. Such a modification would minimize development cost of a dual-mode engine by maximizing utilization of existing hardware.

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

The study program is for nine months of technical effort followed by a period of a final report (Fig. 1) The study is subdivided into seven tasks including a reporting task.

This study is 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 are to be considered. Engine cycle concepts are formulated for LOX/RP-1, LOX/$CH_4$ and LOX/$C_3H_8$ propellants. Each system must also be capable of operating

# TRIPROPELLANT ENGINE STUDY SCHEDULE

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sequentially with $LOX/H_2$. Flowrates and operating conditions will be established for this initial set of engine systems and the adaptability of the major components of the SSME will be investigated. The end result will be the identification of high $P_c$ engine system concepts that make maximum use of the SSME hardware and best satisfy the dual mode booster engine system application.

Based on the results of the engine system concept studies, recommendations will be made for additional testing to compliment the already planned experimental program using the existing test facility and 40K test hardware available at MSFC. A test plan will be prepared to establish the objectives of each additional experimental test phase.

## SUMMARY

This first bimonthly progress report covers the work conducted from program start (22 August 1977) to 30 September 1977. All of the work conducted has been devoted to the first three tasks. Thermochemical equilibrium performance data were assembled to establish the expected performance calculations of the mode 1 engine propellant combinations and thermodynamic and transport data for the products of combustion. Turbine drive gas characteristics were also established. Thrust chamber and nozzle cooling studies have been devoted to the evaluation of $H_2$, $C_3H_8$, $CH_4$, and RP-1 as coolants in the existing SSME cooling circuit geometry. All of these candidate coolants are feasible without limiting the desired operating conditions with the exception of RP-1, which would limit the maximum $P_c$ to 2000 psia. However RP-1 could be used to cool the nozzle only without imposing the chamber pressure limit. Fifteen candidate engine system cycles have been selected and a preliminary engine system balance has been conducted for 12 of these systems to establish component operating flowrates, pressures and temperatures. In general it was found that the staged combustion cycles employing fuel rich LOX/hydrocarbon turbine drive gases are power limited. It is necessary for these systems to operate at turbine inlet temperatures above 2000R or at reduced chamber pressure to achieve a power cycle balance.

## TASK I - PERFORMANCE DETERMINATION

In this task propellant performance data, combustion gas thermodynamic properties and turbine drive gas parameters are generated as required to support the other tasks. Main chamber performance predictions for the various mode 1 propellant combinations were predicted based on the JANNAF ODE program which yields theoretical $I_s$ and c*. A table of the predicted theoretical performance for sea level and vacuum operation is presented in Table 1 for the four propellant combinations of interest. The mixture ratio was selected based on peak $I_s$. Fuel densities are also presented as an

Table 1 . Fuel De[illegible] and Theoretical
Performan[illegible]

| | MAIN CHAMBER MIXTURE RATIO | FUEL $\rho$ LB/FT$^3$ | $I_s$ S.L. | $I_s$ VAC |
|---|---|---|---|---|
| $O_2/H_2$ | 6.0 | 4.4 | 410 | 450.4 |
| $O_2$/RP-1 | 2.8 | 50.0 | 328 | 358 |
| $O_2/CH_4$ | 3.5 | 26.4 | 335.3 | 368 |
| $O_2/C_3H_8$ | 3.0 | 36.6 | 331. | 363 |

$P_c$ = 3000 $\epsilon$ = 35:1

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indicator of relative system volumes. Delivered performance is calculated based on assumed efficiencies of $\eta_{c*}$ = .98 and $\eta_{CF}$ = .9859 applied to the theoretical performance for the LOX/hydrocarbon combinations and $\eta_{c*}$ =.9975 and $\eta_{CP}$ = .9859 for LOX/$H_2$. In those cases where some hydrogen is injected into the main chamber along with the LOX/hydrocarbon, a mass averaged performance calculation was assumed. A limited number of theoretical tripropellant performance runs have been made which verify the validity of this assumption. Furthe tripropellant performance predictions will be made for additional cases where necessary. 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 2.

Table 2. Gas Generator Cycle Secondary Flow Specific Impulse

| PROPELLANTS | $I_{s_{SL}}$ | $I_{s_{VAC}}$ |
|---|---|---|
| $O_2/H_2$ | 248.2 | 282.7 |
| $O_2/C_3H_8$ | 122.2 | 142.7 |
| $O_2/CH_4$ | 121.5 | 141.6 |

Turbine drive gas characteristics for oxidizer rich and fuel rich conditions are presented in Table 3. The function of $\gamma$ and pressure ratio is specifically $1 - (\frac{1}{PR})^{\gamma-1/\gamma}$ and is a measure of the energy available as the gas expands in the turbine. High values of $C_P$ and $\mathcal{f}(\gamma, PR)$ tend to minimize turbine flow for a given horsepower requirement.

TASK II - CHAMBER COOLING STUDIES

This task effort will be concerned with providing the heat transfer and cooling analysis support for the selected engine systems to be studied.

Table 3 . Turbine Drive Gas Characteristics

$T = 2000R$

$PR = 1.6$

| | LOX RICH | | | | FUEL RICH | | | |
|---|---|---|---|---|---|---|---|---|
| | MR | $\gamma$ | $C_p$ | $f(\gamma, PR)$ | MR | $\gamma$ | $C_p$ | $f(\gamma, PR)$ |
| $O_2/H_2$ | 86 | 1.293 | .296 | .1 | 1.14 | 1.345 | 1.78 | .113 |
| $O_2$/RP-1 | 33 | 1.31 | .294 | .105 | .41 | 1.135 | .66 | .054 |
| $O_2/CH_4$ | 37.5 | 1.289 | .284 | .1 | .43 | 1.16 | .875 | .0627 |
| $O_2/C_3H_8$ | 34 | 1.288 | .283 | .099 | .44 | 1.147 | .691 | .0584 |

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 will evaluate LOX cooling, $H_2$ cooling of an extendible nozzle and any cooling system variations that might offer some improvement to the engine systems being studied.

## $H_2$ Coolant with LOX/Hydrocarbon Combustion

The current SSME chamber and 35:1 development nozzle design were analyzed to determine the coolant bulk temperature rise, $\Delta P$, 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 up-pass circuit where the nozzle and chamber are cooled in parallel. The hydrogen is then injected into the chamber and burned with the $O_2$ and RP-1. The second circuit is a downpass series circuit with the hydrogen dumped at the nozzle exit. 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

$$P_o = 3237 \text{ psia}$$
$$T_o = 6512 \text{ F}$$
$$\dot{\omega}_g = 1455 \text{ lb}_m/\text{sec}$$
$$MR = 2.8$$
$$MW = 24.484 \text{ lb}_m/\text{lb}_m\text{-mole}$$

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 equation following

$$h_g\Big)_2 = h_g\Big)_1 \frac{k_2}{k_1} \left[ \frac{\omega_g)_2}{\omega_g)_1} \quad \frac{\mu_1}{\mu_2} \right]^{0.8} \left[ \frac{Pr_2}{Pr_1} \right]^{0.4}$$

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 percent 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. 2. The analogous profiles for the 35:1 nozzle are shown in Fig. 3. The effect of burning the hydrogen coolant with the $O_2$ and RP-1, in the chamber has not been considered in determining these profiles.

The possibility of a carbon layer build up 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 on the coolant requirement is shown 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.

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

Coolant bulk temperature rise and pressure drop are calculated using "REGEN" computer program models that have previously been set up for SSME analyses. The pressure drops computed using these models are increased by 10 percent 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 timesharing computer system.

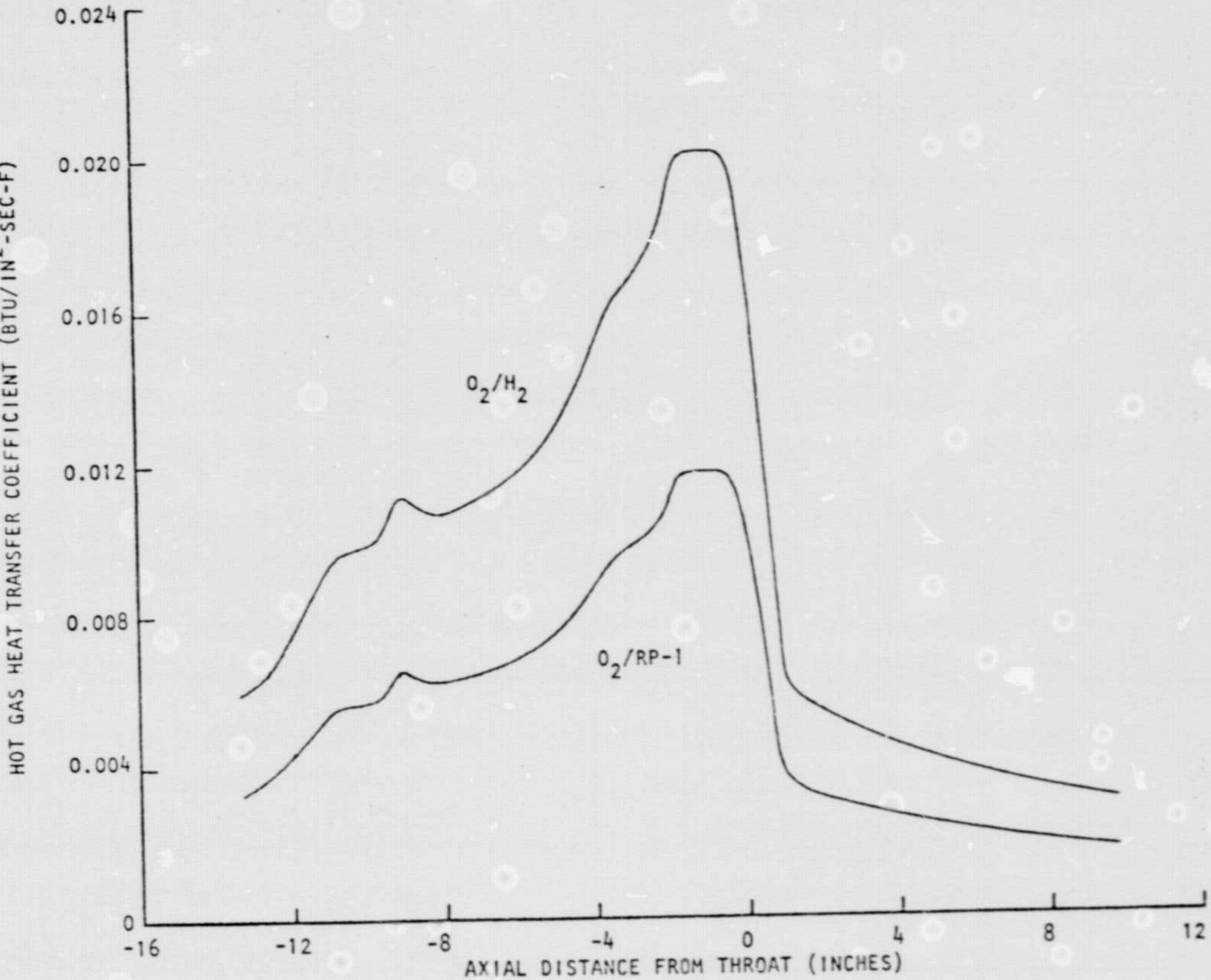


Figure 2. Main Combustion Chamber Heat Transfer Coefficient Profiles for $O_2/H_2$ and $O_2$/RP-1 Propellant Combinations ($P_c$= 3237 psia)

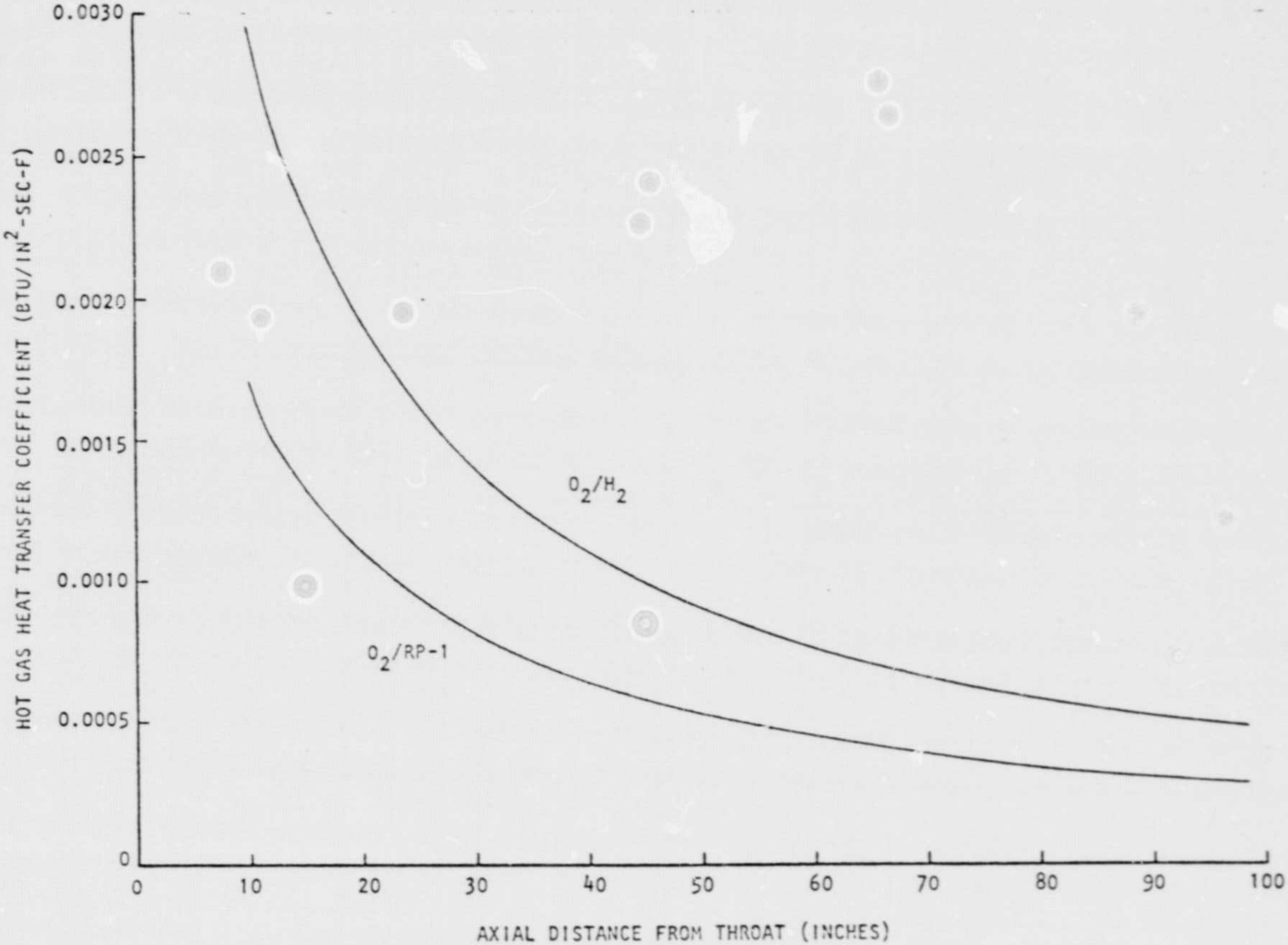


Figure 3. 35:1 Nozzle Heat Transfer Coefficient Profiles for $O_2/H_2$ and $O_2$/RP-1 Propellant Combinations ($P_c$ = 3237 psia)

For the up-pass circuit with the chamber and nozzle cooled in parallel the hot gas wall temperature as a function of flowrate is shown in Fig. 4a. Assuming no carbon coating, the nozzle would require a coolant flowrate of 18.5 $lb_m$/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 $lb_m$/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 $lb_m$/sec. When assuming a carbon coating, the temperature limiting location is in the throat (-1") and requires a coolant flowrate of 10.7 $lb_m$/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. 4b. The coolant pressure drop is given in Fig. 4c. 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 ($\sim$ 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. 5a. 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 $lb_m$/sec is required to maintain the same

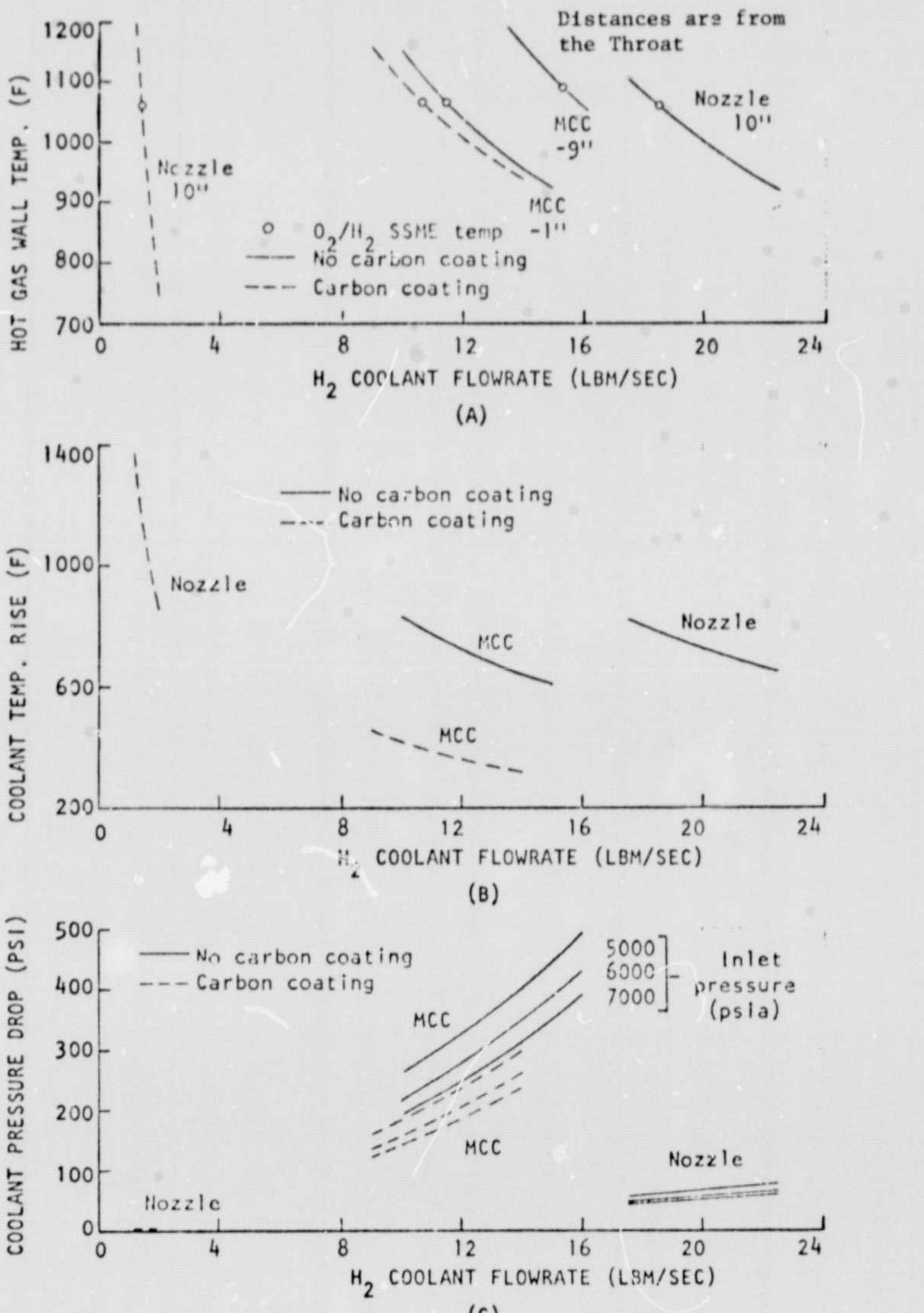


Figure 4. Hot Gas Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop as Functions of Coolant Flowrate for Up-pass Parallel Coolant Circuits for the SSME Main Combustion Chamber and 35:1 Nozzle

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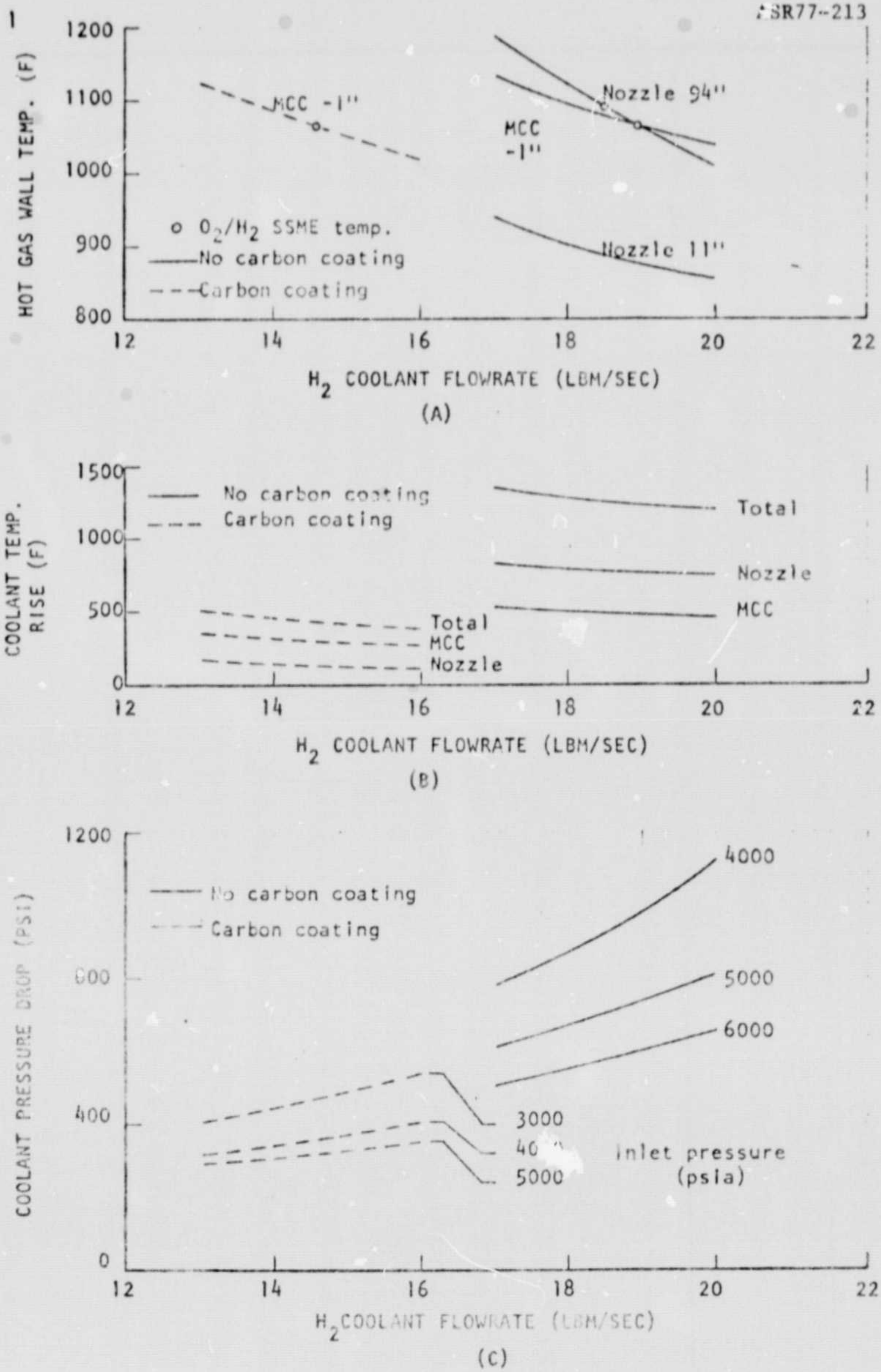


Figure 5. Hot Gas Temperature, Coolant Temperature Rise, and Coolant Pressure Drop as Functions of Coolant Flowrate for Downpass Series Coolant Circuits for the SSME Main Combustion Chamber and 35:1 Nozzle

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 $lb_m$/sec.

The coolant bulk temperature rise is shown in Fig. 5b. The coolant pressure drop (chamber plus nozzle) is shown in Fig. 5c, for three inlet pressures. When assuming no carbon coating, the inlet pressure should be maintained to at least near 4000 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.

| | | Nozzle $\dot{\omega}_c$ | Chamber $\dot{\omega}_c$ | Total | Fraction of $O_2/H_2$ SSME $\dot{\omega}_c$ |
|---|---|---|---|---|---|
| Up-Pass 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 |

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The up-pass parallel cool[illegible] 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. The small differences in the hot gas properties for the three hydrocarbon fuels would have an insignificant impact on the coolant parameters. Additional analysis was conducted for a 4000 psia chamber pressure case to be used with the gas generator cycle engine candidates.

## Hydrocarbon Fuel Cooling with LOX/Hydrocarbon

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 600F. 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. 6 through 9 , for the 3230 psia chamber pressure conditions. This analysis has also been conducted for a 4000 psia chamber pressure. These results are currently being evaluated and incorporated into the candidate engine system balances.

## TASK III CYCLE AND POWER BALANCE

The objectives of this task are to define the cycles and perform cycle power balances to determine the required component flow rates, turbine inlet temperatures and pump discharge pressures based on the pressure losses of the various components. Power cycles to be examined include staged combustion and gas generator.

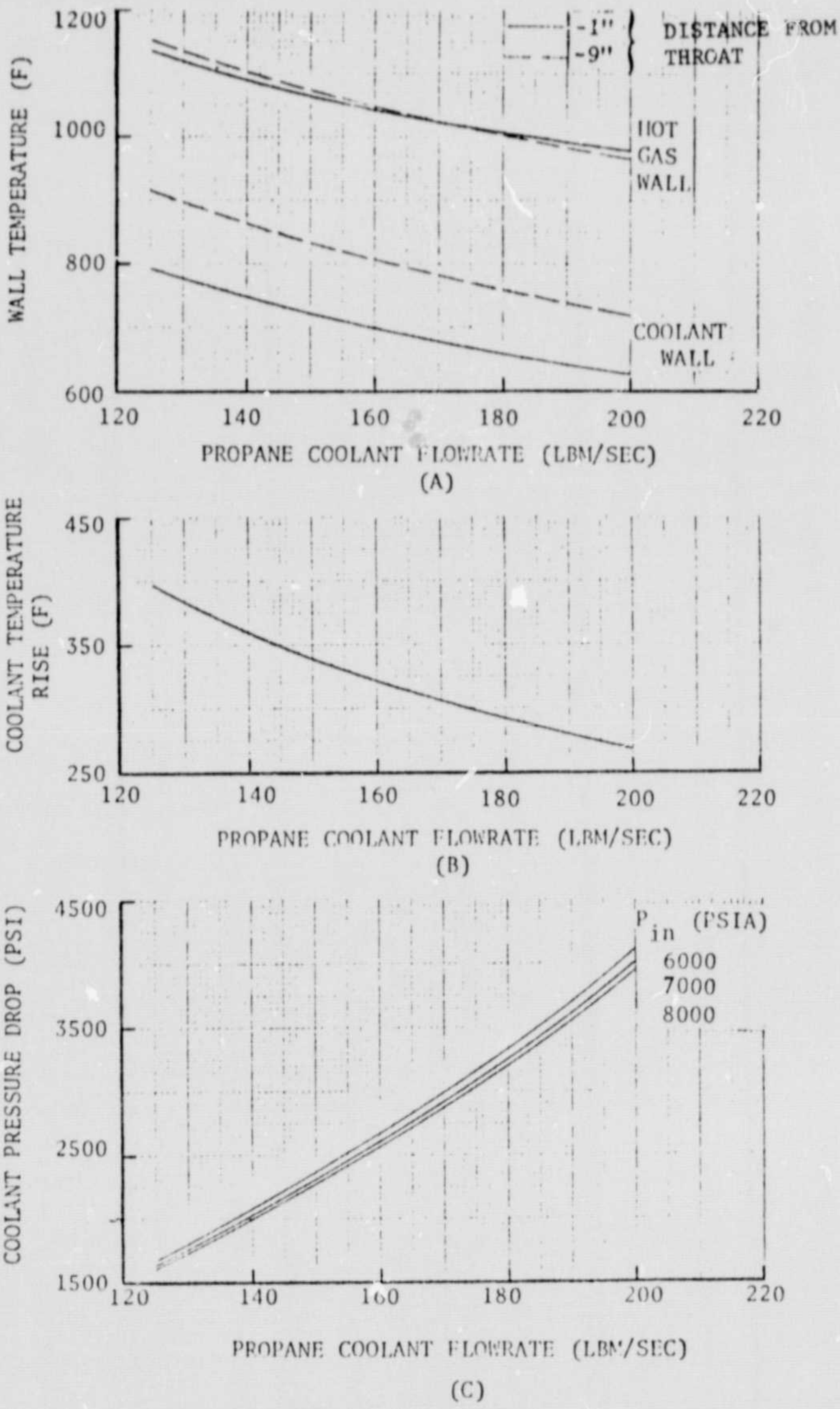


FIGURE 6. WALL TEMPERATURE, COOLANT TEMPERATURE RISE, AND COOLANT PRESSURE DROP FOR AN $O_2$/PROPANE COMBUSTOR ($P_c$ = 3230 PSIA) BYPASS PROPANE COOLED

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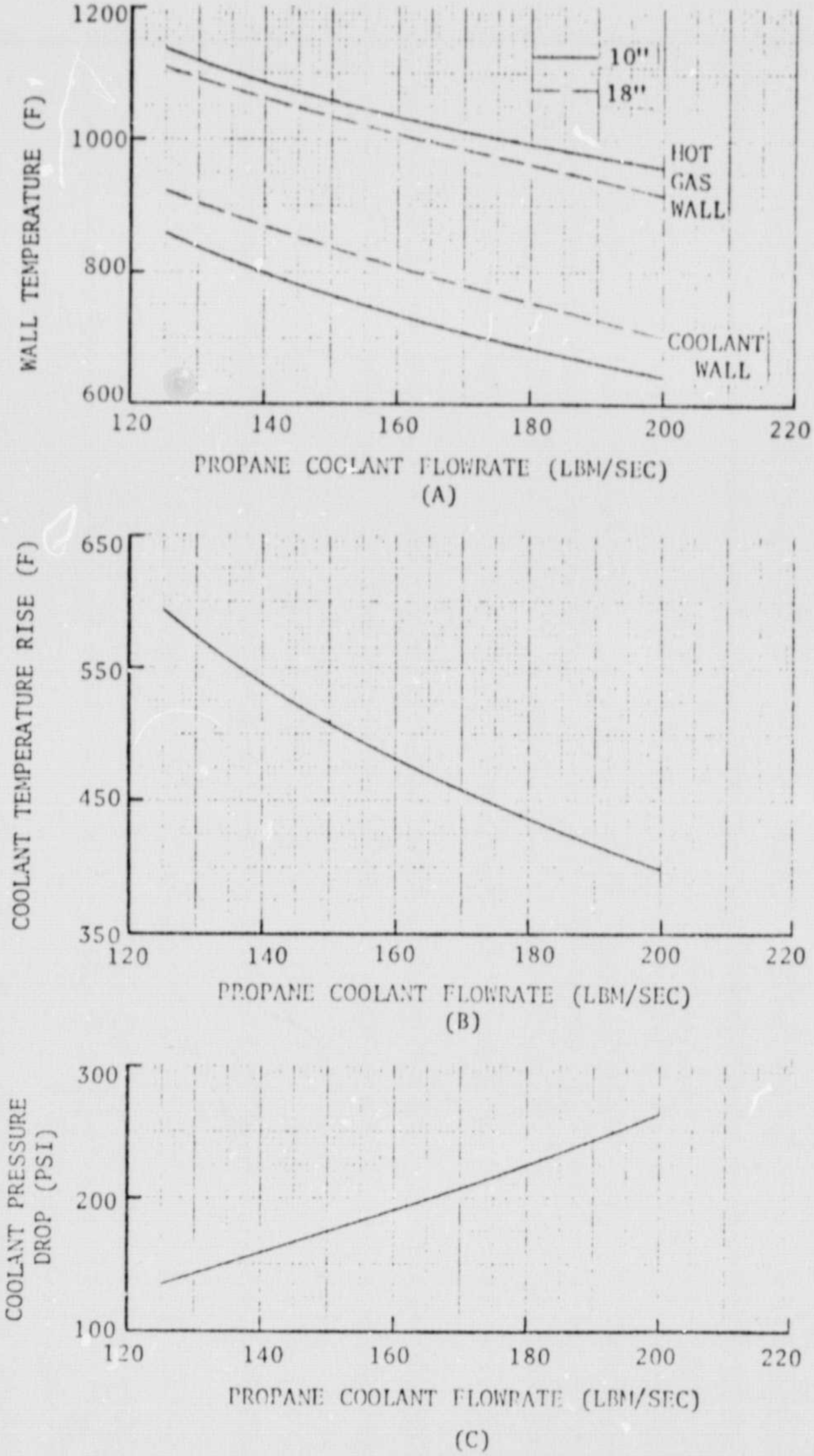


FIGURE 7. WALL TEMPERATURE, COOLANT TEMPERATURE RISE, AND COOLANT PRESSURE DROP FOR AN $O_2$/PROPANE 35:1 NOZZLE ($P_c$ = 3230 PSIA) UPPASS PROPANE COOLED

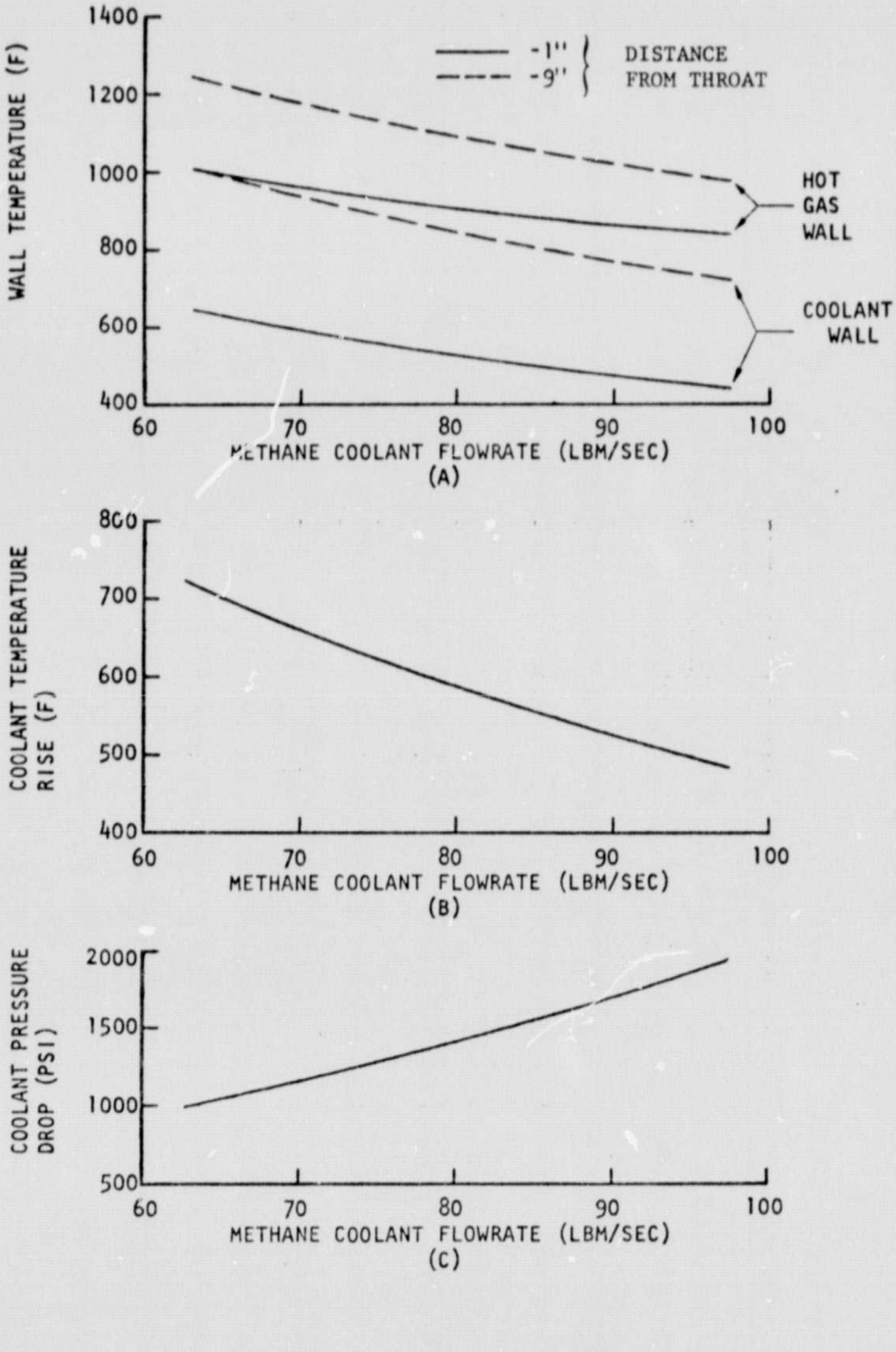


Figure 8. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a $P_c$=3230 PSIA $O_2/CH_4$ $CH_4$ Cooled Up-pass Chamber



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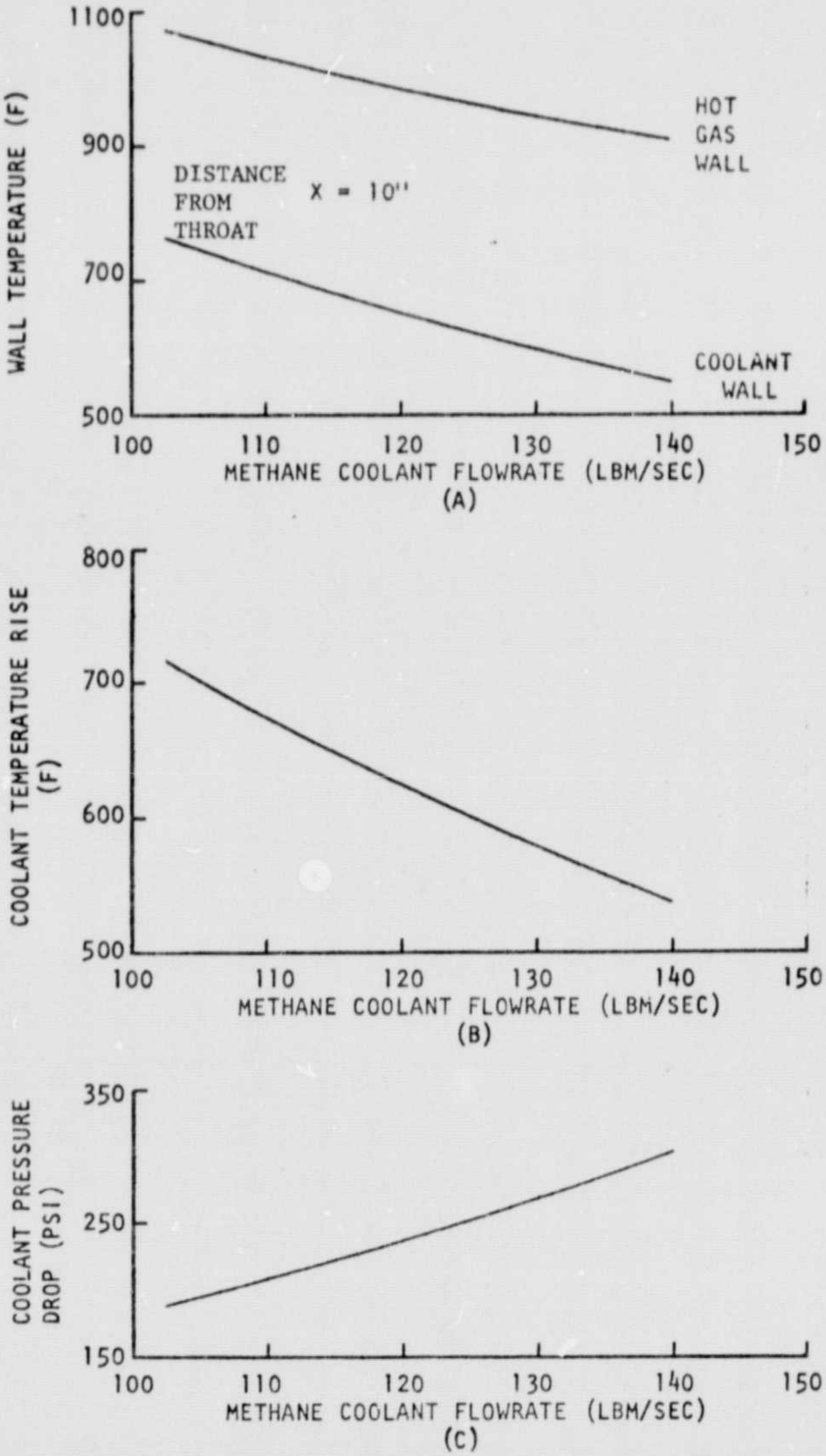


Figure 9. Wall Temperature, Coolant Temperature Rise, and Coolant Pressure Drop for a $P_c$=3230 psia $O_2/CH_4$ $CH_4$ Cooled Up-pass 35:1 Nozzle

Previous tripropellant engine application studies have shown that there is an interest in operating these engines in either a series burn or parallel burn configuration as illustrated in Fig. 10 . 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 tripropellant engine used in the parallel burn configuration does not require the sequential burning of two fuels but may use the second fuel 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.

The approach taken in this study was to establish some ground rules for the single mode and dual mode engines. Based on these ground rules (Fig. 11 and 12 ) a set of candidate engine cycle and propellant/coolant combinations were established for preliminary analysis. This initial set of fifteen candidate engine systems is presented in Fig. 13 . Actually a large number of system combinations could be considered but this would result in a list of candidate cycles far too large to be studied within the limits of the funds. The selected fifteen systems were considered to be representative of the power cycles, propellant combinations and cooling techniques of interest. It was also anticipated that if through the study of these systems certain features were found to be particularly advantageous, this may suggest an additional system concept to be considered. One other general ground rule that was used in establishing this list of candidate systems, was to minimize the use of $H_2$ in the mode 1 operation.

Engine balances were conducted for 12 of the 15. The engine power balance for engine system concepts No. 10, 11 and 14 were delayed awaiting heat transfer information. The major component flowrates, turbine inlet temperature and pump discharge pressure requirements that were established in this analysis are presented in Table 4. In general, it was found that turbine inlet

# TRIPROPELLANT ENGINE OPERATING MODES

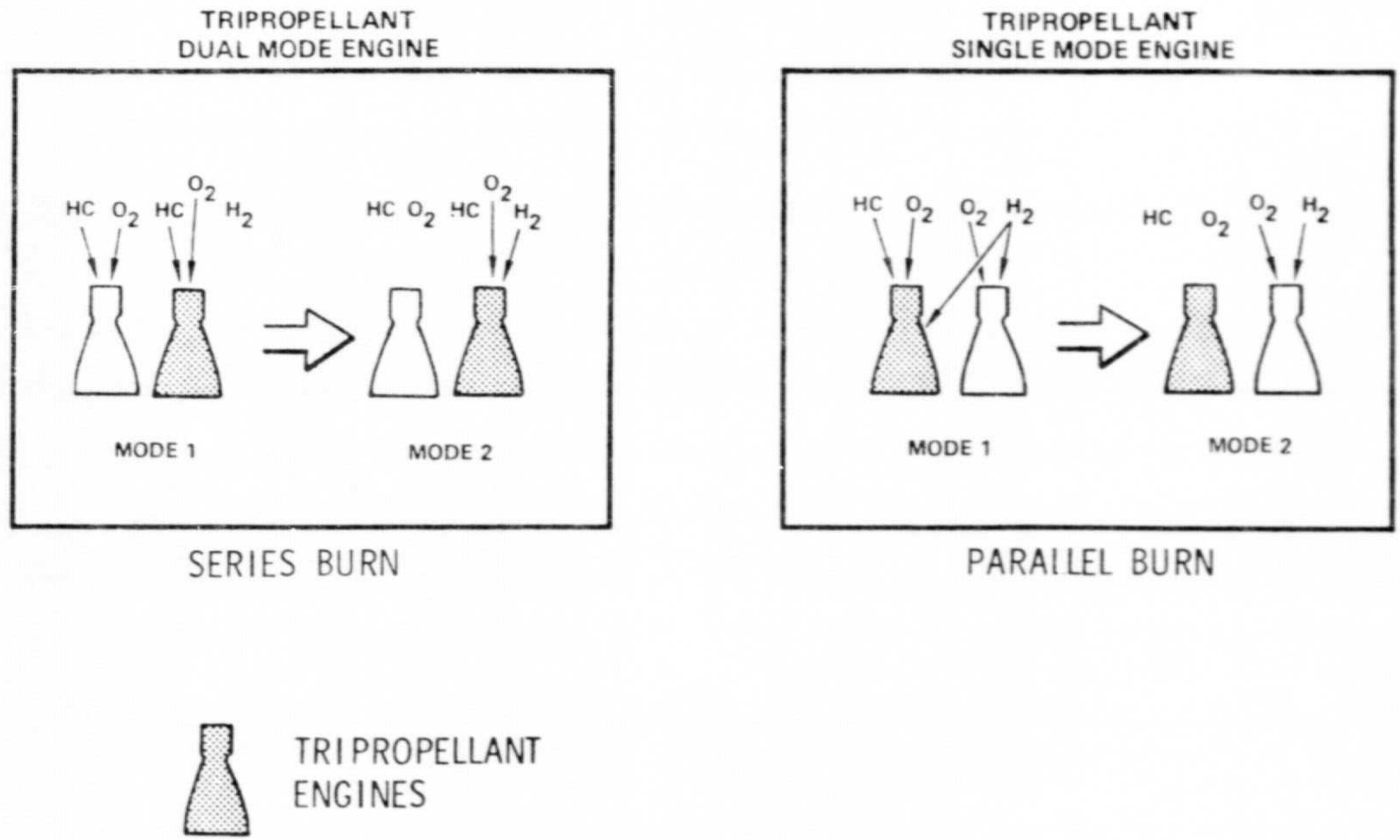


Figure 10

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# GROUND RULES FOR CANDIDATE SINGLE MODE CYCLE SELECTION

- POWER CYCLE TYPES

  STAGED COMBUSTION
  GAS GENERATOR

- PROPELLANTS

  LOX/RP-1, LOX/$CH_4$, LOX/$C_3H_8$

- $P_c$ = 3000 TO 4000 PSIA

  $\epsilon$ = 35.1

- COOLANTS

  $CH_4$, $C_3H_8$, $H_2$

- TURBINE INLET TEMP TO 2200°R

- ALL PREBURNERS EITHER FUEL OR LOX RICH

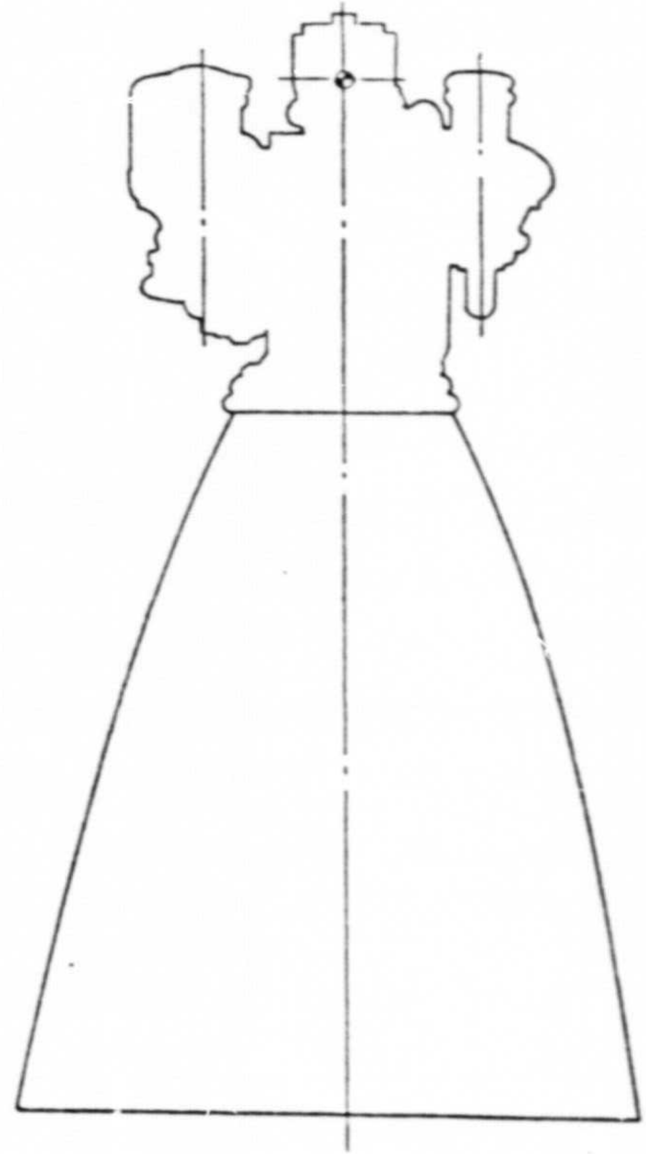

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Figure 11

# GROUND RULES FOR CANDIDATE DUAL MODE CYCLE SELECTION

- 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

  $\epsilon$ = 35:1 MODE 1

  = 150:1 MODE 2
- COOLANTS:
  - $LH_2$, $CH_4$, $C_3H_8$, $LO_2$
- TURBINE INLET TEMP TO 2200°R
- ALL PREBURNERS EITHER LOX OR FUEL RICH

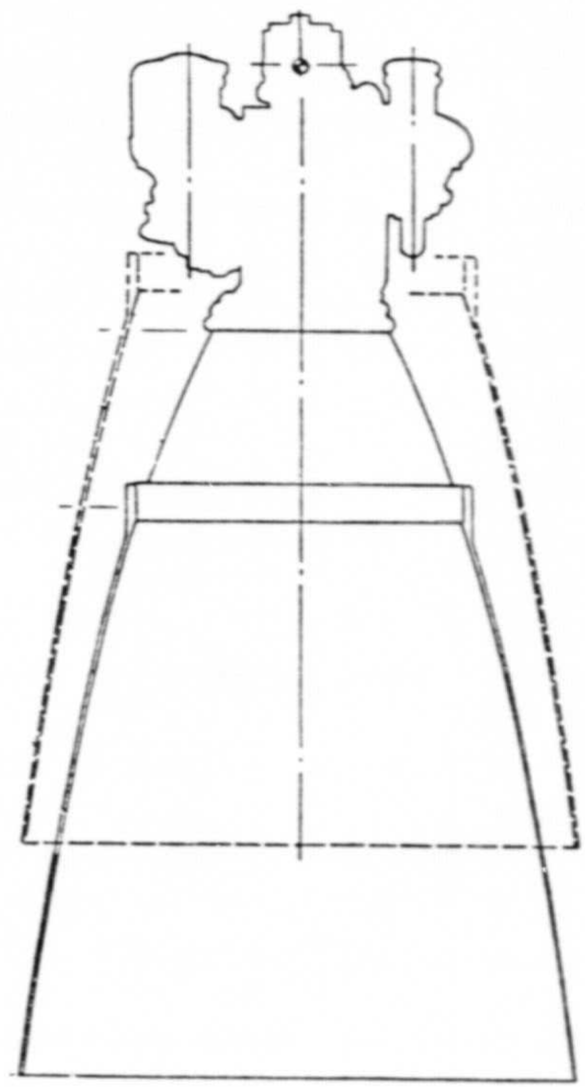

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Figure 12

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# CANDIDATE ENGINE CONCEPTS

| CONCEPT NO. | PROPELLANTS | CYCLE TYPE | COOLANT | |
|---|---|---|---|---|
| 1 | $LOX/RP\text{-}1/H_2$ | S.C. | $H_2$ | ALL PREBURNERS FUEL RICH LOX/RP-1 |
| 2 | $LOX/RP\text{-}1/H_2$ | S.C. | $H_2$ | ALL PREBURNERS OX RICH LOX/RP-1 |
| 3 | $LOX/CH_4/H_2$ | S.C. | $H_2$ | ALL PREBURNERS FUEL RICH $LOX/CH_4$ |
| 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. | $H_2$ | ALL PREBURNERS FUEL RICH $LOX/C_3H_8$ |
| 6 | $LOX/C_3H_8/H_2$ | S.C. | $H_2$ | ALL PREBURNERS OX RICH $LOX/C_3H_8$ |
| 7 | $LOX/RP\text{-}1/H_2$ | 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 | $LOX/RP\text{-}1/H_2$ | S.C. | $O_2$ | ALL PREBURNERS FUEL RICH LOX/RP-1 |
| 11 | $LOX/RP\text{-}1/H_2$ | 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 FUEL RICH $LOX/CH_4$ |
| 13 | $LOX/C_3H_8/H_2$ | S.C. | $C_3H_8/H_2$ | ALL PREBURNER FUEL RICH $LOX/C_3H_8$ |
| 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. $LOX/C_3H_8$ |

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Figure 13

S.G. - Staged Combustion
G.G. - Gas Generator

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Table 4. Candidate Engine Concepts

| CONCEPT NO. | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 13B |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| $P_C$ | 3230 | 3230 | 3230 | 3230 | 3230 | 3230 | 4000 | 4000 | 4000 | | | 3230 | 3230 | | 4000 | 3230 |
| S.L. THRUST | | 460K | 465K | 465K | | 464.4K | 470K | 470K | 470K | | | 470K | 470K | | 470K | 470K |
| VAC THRUST | | 500K | 508.6K | 508.6K | | 507.6K | 502K | 504K | 503.5K | | | 516.2K | | | 504K | 515K |
| PROPELLANTS | | $O_2$/RP-1 | $O_2/CH_4$ | $O_2/CH_4$ | | $O_2C_3H_8$ | $O_2$/RP-1/$H_2$ | $O_2/CH_4/H_2$ | $O_2/C_3H_8H_2$ | | | $O_2/CH_4$ | | | $O_2/C_3H_8$ | $O_2/C_3H_8$ |
| COOLANT | | $H_2$ | $H_2$ | $H_2$ | | $H_2$ | $H_2$ | $H_2$ | $H_2$ | | | $CH_4$ | | | $C_3H_8$ | $C_3H_8$ |
| TURBINE DRIVE FLUID | | $O_2$PR-1 | $O_2CH_4$ | $O_2/CH_4$ | | $O_2/C_3H_8$ | $O_2/H_2$ | $O_2/H_2$ | $O_2/H_2$ | | | $O_2/CH_4$ | | | $O_2/C_3H_8$ | $O_2/C_3H_8$ |
| M.R. | | 2.8 | 3.5 | 3.5 | | 3.0 | 2.8 | 3.5 | 3.0 | | | 3.5 | | | 3.[illegible] | 3.0 |
| $I_S$ S.L., SEC | | 333.8 | 339.2 | 339.2 | | 335.7 | 329.7 | 336.9 | 334 | | | 324 | | | 313.8 | 320 |
| $I_S$ VAC, SEC | | 362.5 | 371 | 371 | | 366.7 | 352.3 | 361.3 | 358 | | | 355 | | | 336.6 | 351 |
| $T_{TURBINE}$, R | ~2250 | 2000 | 2250 | 2000 | ~2400 | 2000 | 2000 | 2000 | 2000 | | | 2000 | 2375 | | 2000 | 2000 |
| PREBURNERS/GG, LB/SEC | | | | | | | | | | | | | | | | |
| $O_2$ | | 733.2 | 148.7 | 945.9 | | 887.8 | 13.9 | 15.8 | 14.7 | | | 659.2 | | | 33.6 | 776 |
| FUEL | | 22.2 | 252 | 25.2 | | 26.1 | - | - | - | | | 17.6 | | | 76.4 | 22.8 |
| $H_2$ | | - | - | - | | - | 17.4 | 19.7 | 18.3 | | | | | | - | - |
| $\dot{\omega}_{TURBINE}$, LB/SEC | | | | | | | | | | | | | | | | |
| $O_2$ | | 396.8 | 161.2 | 459.9 | | 459.4 | 18.5 | 18.9 | 18.5 | | | 378.6 | | | 62.2 | 337.6 |
| FUEL | | 187.3 | 158.9 | 334.8 | | 267.2 | 5.4 | 8.5 | 6.9 | | | 299 | | | 47.9 | 461.2 |
| $H_2$ | | 171.3 | 83.0 | 186.3 | | 187.2 | 7.5 | 7.5 | 7.5 | | | | | | - | - |
| $\dot{\omega}$ COOLANT, LB/SEC | | 34 | 34 | 34 | | 34 | 34 | 34 | 34 | | | 97.5<br>120 | | | 140COMB<br>140NOZ | 140COMB<br>140NOZ |
| $\dot{\omega}_{O_2}$ (TOTAL), LB/SEC | | 1045 | 1085 | 1085 | | 1063.5 | 1056 | 1084 | 1058 | | | 1128 | | | 1074.5 | 1101 |
| $\dot{\omega}_{FUEL}$ (TOTAL), LB/SEC | | 300.3 | 252 | 252 | | 286.5 | 335.7 | 277 | 314 | | | 322 | | | 423.3 | 367 |
| $\dot{\omega}_{H_2}$ (TOTAL), LB/SEC | | 34 | 34 | 34 | | 34 | 34 | 34 | 34 | | | | | | | |
| $\dot{\omega}_{TOTAL}$, LB/SEC | | 1379 | 1371 | 1371 | | 1384 | 1425 | 1395 | 1407 | | | 1450 | | | 1497 | 1468 |
| $\dot{\omega}_{H_2}$ T.C., LB/SEC | | 34 | 34 | 34 | | 34 | 17 | 14.3 | 15.7 | | | | | | - | - |
| PUMP DISCHARGE PRESSURE | | | | | | | | | | | | | | | | |
| LOX P.B.<br>CHAMBER | | 7331<br>4123 | 7331<br>4123 | 7331<br>4123 | | 7331<br>4123 | 5106 | 5106 | 5106 | | | 4123<br>7331 | | | 5106 | 4123<br>7331 |
| FUEL P.B.<br>CHAMBER | | 7331<br>4123 | 7331<br>4123 | 7331<br>4123 | | 7331<br>4123 | 5106 | 5106 | 5106 | | | 6064<br>7331 | | | 5107 | 6064<br>7331 |
| $H_2$ | | 4000 | 4000 | 4000 | | 4000 | 6084 | 6084 | 6084 | | | - | | | - | - |

pressures exceeding 2200R were required to achieve a power balance on the fuel rich precombustor systems (no. 1, 3, 5, and 12 and 13). This is a result of the lower energy available per pound turbine drive gas when comparing LOX/hydrocarbons with $LOX/H_2$. It is felt that these turbine inlet temperatures exceed the capability of the currently available hardware. Even ignoring the high turbine temperatures, the system is undesirable as no margin exists (all of the available fuel is used in the preburner) to compensate for the possibility of reduced turbopump performance. As a result only LOX-rich precombustors will be considered for those systems. The information shown for concept 12 and 13B reflect that change. Engine schematic improvements are being studied along with the Present Component Adaptability studies that have been initiated. These schematics will be finalized during the next report period.

## TASK IV CONTROL REQUIREMENTS DEFINITION

During this task, control methods, their requirements, and the location of control valves in the flow schematic will be studied for the candidate engine system concepts. No work was scheduled or conducted on this task during this report period.

## TASK V SSME COMPONENT ADAPTABILITY

This task is organized to interact with the efforts of Tasks 1 through 4 to evaluate the possibility of adapting the already designed and proven SSME components to the candidate systems. This effort has just been initiated in the areas of combustion chambers and pumps. There are no results to report this period.

TASK VI COMPONENT TEST PLANS

The results of Tasks I through V will be utilized to identify critical areas for experimental component evaluation. Based on this information test plans will be generated for additional testing to compliment the current NASA test plans for 40K hardware with LOX/RP-1. This effort is planned to start in mid-January 1978.

CURRENT PROBLEMS

There were no technical problems during this report period.

WORK PLANNED

Study effort in Tasks I, II and III should be nearing completion by the end of the next report period. A review for NASA on the completed analysis is planned shortly after. The Task I effort will continue on a low level to provide main chamber performance data and turbine drive gas characteristics as needed to support the cycle analysis. Cooling studies will be continuing in the areas of $O_2$ cooling of the nozzle and chamber and $H_2$ cooling of an extendible nozzle for mode 2 operation. In Task III, the preliminary power balances will be completed and the flow schematics will be finalized. Any cycle improvements that are apparent will be incorporated. SSME component adaptability studies will be conducted for combustion devices, turbines and pumps.

TRIPS AND MEETINGS

A study kick-off meeting was held at the Marshall Space Flight Center on September 20, 1977 with Dale Blount, the contract monitor and other concerned NASA personnel. Rocketdyne presented their intended approach to the study and obtained general agreement. There was a NASA request that we concentrate on the series burn, dual mode tripropellant engine concepts and provide information on the single mode engines only as its available from the main effort of the study.

## PROGRAM PERFORMANCE AND EXPENDITURES

Program performance and expenditures as of 30 September 1977 are as shown in Table .

Table 5 . Program Expenditures, Tripropellant Engine Study

Month Ending September 1977
NAS8-32613
(GO 9886)

| Total Cumulative Cost to Date | ETC | EAC | Percent Complete |
|---|---|---|---|
| 8.8K | 33.1K | 41.9K | 21 |

## SYMBOL NOMENCLATURE

| | |
|---|---|
| $P_O$ | Combustion Pressure |
| $T_O$ | Combustion Gas Temperature |
| $\dot{W}_g$ | Combustion Gas Flowrate |
| MR | Combustion Gas Mixture Ratio Oxidizer-to-Fuel |
| MW | Molecular Weight of Combustion Gases |
| hg | Hot Gas Heat Transfer Coefficient |
| K | Gas Thermal Conductivity |
| $\mu$ | Gas Viscosity |
| Pr | Prandtl Number |
| X | Distance |
| G | Mass Velocity |

# REFERENCES

1. R-3909 NAS8-4011, Investigation of Cooling Problems at High Chamber Pressures, May 1963

2. Wagner, W. R. and J. M. Shoji, Advanded Regenerative Cooling Techniques for Future Space Transportation Systems, AIAA Paper No. 75-1247

3. Luscher, W. P. and J. A. Mellish, Advanced High-Pressure Engine Study for Mixed Mode Application, Final Report NASA CR-135141, January 1977.