NASA CR-167714

:. W.C. Boyd EP22

Combustion Performance And Heat Transfer Characterization Of LOX/Hydrocarbon Type Propellants

Contract NAS 9-15958
Task III Data Dump
July 1982

Prepared For:
National Aeronautics And Space Administration
Lyndon B. Johnson Space Center
Houston, Texas 77058

By: S. W. Hart

(NASA-CR-167714) COMBUSTION PERFORMANCE AND HEAT TRANSFER CHARACTERIZATION OF LOX/HYDROCARBON TYPE PROPELLANTS. TASK 3: DATA DUMP Progress Report (Aerojet Liquid Rocket Co.) 202 p HC A10/MF A01 CSCL 21I G3/28

N82-30412

Unclas
28747

Aerojet Liquid Rocket Company

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TASK III DATA DUMP

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BY:

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I. INTRODUCTION

This document presents the results of the Task III effort of Contract NAS 9-15958. The general objectives of NAS 9-15958 are to evaluate combustion performance and heat transfer characteristics of LOX/hydrocarbon propellants and thus establish a data base for propellant combustion screening of future advanced Space Transportation System (STS) engines, specifically the Orbital Maneuvering System (OMS) engines and Reaction Control System (RCS) engines.

Tasks I and II of this contract included chamber cooling analyses, heated tube tests of propane, and subscale thrust chamber hot fire tests utilizing LOX/hydrocarbon propellants. Task III was a preliminary characterization of OMS and RCS engine point designs over a range of thrust and chamber pressure for several hydrocarbon fuels. It is this effort which is reported herein.

The Task III results are input to another program, LOX/Hydrocarbon Auxiliary Propulsion System Study (NAS 9-16305), being conducted by McDonnell Douglas Astronautics Company, St. Louis Division. This contract will characterize the entire STS pod system.

ALRC has been under contract (P.O. Y1E015) to provide additional OME and RCE data in support of NAS 9-16305. The LOX/C_2H_4OH (ethyl alcohol) engine data presented herein were generated on the MDAC-STL contract. The data are included to provide comparisons to the propellant combinations evaluated on NAS 9-15958, namely LOX/CH_4, LOX/C_3H_8, and LOX/NH_3.

Task III was conducted in two phases in which OMS and RCS engine point designs were established. This included definition of: interface pressures, performance and operating parameters, combustion chamber cooling and turbopump requirements, component weights and envelopes, and propellant conditioning requirements for liquid-to-vapor phase engine operation. "Baseline" engine point designs were evaluated in the initial phase and "parametric" engine point designs were evaluated in the second phase. A total of 22 OMS and 20 RCS engine point designs were characterized during the Task III effort.

Appendix I and II contain the Baseline and Parametric phase data sheets respectively. The results of an additional 7 OME point designs over and above the original work scope are tabulated in Appendix III. Appendix IV is a listing of the groundrules and evaluation criteria used in this study.

Appendices V and VI contain ALRC internal reports of chamber thermal analyses and engine performance analyses.

II. SUMMARY

A. ENGINE CONCEPT DEFINITION

The engine point designs that were analyzed on this contract are defined in Table I for the OMS engines and Table II for the RCS engines. Both Tables include the added scope OME point designs and the LOX/C_3H_5OH point designs evaluated for MDAC-STL. These tables define: propellant feed mode, vacuum thrust, chamber pressure, and OME chamber materials. Note that all OME chambers are regeneratively cooled.

The OMS engine design points include pressure-fed and pump-fed concepts and the four LOX/hydrocarbon propellant combinations noted above. The off-nominal MR and Pc operating ranges are: Pressure-Fed: $\Delta MR = \pm 20\%$, $\Delta Pc = \pm 25\%$; Pump-Fed: $\Delta MR = \pm 5\%$, $\Delta Pc = \pm 10\%$. The engines are designed at the most severe operating point within this MR - Pc box. Table III describes the major OME component designs used in the study.

The RCS engines (870 and 25-lbf) are adiabatic wall (coated Columbium) chamber concepts which require film cooling in all cases. The off-nominal MR and Pc operating ranges are $\pm$ 40% for both parameters, with design requirements again set at the worst point of the operating box. The same 4 propellant combinations were included for the RCE point designs.

Nozzle expansion ratios were based on the current OME and RCE packaging envelopes.

Most of the OME design points had NBP propellant temperatures at the engine interface except for C_2H_5OH which was 60°F. Ambient temperature C_3H_8 was evaluated for a pressure-fed OMS engine point design. RCE point designs were evaluated for both liquid and vapor at the thruster valve inlet.

OME chamber materials evaluated were zirconium-copper (Zr-Cu), nickel (Ni), and 300 series stainless (CRES). All LOX/NH_3 chambers were CRES for compatibility reasons. Ni and CRES chambers were evaluated for LOX/C_3H_8 and LOX/CH_4 in addition to Zr-Cu, after it was learned that a potential incompatibility exists with Zr-Cu and C_3H_8.

Other OME concept variations characterized were: boost pumps, LOX regeneratively cooled chamber (Zr-Cu), expander cycle (LOX/CH_4), and Cg (gas-side heat transfer coefficient)profiles.

Further engine concept definition is shown on Figures 1 and 2 which are flow schematics indicating arrangement of major components. Figure 1 depicts the OMS engine schematics. The two pump-fed engine cycles are simplified in that they do not show redundant turbopump (TPA) and gas generator (GGA) assemblies nor accumulators required for engine start.

The RCS engine schematics, Figure 2, only show one thruster for simplicity, however, there would be approximately the same number as on the current Shuttle Orbiter (44). Most of the RCE's are 870-lbf with 6 being 25-lbf units.

In order to achieve the advantages of commonality, the TPA's and GGA's for the RCS engines are considered to be the same as for the OMS engines. Because the total flow rate during RCE operation is much smaller than the TPA/GGA design flow rates, propellant accumulators are included in these concepts, permitting RCE operation without the TPA/GGA being operated. The RCS engine concepts which utilize vapor phase propellant incorporate a propellant conditioning system to convert the propellant liquid into a vapor. The propellant conditioning hardware consists of a fuel-rich gas generator (Tc = 2000°R) and a concentric tube/ shell heat exchanger.

B. ENGINE COMPARISONS

Key results are summarized on Tables IV and V for the OMS engine point designs and Tables VI and VII for the RCS engine point designs. These key results include: performance and specifically vacuum specific impulse (Isp), engine inlet pressures, and engine weight.

Workable designs at all OME and RCE operating points were achieved, except for the LOX/CH_4 expander cycle. With a conventional regeneratively cooled chamber and nozzle sufficient energy could not be extracted from the fuel coolant to power-balance the turbopumps in an expander cycle. At an operating point of 800 psia chamber presssure and 10K lbf thrust, only 80% of the energy required to operate the turbopumps could be transferred to the fuel coolant, cooling the entire combustion chamber and divergent nozzle. Unconventional chamber designs could increase the heat transferred to the fuel. Also, a point design at lower Pc and thrust would improve the power balance of this engine concept.

1. OMS Engines

a. Performance

The major trends in vacuum specific impulse are as follows:

- LOX/CH_4 is highest performing propellant combination of the four considered, based on vacuum specific impulse.

- The highest Pc and lowest vacuum thrust combination yields the highest vacuum specific impulse.

These results are not surprising since LOX/CH_4 has the highest theoretical vacuum Isp of the four propellant combinations. The low thrust, high Pc design point results in the largest nozzle expansion ratio within the constraints of fixed engine envelope. The study criteria precluded modification of the existing engine envelopes.

The maximum vacuum Isp point design was for LOX/C_3H_8 at Pc = 800 psia, F_V = 6K lbf and the Isp was 368.7 seconds. A similar point design for LOX/CH_4 was not evaluated but its estimated vacuum Isp is 370.5 seconds. These performance levels represent the best that could be attained because the engine point designs required no film cooling which would degrade the vacuum Isp.

The other two propellant combinations yielded point designs with significantly lower performance. At Pc = 800 psia and F_V = 10K operating points the vacuum Isp are shown below:

OME PUMP-FED
Performance at Pc = 800 psia/Fv = 10K lbf

Propellant	Engine Vacuum Isp, sec
LOX/CH_4	366.0
LOX/C_3H_8	363.8
LOX/NH_3	317.4
LOX/C_2H_5OH	344.1

The LOX/NH_3 point design performance was degraded from its potential because 33% fuel film cooling was necessary.

A comparison of pressure-fed point designs also verifies the higher performance of LOX/CH_4. The table below shows the vacuum Isp for representative point designs:

OME PRESSURE-FED

Performance at Pc = 100 psia/Fv = 6K lbf

Propellant	TCA Vacuum Isp, sec
LOX/CH_4	343.2
LOX/C_3H_8	337.0
LOX/NH_3	318.8
LOX/C_2H_5OH	319.9

All of these point designs required no fuel film cooling with exception of LOX/NH_3 which required 11%.

A comparison of the above tabulated data shows that a performance increase (23-27 seconds) can be achieved with the pump-fed point designs. This performance difference results because the higher Pc of the pump-fed designs (800 psia) increases the nozzle expansion ratio and the theoretical performance. LOX/NH_3 is an exception where the additional chamber cooling losses at the higher Pc result in no performance improvement.

For a given engine point design, the key in achieving maximum performance is being able to cool the combustion chamber without any fuel film cooling. Initial pressure-fed point designs at Pc = 100 psia indicated LOX/C_3H_8 required 30% fuel film cooling and the LOX/CH_4 point design could not be cooled at all. These results were based on using liquid state fuel (NBP temperature) for regenerative cooling of the chamber. The difficulty in cooling with liquid CH_4 and C_3H_8 is that the burnout heat flux is quickly approached in the coolant jacket. This limiting criteria can be eliminated with vapor regenerative cooling which was subsequently evaluated. The results showed that satisfactory chamber cooling could be achieved without any additional fuel film cooling. Because a significant source of energy is required to vaporize the fuel somewhere between the propellant tank and engine, some added complexity or components such as heat exchangers would be necessary. The most practical approach would be to use the combustion chamber and nozzle as the heat source to provide the change of state. Preliminary analysis shows that CH_4 would be more readily vaporized than C_3H_8 and this could be accomplished in the divergent nozzle section of the engine.

C_2H_5OH was a satisfactory liquid regenerative coolant because of its much higher critical temperature resulting in a higher limit for the burnout heat flux. NH_3 required 11% fuel film cooling at Pc = 100 psia based on liquid state cooling. Vapor cooling with NH_3 was not evaluated but is not considered competitive because the energy required to vaporiz NH_3 would be much greater than for CH_4 or C_3H_8.

At the higher chamber pressure levels of pump-fed operation the need for fuel film cooling was minimized by regenerative cooling with supercritical pressure fuel (CH_4 and C_3H_8 only), thereby eliminating the burnout heat flux consideration. Obviously, this required a higher pump discharge pressure for the fuel. However, the impact of discharge pressure on engine vacuum specific impulse is relatively small: ΔIsp degradation is 0.1 - 0.2 seconds for 100 psi increase in discharge pressure.

As a result, the pump-fed point designs for LOX/CH_4, C_3H_8 and C_2H_5OH required no fuel film cooling with exception of LOX/C_3H_8 in combination with nickel and CRES chamber materials, which required 3% and 25% fuel film cooling, respectively. All pump-fed LOX/NH_3 point designs required fuel film cooling.

Another variable in determining fuel film coolant requirements is the chamber material. Three chamber materials were evaluated, Zr-Cu, Ni and CRES. Zr-Cu is the best material because of its high thermal conductivity, and CRES is the poorest. Because of chamical incompatibility of NH_3 and copper all of the LOX/NH_3 point designs were evaluated with CRES. This accounts for NH_3 being a poor regenerative coolant. Nickel chambers, which were introduced into the study midway, should be evaluated with LOX/NH_3 propellants because the fuel film cooling requirements would be reduced and the vacuum specific impulse increased for these point designs. These design points still would not approach the performance of LOX/CH_4 and LOX/C_3H_8, however.

Other engine variables evaluated were boost pumps and LOX regenerative cooling. Both of these concepts required higher gas generator/turbine flow rates, but resulted in negligible performance reductions.

b. TCA Inlet Pressures

The TCA inlet pressures are those at the thrust chamber valve inlet which is upstream of the engine valves or chamber coolant jacket inlet. These pressure levels represent the tank pressures for pressure-fed concepts and the pump discharge pressures for pump-fed concepts.

For the most part the inlet pressures are similar, especially for the LOX because fuel is used for regenerative cooling. The fuel inlet pressures do vary somewhat because of the coolant jacket ΔP. The minimum fuel inlet pressures for pressure-fed engines are for the vapor regeneratively cooled LOX/CH_4 and LOX/C_3H_8 designs. Because the fuel is vapor both the coolant jacket and the injector ΔP's are smaller. At Pc = 100 psia the oxidizer and fuel inlet pressures for LOX/CH_4 are 143 psia and 147 psia, respectively. For comparison, the fuel inlet pressure for C_2H_5OH is 206 psia, or a 63 psi increase from LOX/CH_4. Note that the fuel inlet pressure for LOX/C_2H_5OH at Pc = 150 is much higher than the other propellant combinations at this Pc level. This results from the fact a two-pass regenerative coolant design was generated. Possibly a better design for this concept would be a one-pass jacket with some film cooling.

The engine pressure drops for the several fuels are about the same at Pc = 800 psia, but at Pc = 400 the LOX/CH_4 and C_3H_8 inlet pressures are higher because supercritical pressures were considered to circumvent the burnout problem. However, differences in pump-fed inlet pressures (or pump discharge pressures) do not affect tank pressure but the TPA ΔP. Because the trade-off of discharge pressure to engine Isp is small there is no big impact on engine performance due to engine inlet pressure differences.

c. Weights

The OMS engine weights are quite uniform. All of the pressure-fed engine weights are within 305 $\pm$ 20 lbm. The Pc = 100 psia point designs are about 30 lbm heavier than Pc = 150 psia, designs.

The only significant pump-fed engine weight variable was thrust level. At Fv = 6K all engine weights were within 327 ± 2 lbm and at Fv = 10K the weights were 393 ± 16 lbm.

The small variations in engine weights are due to the fixed envelope design constraint which results in all engines having the same or nearly the same length and diameter (nozzle exit). Also, the interface of the coolant jacket and the radiation cooled nozzle was at nearly the same diameter for all designs, hence, the coolant jacket weight of that portion downstream of the throat did not change significantly. In addition, the densities of copper, nickel and CRES are not much different (.32/.32/.28 lbm/in^3.)

2. RCS Engine

a. Performance

The RCE performance trends were the same as for the OME's:

- LOX/CH_4 point designs had the highest vacuum specific impulse.
- The higher Pc point designs had the higher performance (primarily due to fixed engine envelope).

Fuel film cooling was considered for all RCE point designs.

At Pc = 150 psia the point design performance figures are:

RCE Performance at Pc = 150 psia, sec

Propellants	820-lbf	25-lbf
LOX/CH_4	313.7	235.3
LOX/C_3H_8	305.4	229.0
LOX/NH_3	292.4	219.3
LOX/C_2H_5OH	288.3	216.2

The difference in vacuum Isp for the four propellant combustions is similar to that for the OME's with the exception of the LOX/NH_3 which appears relatively better. This is because all of the RCE propellant combinations require film cooling whereas in the OME designs only the LOX/NH_3 required it. All RCE designs were based on the same chamber material/concept: coated Columbium with a 2400°F maximum adiabatic wall temperature.

All specific impulse values were computed for TCA propellant inlet temperatures at NBP except for C_2H_5OH which was at ambient (~60°F). This approach yields valid performance for the OME and liquid-injected RCE point designs but not for the vapor-injected RCE point designs.

Some RCE point designs were defined that had gaseous oxygen (ambient temperature) and liquid fuel at the TCA interface. Also, a LO_2/CH_4 point design was defined for vapor state for both the O_2 and CH_4 at the TCA interface. Because enthalpy levels for vapor state propellant are greater, there is a performance increase associated with gas injection. This increase in energy level is significant for O_2 and CH_4 because of the large increase in propellant temperature from NBP to ambient temperature ($\Delta T > 300$°F). The table below defines the approximate increase in the reported RCE vacuum Isp numbers (liquid injection) for vapor injection.

Propellant	Propellant* States	Est. Vac. ΔIsp (from L/L)
LOX/C_3H_8	G/L (Oxid a gas & fuel liquid)	+9 seconds
LOX/CH_4	G/G	+18 seconds
LOX/NH_3	G/L	+ 7 seconds
LOX/C_2H_5OH	G/L	+ 7 seconds

*G is at ~70°F; L is at NBP

To reiterate, all of the vacuum Isp numbers presented on the enclosed tables and figures are based on TCA inlet enthalpy levels either at NBP or 70°F whichever is lower. For those RCE point designs which have vapor injection the above ΔIsp figures should be added to the reported Isp to account for the higher energy level of the vapor state.

Caution must be used in applying these Isp increases because if the energy to vaporize the propellant is not free the increase will not be realized. If, for example, the LOX is heated up by the fuel (C_2H_5OH) there is no net performance gain because the enthalpy of the fuel was reduced.

b. Inlet Pressure

Other than Pc level, the inlet pressures were found to be a function of the inlet state: liquid vs. vapor. The vapor resulted in a significant reduction in inlet pressure (~20% for Pc = 150 psia) from liquid.

This fact would result in lower supply/tank pressures for vapor propellants;however, the liquid state propellants would permit greater supply pressure fluctuations for a given thruster Pc and MR range. Also, vapor propellants would reduce the thermal soak-back problem at the thruster inlet. Vapor state presents other system problems such as greater accumulator vessel size because of lower density and energy source required for vaporization.

c. TCA Weight

Because the RCE thrusters are small their weight and weight differences are also small. All the 870-lbf RCE point designs have TCA weights of 22.7 ± 2 lbm. For 38 870-lbf RCE thrusters the weight range is ± 80 lbs.

Propellant conditioning component weights for the RCS are discussed in Reference (a) and shown in the data dump tables.

III. CONCLUSIONS/RECOMMENDATIONS

Conclusions drawn from the preceding discussion and from the entire study are shown on Table VIII. Selection of a propellant combination, a Pc level, and a thrust level (OME) were not made for the OME or RCE because of the overall system impacts which were not considered in the study.

Recommendations for further study and investigation are shown on Table IX. In general, they relate to analytical and test efforts which will provide more insight into the potential engine design problem areas and those where there is a lack of information.

NOMENCLATURE

A: Area

C_p Fluid Specific heat

Core: central or main portion of chamber exhaust flow. For this study core flow is at optimum MR.

D: diameter

Eng: Engine

F: Thrust

ft: feet

GGA: gas generator assembly

HP: horsepower

Isp: specific impulse

MR: Mixture Ratio (O/F)

MW: Molecular weight

ODK: One dimensional kinetics

OME: Orbital Maneuvering Engine

OMS: Orbital Maneuvering System

NPSP: Net Positive Suction Pressure

P: pressure

Q: Heat flux

RCE: Reaction Control Engine

RCS: Reaction Control System

RPM: Revolutions per Minute

Sec: second

T: temperature

TCA: Thrust Chamber Assembly (generally includes: Injector, valve, chamber, nozzle extension)

TPA: Turbopump Assembly (includes pump and turbine)

u: tip speed

v: velocity, vacuum

$\dot{W}$: flow rate

OTHERS

ϵ: Nozzle area ratio

η: efficiency

Δ: delta or difference

Σ: sigma or summation

γ: Specific heat ratio

Subscripts

Act: actual

aw: adiabatic wall

BO: burnout

c: chamber, coolant, combustion

C: Cold fluid

cj: coolant jacket

D: discharge

e, ex: exit

f: fuel

ffc: fuel film cooling

g: gas

GG: gas generator

H: hot fluid

i, in: inlet

inj: injector

l: liquid

out: outlet

ox, o: oxidizer, out

r: recovery, ratio

s: suction

Nomenclature (cont.)

Subscripts (cont.)

t:	throat
TCV:	Thrust Chamber Valve
tot:	total
turb:	turbine
wg:	gas side wall
wl:	liquid side wall

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Task III Engine Analysis

INTRODUCTION

OME POINT DESIGNS

PROPELLANTS	FEED MODE	F_v, LBF	Pc, PSIA	COMMENTS
LOX/C_3H_8	PRESSURE	6K	100	*
			100 [1]	Zr-Cu & Ni CHAMBER
			150	*
	PUMP	6K	400	Ni (2Cg PROFILES) & CRES CHAMBERS
			800	*
		10K	400	*
			800	* WITH & W/O BOOST PUMPS
LOX/CH_4	PRESSURE	6K	100	Zr-Cu & Ni CHAMBERS
			150	*
	PUMP	6K	400	Ni CHAMBER
		10K	400	*
			800	* FUEL & LOX REGEN GG & EXPANDER CYCLE

1. C_3H_8 INLET TEMP = AMBIENT. ALL OTHER CASES HAD NBP PROPELLANTS EXCEPT C_2H_5OH WHICH WAS 60°F

* Zr-Cu CHAMBERS

TABLE I.

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Task III Engine Analysis

INTRODUCTION

OME POINT DESIGNS - CONT.

PROPELLANTS	FEED MODE	F_v, lbF	Pc, Psia	COMMENTS
LOX/NH_3	PRESSURE	6K	100	CRES CHAMBER
	PUMP	10K	400	CRES CHAMBER
			800	CRES CHAMBER
LOX/C_2H_5OH	PRESSURE	6K	100	*
			150	*
	PUMP	10K	400	*
			800	*

* Zr-Cu CHAMBER

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TABLE I

Task III Engine Analysis

INTRODUCTION

RCE POINT DESIGNS

PROPELLANTS	F_V, lbF	Pc, Psia	COMMENTS
LOX/C_3H_8	25 & 870	100, 150, 250, 300	1. ENGINE DESIGNS ARE SIMILAR TO CURRENT RCEs, I.E., ADIABATIC WALL (2400°F) COLUMBIUM CHAMBER.
LOX/CH_4	25 & 870	150, 250	
LOX/NH_3	25 & 870	150, 250	
LOX/C_2H_5OH	25 & 870	150, 250	

TABLE II

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Task III Engine Analysis

INTRODUCTION

OME COMPONENT DESIGN CONCEPTS (TYPICAL)

• INJECTOR	• CONVENTIONAL: SAME ERE AS CURRENT DESIGN
• CHAMBER	• FUEL REGEN COOLED: SINGLE PASS/COUNTERFLOW
	• MATERIALS: Zr - Cu: C_3H_8, CH_4, C_2H_5OH NI: C_3H_8, CH_4 CRES: NH_3, C_3H_8
• NOZZLE EXTENSION	• RADIATION COOLED (Cb)
• ENGINE CYCLE	• PRESSURE-FED & PUMP-FED
	• GGA CYCLE (EVAL. ONE EXPANDER CYCLE)
• TPA CONFIGURATION	• SEPARATE TPAs FOR OXID AND FUEL
	• WITH AND WITHOUT BOOST PUMPS
• GAS GENERATORS	• INDIVIDUAL OXID AND FUEL GGA, BOTH FUEL RICH Tc = 2000°R
• TURBINE EXHAUST	• DUMPED THROUGH A 2:1 EXPANSION NOZZLE

TABLE III

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Task III Engine Analysis

SUMMARY

PRESSURE-FED OME SUMMARY OF RESULTS (F_V = 6K lbF)

PROPELLANTS	Pc, Psia	CHAMBER MAT'L	TCA MR	TCA Isp_V, Sec	%FFC	INTERFACE PRESS: OX/F	ENGINE* WEIGHT	COMMENTS
LOX/C_3H_8	100	Zr-Cu	1.92	324.3	30 (10.3)**	143/183	323.1	LIQUID REGEN
	100	Zr-Cu	2.75	337.0	Ø	143/157	322.1	AMBIENT C_3H_8 VAPOR REGEN
	100	Ni	2.75	337.0	Ø	143/161	318.0	AMBIENT C_3H_8 VAPOR REGEN
	150	Zr-Cu	1.79	326.7	35 (12.5)	209/381	285.8	LIQUID REGEN
LOX/CH_4	100	Zr-Cu	3.00	343.2	Ø	143/147	318.0	VAPOR REGEN
	100	Ni	3.00	343.2	Ø	143/150	318.0	" "
	150	Zr-Cu	3.40	346.2	Ø	209/231	289.4	" "
LOX/NH_3	100	CRES	1.25	318.8	11 (4.9)	143/159	317.7	LIQUID REGEN
LOX/C_2H_5OH	100	Zr-Cu	1.60	319.9	Ø	143/206	313.7	" "
	150	Zr-Cu	1.60	326.4	Ø	209/467	286.1	" " 2 PASS REGEN

*VAPOR REGEN ENGINE WGTS DO NOT INCLUDE ALLOWANCE FOR POTENTIAL HEAT EXCHANGER NOZZLE EXTENSION

**%FFC OF TOTAL ENGINE FLOW

TABLE IV

Aerojet Liquid Rocket Company

Task III Engine Analysis

SUMMARY

PUMP-FED OME SUMMARY OF RESULTS

PROPELLANTS	F_V/Pc	CHAMBER MAT'L	ENGINE MR	ENGINE Isp_V	% FFC	PUMP DISCHARGE PRESSURE: OX/FUEL	ENGINE WEIGHT	COMMENTS
LOX/C_3H_8	6K/400	Ni	2.58	354.9	3 (0.8)*	535/980**	328.3	NOM Cg
	"	Ni	2.66	355.3	-0-	535/980	328.3	FLAT Cg
	"	CRES	2.02	343.3	25 (7.9)	535/980	325.3	
	6K/800	Zr-Cu	2.81	368.7	-0-	1020/1196	327.9	
	10K/400	"	2.69	351.7	-0-	535/980	393.4	
	10K/800	"	2.83	363.8	-0-	1020/1135	396.5	W/O BOOST PUMPS
	"	"	2.82	363.4	-0-	1020/1135	431.0	WITH BOOST PUMPS
LOX/CH_4	6K/400	Ni	3.41	360.8	-0-	535/1080	326.9	
	10K/400	Zr-Cu	3.44	356.0	-0-	535/980	382.4	
	10K/800	"	3.43	366.0	-0-	1020/1208	382.6	FUEL REGEN
	"	"	3.39	365.3	-0-	1308/1020	382.1	OXID REGEN
	"	"			NOT FEASIBLE			EXPANDER CYCLE
LOX/NH_3	10K/400	CRES	1.21	326.5	13 (5.8)	533/614	376.8	
	10K/800	"	0.93	317.4	33 (17)	1020/1310	383.8	
LOX/C_2H_5OH	10K/400	Zr-Cu	1.77	335.5	-0-	555/1121	404.1	2 PASS REGEN
	10K/800	"	1.75	344.1	-0-	1040/1338	389.4	

* % FFC OF TOTAL ENGINE FLOW ** ALL C_3H_8 & CH_4 DESIGNS HAVE SUPERCRITICAL REGEN COOLING

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TABLE V

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Task III Engine Analysis

SUMMARY

870-LBF RCE SUMMARY OF RESULTS

PROPELLANTS	Pc, PSIA	PROP. INLET* STATE: OX/F	MR	I_{sp_V}	% FFC	PROP. INLET PRESSURE: OX/F	TCA WGT
LOX/C_3H_8	100	L/L	2.24	297.9	18.6 (5.7)	177	23.1
	150	L/L	2.23	305.4	19 (5.9)	256	22.0
	150	G/L	2.23	305.6	18.8 (5.8)	202/256	22.9
	250	G/L	2.20	315.5	20 (6.3)	316/414	21.8
	300	L/L	2.19	318.8	20.5 (6.4)	492	20.6
LOX/CH_4	150	L/L	2.49	313.7	17 (4.9)	256	22.0
	150	G/G	2.48	313.9	17.2 (5.0)	204	24.8
	250	G/G	2.43	322.2	19 (5.5)	316	23.7
LOX/NH_3	150	L/L	1.12	292.4	20 (9.4)	246	21.8
	250	G/L	1.12	301.6	20 (9.4)	316/398	21.5
LOX/C_2H_5OH	150	L/L	1.30	288.3	18.9 (8.2)	252	21.8
	250	G/L	1.41	301.0	21.8 (9.0)	316/408	21.5

*USED FOR INTERFACE PRESSURE CALCULATIONS. PERFORMANCE AND THERMAL ANALYSES ARE BASED ON G/G ONLY AT INLET.

TABLE VI

Aerojet Liquid Rocket Company

Task III Engine Analysis

SUMMARY

25-LBF RCE SUMMARY OF RESULTS

PROPELLANTS	Pc, PSIA	PROP. INLET STATE: OX/F	MR	I_{sp_V}	% FFC	PROP. INLET PRESSURE: OX/F	TCA WGT
LOX/C_3H_8	100	L/L	2.75	223.4	23	SAME AS 870-LBF RCES	8 ± 1 LBM
	150	L/L	"	229.0	"		
	150	G/L	"	229.2	"		
	250	G/L	"	236.6	"		
	300	L/L	"	239.1	"		
LOX/CH_4	150	L/L	3.00	235.3	21		
	150	G/G	"	235 4	"		
	250	G/G	"	241.6	"		
LOX/NH_3	150	L/L	1.4	219.3	39		
	250	G/L	"	226.2	"		
LOX/C_2H_5OH	150	L/L	1.6	216.2	36		
	250	G/L	1.8	225.8	33		

TABLE VII

Task III Engine Analysis

OVERVIEW

CONCLUSIONS

- PRESSURE-FED OME
  - LOX/CH_4 WITH FUEL VAPOR REGEN COOLED CHAMBER OFFERS THE: (1) HIGHEST TCA PERFORMANCE (ISP_v) AND (2) LOWEST INLET (I.E., TANK) PRESSURES.
  - CH_4 AND C_3H_8 POINT DESIGN RESULTS ARE QUESTIONABLE TO THE EXTENT THAT GAS-SIDE COKING IS UNCERTAIN.
  - TCA WEIGHTS ARE NOT A SELECTION FACTOR (I.E., ∼ SAME)
  - NICKEL REGEN CHAMBERS CAN BE COOLED WITH VAPORIZED CH_4 OR C_3H_8.
- PUMP FED OME
  - HIGHEST ISP_v POINT DESIGNS: LOX/CH_4
$F_v = 6K$
$Pc = 800$
  - ISP_v DIFFERENCE BETWEEN $F_v = 6K$ & $10K$ and LOX/CH_4 & LOX/C_3H_8 POINT DESIGNS IS ∼ 7 SEC MAX.
  - DEVELOPMENT OF LOX/CH_4 & LOX/C_3H_8 AT $Pc < P_{CRIT}$ (673) NEEDS RESOLUTION OF COKING UNCERTAINTY. FINAL COKING FACTOR IS REQUIRED TO MAKE SELECTION OF CH_4 VS C_3H_8.
  - AT $Pc = 400$ NICKEL CHAMBERS ARE FEASIBLE.
  - LOX REGEN COOLING IS FEASIBLE.
  - AT LOW NPSP VALUES (2-10) BOOST PUMPS SHOULD BE CONSIDERED.
  - SUBCOOLED PROPANE (-295°F) MAY ELIMINATE NEED FOR BOOST PUMPS AT LOW NPSP VALUES.

TABLE VIII

Aerojet Liquid Rocket Company

Task III Engine Analysis

OVERVIEW

CONCLUSIONS (CONT.)

- RCE
  - MAX ISP_V IS ACHIEVED WITH LOX/CH_4 (6 - 8 SEC HIGHER THAN LOX/C_3H_8)
  - VAPOR STATE PROPELLANTS YIELDS MINIMUM TCA INLET PRESSURES.
  - LIQUID STATE PROPELLANTS SHOW LOWER Pc & MR EXCURSIONS FOR GIVEN INLET PRESSURE RANGE.

TABLE VIII

Aerojet Liquid Rocket Company

Task III Engine Analysis

OVERVIEW

RECOMMENDATIONS

- OME
  - CONDUCT COMBUSTION TECHNOLOGY PROGRAMS TO CHARACTERIZE GAS-SIDE COKING FOR LOX/CH_4 AND LOX/C_3H_8. PROGRAMS SHOULD ALSO BE DIRECTED TOWARD CONFIRMING: (1) GAS-SIDE C_G PROFILE, (2) COOLANT SIDE COKING CHARACTERISTICS, (3) LIQUID-SIDE h_L COEF., AND (4) FILM COOLING ENTRAINMENT FACTOR.
  - COMPLETE F_V/Pc POINT DESIGN ENGINE STUDY MATRIX FOR NICKEL CHAMBERS WITH CH_4 & C_3H_8 (POTENTIAL SOLUTION FOR COPPER INCOMPATIBILITY).
  - CONDUCT FUEL RICH COMBUSTION TECHNOLOGY PROGRAMS TO CONFIRM GGA EXHAUST CHARACTERISTICS.
  - FOR SELECTED POINT DESIGNS PERFORM THERMAL ANALYSIS AND DEFINE PRESSURE SCHEDULE FOR ALL FOUR CORNERS OF Pc/MR OPERATING BOX
  - CONDUCT SPARK AND TORCH IGNITER PROGRAMS/STUDIES.
  - EVALUATE SMALL/HIGH SPEED TURBOPUMPS WITH CRYOGENIC PROFELLANTS. I.E., DESIGN AND LAYOUT.
  - EVALUATE LOW NPSP BOOST PUMPS.
  - NEXT STUDY PHASE INCLUDE ENGINE START/SHUTDOWN TRANSIENT ANALYSES INCLUDING CHILLDOWN REQ'TS AND CONTROL SYSTEM REQ'TS FOR VARYING INLET/SUCTION PRESSURES.
- RCE
  - CONDUCT COMBUSTION TECHNOLOGY PROGRAMS TO CHARACTERIZE CARBON FORMATION AND FILM COOLANT ENTRAINMENT FACTORS.
  - PERFORM PRELIMINARY DESIGN AND ANALYSIS STUDIES TO EVALUATE HEAT SOAK-BACK TO VALVE AND INJECTOR.

TABLE IX

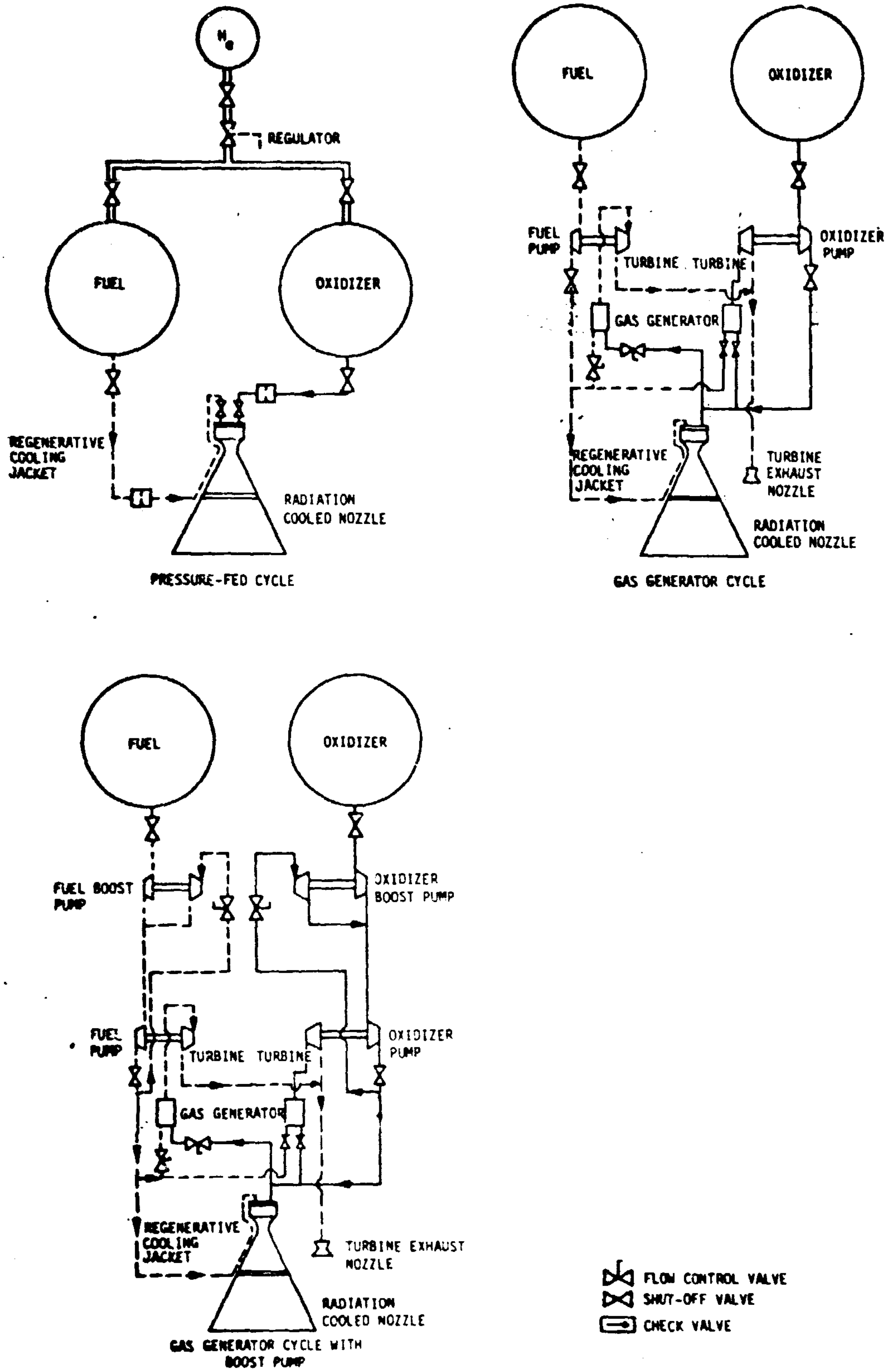

Figure 1 Candidate OME Cycles

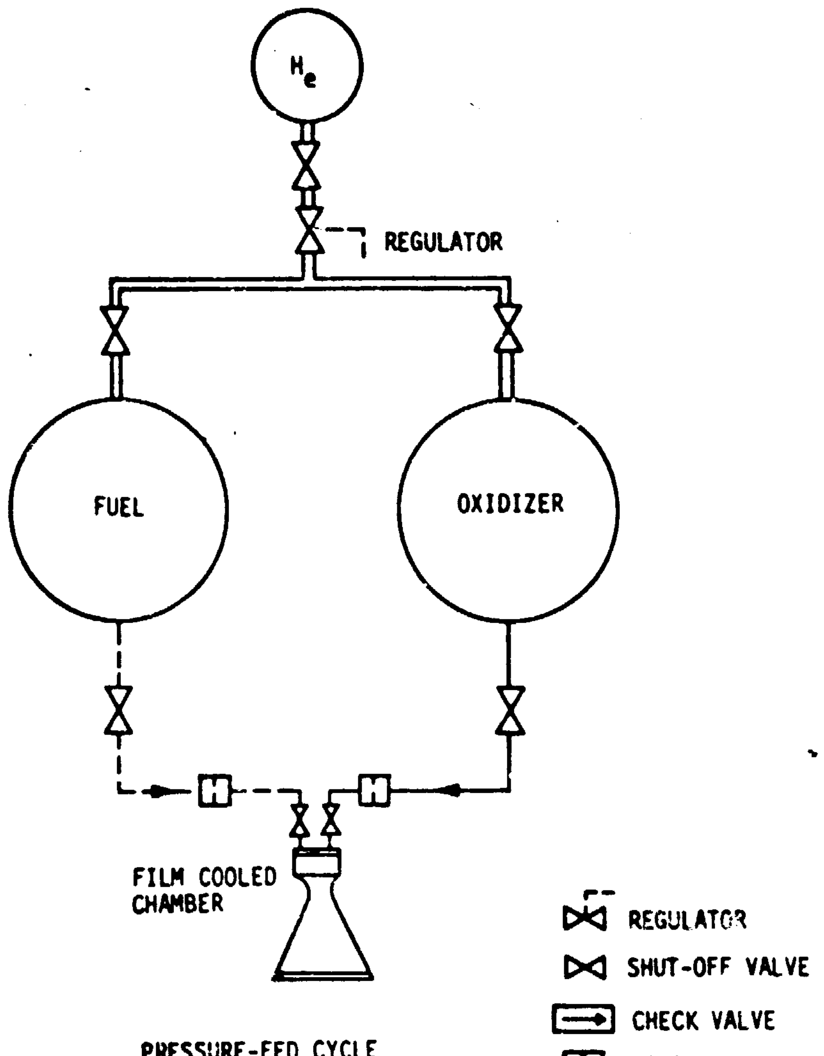

Figure 2 Candidate RCE Cycles

Page 1 of 2

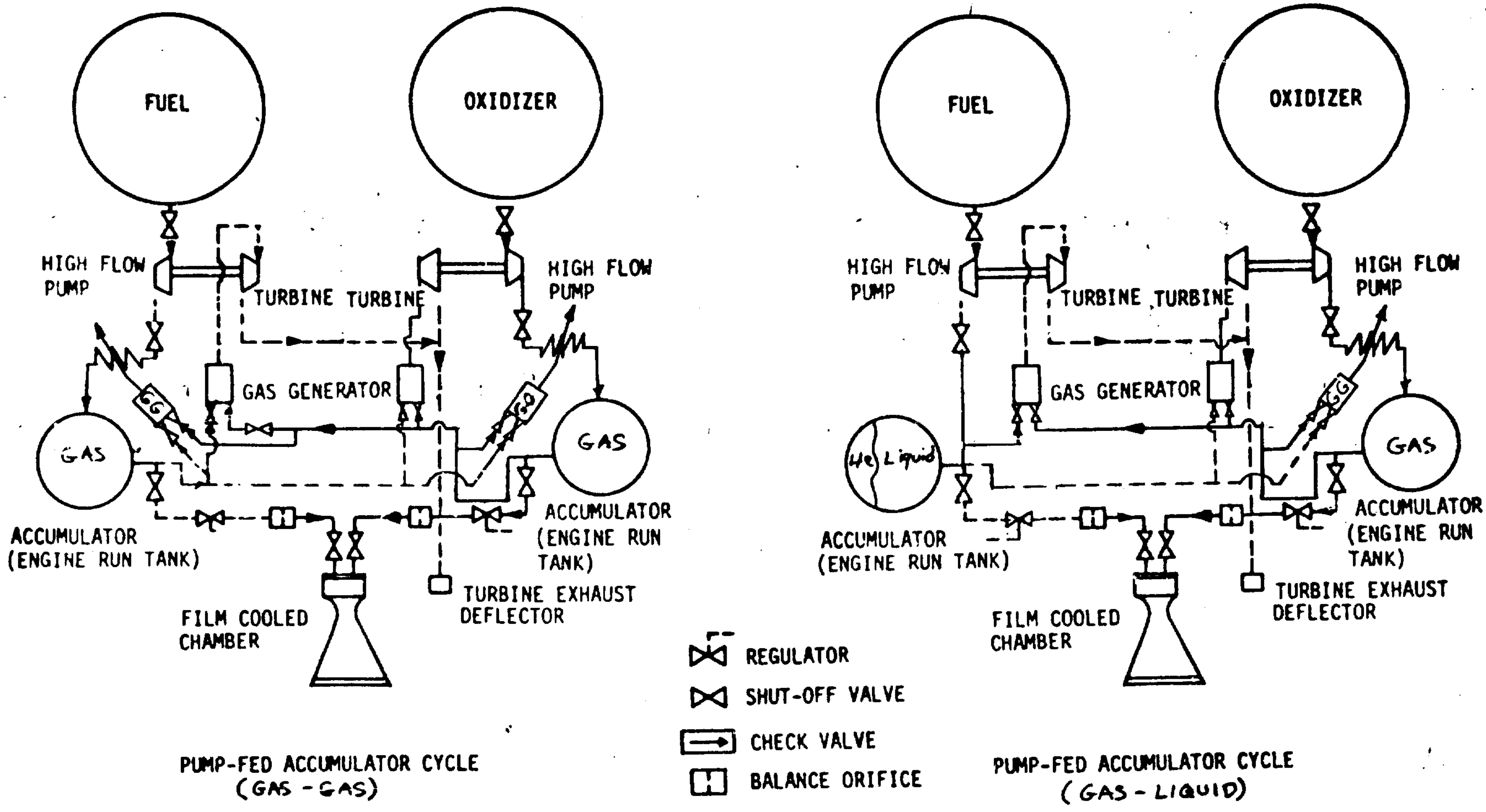

Figure 2 Candidate RCE Cycles

APPENDIX I

BASELINE POINT DESIGN DATA DUMP

3.0 PRESSURE SCHEDULE

3.1 OME Concept

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K (Boost Pump)	100/6K	800/10K	100/6K	800/10K	
• Plenum Pc, psia	100	800	800		800	100	800	
• Face Pc, psia	103	824	824		824	103	824	Based on A_c/A_t=3.3
• ΔPinj, psi*	35	160	160		160	35	160	Based on Chug Criteria
• ΔP_{TCV}, psi	5	20	20		20	5	20	Typical for existing engines
• ΔP_{lines}, psi	-	16	16		16	-	16	2% of Pc
• ΔP_{cj}, psi	40	115	115		188	16	290	
• ΣΔ P, psi	80	311	311		384	56	486	
• Interface or pump discharge pressure (ox/fuel), psia	143/183	1020/1135	1020/1135		1020/1208	143/159	1020/1310	
• ΔPinj/Pc	0.35	0.20	0.20	-	0.20	0.35	0.20	
• ΔP_{cj}/Pc	0.40	0.14	0.14		0.23	0.16	0.36	

NOTE: Pressure schedules for both OME and RCE engines are at the nominal operating Pc and MR.

*Pump-fed min. ΔPinj=0.2 x Pc

Pressure-fed ΔPinj:

- Liquid injection: min. ΔPinj/Pc=0.2
- Gas injection: min. ΔPinj/Pc=0.15

Note: Min. ΔPinj/Pc occurs at low Pc and high MR corner of operating box.

3.2 RCE Concept

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• Plenum Pc, psia	150	250	150	250	150	250	
• Face Pc, psia	154	258	154	258	154	258	Based on Ac/At=3.3
• ΔP_{inj}, psi* (ox/fuel)	82	38/136	82	38/38	72	38/120	Based on chug criteria
• ΔP_{TCV}, psi	20	20	20	20	20	20	Typical for existing engines.
• $\Sigma\Delta P$, psi	120	58/156	102	58/58	92	58/140	
• Interface Pressure, psia	256	316/414	256	316/316	246	316/398	
• $\Delta P_{inj}/P_c$ (ox/fuel)	0.55	0.15/0.54	0.55	0.15/0.15	0.48	0.15/0.48	

Note: Pressure schedule is based on nominal operating Pc and MR

*Based on pressure-fed criteria (reference 3.1 OME concept).

4.0 CHAMBER THERMAL ANALYSIS

4.1 OME Concepts

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K	100/6K	800/10K	100/6K	800/10K	
			(Boost Pump)					
• Thrust, lbs	4400	9058	Same as design point w/o boost pump	Regenerative cooling with subcooled CH_4 is not feasible. Gaseous CH_4 will be examined in Phase II of study.	8867	4550	8905	
• Pc, psia	75	720			720	75	720	
• MR_{TCA}	2.31	3.15			3.68	1.495	1.13	
• MR_{Core}	3.3	3.15			3.68	1.68	1.47	
• $\dot{W}ox$, lb/sec	9.62	18.84			18.79	8.66	14.52	
• $\dot{W}_f$, lb/sec	4.16	5.98			5.11	5.79	9.88	
• No.of Regen Passes	1 (up)	1 (up)			1 (up)	1 (up)	1 (up)	Counterflow
• ΔPc.j., psi	17	90			146	7	140	
• Pc.j.-in.psia	150	1080			1080	150	1080	
• Pc.j.out, psia	133	990			934	143	940	
• Tc.j.-in.°F	-44	-44			-259	-28	-28	
• Tc.j.-out* °F	28	186			-12	34	30	
• ΔTc.j.,°F	72	230			247	62	58	
• Regen ε	6.2	23.8			23.6	6.2	30.9	Radiation cooled nozzle attachment 6% entrainment factor
• $\dot{W}_{ffc}$, lb/sec	1.25**	0			0	0.64***	2.96***	
• %Fuel Film Coolant	30	0			0	11	30	
• T_{ffc-in}, °F	28	-			-	-28	-28	
• Twg max, °F****	222[1]	835[1]			890[1]	440[2]	878[2]	1. 1000°F max 2. 800°F max.

NOTE: All thermal analyses were performed at low Pc and high MR corner of operating box. This is the most severe operating point.

*Bulk temperature of coolant is based on coked gas side wall: C_3H_8 $Q_{act}/Q = 0.42$
CH_4 $Qact/Q = 0.765$

**Total fuel flow used for regenerative cooling.

***Fuel film cooling does not pass through regenerative coolant jacket.

****Twg is based on a carbon free wall surface.

4.0 CHAMBER THERMAL ANALYSIS (cont.)

4.1 OME Concepts (cont.)

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K (Boost Pump)	100/6K	800/10K	100/6K	800/10K	
• Tw_1 max	157[1]	796[1]			852[2]	152[1]	201[1]	1. 800°F max 2. 1000°F Max
• h_g*, BTU/in^2-sec-°F	.00133	.00493			.00490	.00094	.00791	
• h_1*, BTU/in^2-sec-°F	.00654	.0321			.0404	.0186	.0947	
• Tr*, °F	1510	5960			5840	2773	1728	
• Q/A_g max, BTU/in^2 sec	1.74	26.3			26.7	2.20	6.9	
• Q/A_1 max, BTU/in^2 sec	1.02	16.9			14.6	2.32	7.5	
• $\frac{Q/A}{Q/A_{Bo}}$ max	0.77[1]	NA			NA	0.59[2]	0.54[2]	1. 0.77 max 2. 0.60 max
• Q Total, BTU/sec	160	868.8			1450	345	617	
• V_c max, ft/sec	38.3	136			242	28.1	180	
• V_c (Mach No)max	-	.044			0.234	-	-	0.3 max
• No of channels	350	145			143	328	144	
• Min Ch Depth, in	.038	.040			.036	.060	.041	
• ΔPc.j./Pc	0.23	0.12			0.13	0.09	0.19	
• Limiting Criteria	$\frac{Q/A}{Q/A_{Bo}}$	T_{wl}			T_{wg}-$T_{closeout}$	$\frac{Q/A}{Q/A_{Bo}}$	T_{wg}	

*@ max. flux.

4.2 RCE Concepts

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• 870 Lb Thrusters							
• Thrust, lbs	520	520	520	520	520	520	
• Pc, psia	90	150	90	150	90	150	
• MR_{TCA}	3.14	3.09	3.48	3.57	1.56	1.56	
• MR_{core}	3.85	3.85	4.20	4.41	1.96	1.96	
• $\dot{W}_{ox}$, lb/sec	1.386	1.335	1.390	1.361	1.123	1.088	
• $\dot{W}_f$, lb/sec	.442	.431	0.400	.381	.719	.697	
• $\dot{W}_{ffc}$, lb/sec	.082	.084	.069	.073	.147	.142	Saturated vapor at injection
• % Fuel Film Coolant (of $\dot{W}_f$)	19	20	17	19	20	20	6% entrainment factor
• Taw Max, °F	2400	2400	2400	2400	2400	2400	2400°F maximum
• % Fuel Film Cool. (of total flow)	4.5	4.8	3.9	4.2	8.0	8.0	
• 25 Lb Thrusters							Concept is similar to LO_2/RP-1 igniter which has duct film cooling. Core MR is 20:1 to reduce Twg. Selected overall MR is at Max. Isp.
• Thrust, lbs							
• Pc, psia							
• MR_{TCA}							
• MR_{core}							
• $\dot{W}_{ox}$, lb/sec							
• $\dot{W}_f$, lb/seo							
• $\dot{W}_{ffc}$, lb/sec							
• % Fuel Film Coolant							
• Taw Max, °F							

5.0 TPA AND GGA ANALYSIS

5.1 OME Concepts

PROPELLANTS	LOX/C_3H_8		LOX/CH_4	LOX/NH_3	COMMENTS
PUMPS	MAIN	MAIN + B/P	MAIN	MAIN	
• Pumps					
• $\dot{W}_{ox}$, lbm/sec	20.39	20.39/22.3*	20.84	15.84	
• $\dot{W}_f$, lbm/sec	6.80	6.8/7.3	5.96	14.98	
• $NPSP_{ox}$, psia	20.3	1.0/37	41	34	
• $NPSP_f$, psia	20.3	1.0/25	12.3	23	
• P_{iox}, psia	35	15.7/51	[illegible]	49	
• P_{if}, psia	35	15.7/39	27	38	
• P_{Dox}, psia	1040	51/1040	1040	1040	
• P_{Df}, psia	1155	39/1155	1123	1330	
• T_{sox}, °R	162.7	162.7	162.7	162.7	
• T_{s_f}, °R	416.2	416.2	217	432	
• Spec. Spd_{ox}	1592	4157/1573	2870	2500	
• Spec SPd_f	1546	4157/1184	1293	1,870	
• Suct. Spec. Spd_{ox}	30,000	30K/20K	30,000	30,000	
• Suct. Spec Spd_f	30,000	30K/20K	34,750	35,770	
• No. of $Stages_{ox}$	1				
• No. of $Stages_f$	1				

*Boost Pump/Main Pump

NOTE: See page 3 of 3 for additional pump data

5.0 TPA AND GGA ANALYSIS (cont.)

5.1 OME Concepts (cont.)

PROPELLANTS	LOX/C_3H_8		LOX/CH_4	LOX/NH_3	COMMENTS
PUMPS	MAIN	MAIN + B/P	MAIN	MAIN	
• Imp. D_{ox}, in	2.0	2.4/2.1*	1.26	1.23	
• Imp. D_f, in	1.38	1.8/1.8	1.63	1.62	
• η_{ox}, %	62.6	74/60.3	58.4	57.8	
• η_f, %	58.4	72/57	59.5	61	
• Turbines					
• Pin, psia	790	936/790	790	790	
• Pout, psia	79	88/79	79	79	
• Pr	10	-/10.0	10	10	
• $\dot{W}_{GG_{ox}}$, lbm/sec	0.42	.45	0.278	0.268	For ox TPA
• $\dot{W}_{GG_f}$, lbm/sec	0.28	.30	0.231	0.494	For fuel TPA
• T_{in}, °R	2000	-/2000	2000	2000	
• Tout, °R	1647	-/1624	1566	1539	
• ΔT, °R	353	-/376	434	461	
• Spec. Spd_{ox}	7.8	7.5	9.3	9.7	
• Spec Spd_f	12	9.6	10	14	
• No. of $Stages_{ox}$	1	1/1	1	1	
• No. of $Stages_f$	1	1/1	1	1	
• Tip D_{ox}, in	7.0	3.2/7.5	4.6	4.6	
• Tip D_f, in	4.0	2.4/5.1	3.9	4.3	
• η_f, %	67	51/62	64.6	68	
• η_{ox} %	62	46/64	64.0	64.3	

*Boost Pump/Main Pump

NOTE: See page 3 of 3 for additional turbine data

5.0 TPA AND GGA ANALYSIS (CONT.)

5.1 OME Concepts (cont.)

PROPELLANTS	LOX/C_3H_8		LOX/CH_4	LOX/NH_3	COMMENTS
PUMPS	MAIN	MAIN + B/P	MAIN	MAIN	
• Gas Generator					
• Pc_{GG}, psia	800	800	800	800	
• $\dot{W}_{GG_{ox}}$, lbm/sec	0 [illegible]	0.45	0.28	0.268	For Ox TPA
• $\dot{W}_{GG_f}$, lbm/sec	0.28	0.30	0.23	0.494	For fuel TPA
• MR	0.36	0.36	1.2	0.57	
• Cp, BTU/lbm-°F	0.64	0.64	0.78	0.59	
• γ	1.18	1.18	1.23	1.25	
• MW	20	20	13.5	16.6	
• Tc, °R	2000	2000	2000	2000	
• Additional Data					
Pumps					
• Oxid Flow, GPM	128.4	128.4/141*	131	100	
• Fuel Flow, GPM	84	84/90	101	158	
• Oxid Speed, RPM	45,630	9,470/42,700	74,550	74,900	
• Fuel Speed, RPM	90,800	14,420/67,000	87,250	80,000	
• Impeller Tip Spd.					
• Oxid, ft/sec	394	104/391	410	402	
• Fuel, ft/sec	549	121/529	623	567	
• Shaft Power					
• Oxid, HP	132	3.9 /144	133	103	
• Fuel, HP	93	1.8/100	112	202	
Turbine					
• Blade Tip Spd (u)					
• Oxid, ft/sec	1411	132/1400	1500	1500	
• Fuel, ft/sec	1576	153/1499	1500	1500	
• Ratio u/spouting velocity (u/v)					
• Oxid	0.32	0.32	0.29	0.32	
• Fuel	0.36	0.34	0.29	0.32	

*Boost Pump/Main Pump

6.0 PERFORMANCE ANALYSIS

6.1 OME Concepts

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K (Bst Pump)	100/6K	800/10K	100/6K	800/10K	-
• Engine Fv, lbf	6000	10,099	10,106		10,084	6000	10,107	
• TCA Fv, lbf	6000	10,000	10,000		10,000	6000	10,000	
• Engine MR	1.92	2.82	2.81		3.43	1.25	.93	
• TCA MR	1.92	3.0	3.0		3.5	1.25	.94	
• Core MR	2.75	3.0	3.0		3.5	1.40	1.40	Max. ODK Isp MR
• Film Barrier MR	0.61	-	-		-	0.50	0.38	
• Turbine Ex. Fv, lbf	-	99	106		84	-	107	
• TCA $\dot{W}_{Tot}$, lbm/sec	18.50	27.06	27.06		27.03	18.82	31.08	
• TCA $\dot{W}_{ox}$, lbm/sec	12.16	20.30	20.30		21.03	10.46	15.06	
• TCA $\dot{W}_f$, lbm/sec	6.34	6.76	6.76		6.00	8.36	16.02	
• $\dot{W}_{turb}$, lbm/sec	-	0.70	0.75		0.51	-	0.76	
• % Fuel Film Coolant (of fuel flow)	30	0	0		0	11	33	
• Eng $\dot{W}_{ox}$, lbm/sec	12.16	20.49	20.50		21.31	10.46	15.34	
• Eng $\dot{W}_f$, lbm/sec	6.34	7.27	7.31		6.23	8.36	16.50	
• Eng Isp, Sec	324.3	363.8	363.4		366.1	318.8	317.4	
• TCA Isp, sec	324.3	369.5	369.5		369.9	318.8	321.7	
• Core Isp (ODK), sec	350.1	387.7	387.7		388.6	337.8	362.7	
• ISP_{turb}, sec	-	141.8	141.8		164.3	-	141.2	
• Ae/At	44	240	240		236	44	224	
• D_t, in.	6.48	2.78	2.78		2.80	6.48	2.95	
• D_e, in	43	43	43		43	43	43	
• % Fuel Film Coolant (of total flow)	10.3	-	-		-	4.9	17	
• Engine Total Flow Rate, lbm/sec	18.50	27.76	27.81		27.54	18.82	31.84	

6.2 RCE Concept

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• 870 lbf Thrusters							
• TCA MR	2.23	2.20	2.49	2.43	1.12	1.12	
• Core MR	2.75	2.75	3.0	3.0	1.40	1.40	Max. ODK Isp MR
• TCA $\dot{W}_{ox}$, lbm/sec	1.97	1.90	1.98	1.91	1.57	1.52	
• TCA $\dot{W}_f$, lbm/sec	0.88	.86	.79	.79	1.40	1.36	
• %$\dot{W}_{ffc}$, (of fuel flow)	19 (5.9)*	20 (6.3)	17 (4.9)	19 (5.5)	20 (9.4)	20 (9.4)	
• TCA Isp, Sec	305.4	315.5	313.7	322.2	292.4	301.6	
• Core Isp(ODK), sec	399.7	351.7	347.1	358.1	327.3	337.0	
• Ae/At	27	46	27	46	25	46	
• D_t, in	2.04	1.56	2.02	1.56	2.06	1.56	
• D_{ex}, in	10.6	10.6	10.6	10.6	10.6	10.6	
• 25 lbf Thrusters							
• TCA MR	2.75	2.75	3.0	3.0	1.4	1.4	
• Core MR	20	20	20	20	20	20	
• TCA $\dot{W}_{ox}$, lbm/sec	.080	.077	.080	.078	.066	.064	
• TCA $\dot{W}_f$, lbm/sec	.029	.028	.027	.026	.047	.046	
• %$\dot{W}_{ffc}$, (of total flow)	23	23	21	21	39	39	
• TCA Isp, sec	229.0	236.6	235.3	241.6	219.3	226.2	
• Core Isp(ODK), sec	-	-	-	-	-	-	
• Ae/At	27	46	27	46	25	46	
• D_t, in	0.36	0.27	0.35	0.27	0.36	0.27	
• D_{ex}, in	1.80	1.80	1.80	1.80	1.80	1.80	

*Film cooling as % of total flow

7.0 WEIGHT (LBM)

7.1 OME Concepts

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K (Boost Pump)	100/6K	800/10K	100/6K	800/10K	
• TCA (each)								
. Injector	20.4	8.4	8.4		8.5	20.1	9.5	
. Chamber	93.0	63.3	63.3		65.1	71.0	62.1	
. Nozzle	82.3	79.7	79.7		80.8	81.2	82.7	
. Controls+TCA Instr.	18.3	19.3	19.3		19.3	18.3	19.3	Does not include TCA Valve
	196.0	170.7	170.7		173.7	190.6	173.6	
• Thrust Structure Assy.	21.3	30.5	30.5		30.5	21.3	30.5	Scaled from OME
• Gimbal System	52.9	74.4	74.4		74.4	52.9	74.4	Scaled from OME
• Plumbing*	21.7	16.9	20.9		16.9	21.7	16.9	
• TPA (ox/fuel)*	-	24.7/5.7	28.6/11.1		7.6/6.0	-	7.5/7.3	
• Boost Pump(ox/fuel)*	-	-	7.3/3.3		-	-	-	
• GGA (ox/fuel)*	-	2.4/2.4	2.4/2.4		2.4/2.3	-	2.3/2.5	
• Controls & Instr								
. TCA Valve	21.0	21.0	21.0		21.0	21.0	21.0	
. Pneumatic Pack	7.6	7.6	7.6		7.6	7.6	7.6	
. Purge Valves	Ø	Ø	Ø		Ø	Ø	Ø	
. Instru*	2.6	9.6	9.6		9.6	2.6	9.6	
. GGA Valves*	-	7.8	7.8		7.8	-	7.8	
. TPA Controller*	-	22.8	25.0		22.8	-	22.8	
. Boost Pump* Circuit Valves	-	-	8.4		-	-	-	
	31.2	68.8	79.4		68.8	31.2	68.8	
Total, lbm	323.1	396.5	431.0		382.6	317.7	383.8	

*For two TPA's double these weights.

7.2 RCE Concepts

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• <u>870-lbf Thruster</u>							
• TCA (each)							
. Valves	2.5	3.4	2.5	5.3	2.3	3.1	
. Injector	5.3	4.2	5.3	4.2	5.3	4.2	
. Chamber/Nozzle	4.8	4.8	4.8	4.8	4.8	4.8	
. Insulation + Miscellaneous	9.4	9.4	9.4	9.4	9.4	9.4	
	22.0	21.8	22.0	23.7	21.8	21.5	
• Propellant Conditioning							
. Heat Exchanger (ox/fuel)	-	26.3/-	-	26.3/26.5	-	23.7/-	
. GGA(ox/fuel)	-	5.5/-	-	5.5/4.5	-	5.7/-	
		31.8		62.8		29.4	
• Controls & Instr							
. Pressure Reg. (ox/fuel)		6.2/2.0		6.2/6.2		6.2/2.0	
. Accumulator Valves (ox/fuel)		7.6/3.8		7.3/7.3		7.4/4.2	
. TPA GGA Valves		7.8		7.6		8.5	
. Prop.Cond GGA Valves		3.3		10.9		5.9	
. Main Propellant Valves(ox/fuel)		4.3/3.8		4.2/4.9		4.5/4.2	
. Instr.		14.4		19.2		14.4	
. TPA Controller		36.0		36.0		36.0	
		89.2		109.8		92.3	

7.2 RCE Concepts (continued)

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• <u>25-lbf Thruster</u>							
TCA (each)	W_B	W_B+0.5	W_B	W_B+ 1.0	W_B	W_B+0.5	W_B is notation for the basic 25-lbf thruster weight which is 5-10 lbm. Deviations shown reflect valve weight differences

8.0 ENVELOPE/SIZE

8.1 OME Concepts

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		COMMENTS
Pc/F	100/6K	800/10K	800/10K	100/6K	800/10K	100/6K	800/10K	-
• TCA (each)			(Boost Pump)					Same as existing TCA (approx. 77" X 46")
. Length, in.	-	-	-	-	-	-	-	
. Nozzle Dia.in	-	-	-	-	-	-	-	
• TPA (ox/fuel)								
. Length, in.	-	5.6/5.0	5.2/5.0	-	3.6/4.2	-	3.6/3.8	
. Diameter, in.	-	7.5/4.4	6.0/4.4	-	4.8/4.6	-	4.8/4.4	
• GGA(ox/fuel)								
. Length, in.	-	10	10	-	10	-	10	
. Diameter, in	-	4	4	-	4	-	4	
• Boost Pump (ox/fuel)								
• Length, in.	-	-	5.2/5.6	-	-	-	-	
• Diameter, in.	-	-	4.4/3.6	-	-	-	-	

8.2 RCE Concepts

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	150	250	150	250	
• 870 Lbf Thruster	-	-	-	-	-	-	Same as existing TCA's (approx.19" X 11")
• 25 Lbf Thruster	-	-	-	-	-	-	Same as existing TCA's (approx. 11" X 6")
• Heat Exchangers (ox/fuel)							
. Length,in	-	20	-	20/20	-	19	
. Diameter,in	-	12	-	12/11	-	11	
• GGA's (ox/fuel)							
. Length, in	-	11.0/-	-	11.0/11.0	-	11.0/-	
. Diameter, in.	-	5.1/-	-	5.1/4.3	-	5.4/-	

9.0 HEAT EXCHANGER AND GGA ANALYSES

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3		COMMENTS
Pc	150	250	250	250	150	250	
			Ox	Fuel			
• $\dot{W}_C$, lbm/sec		21	21	6		15	
• T_{C_i}, °R		162	162	200		162	
• T_{C_o}, °R		310	310	380		310	
• P_{C_i}, psia		900	900	900		900	
• P_{C_o}, psia		800	800	800		800	Judgment
• $\dot{W}_H$, lbm/sec		3.4	2.9	1.6		2.7	
• T_{H_i}, °R		2,000	2,000	2,000		2,000	Fuel Rich GGA
• T_{H_o}, °R		800	800	800		800	Judgment
• P_{H_i}, psia		600	600	600		600	Judgment
• P_{H_o}, psia		300	300	300		300	Judgment
• ΔQ_C, Btu/sec		2,184	2,184	1,200		1,560	
• % η		80	80	80		80	Assumption
• $\dot{W}_H/\dot{W}_C$		0.16	0.14	0.27		0.18	

APPENDIX II

PARAMETRIC POINT DESIGN DATA DUMP

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TABLE : PRESSURE SCHEDULE - OME

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4		LO_2/NH_3	COMMENTS
Pc/F	100/6K	150/6K	400/10K	800/6K	150/6K	400/10K	400/10K	
• Plenum Pc, psia	100	150	400	800	150	400	400	
• Face Pc, psia	103	154	412	824	154	412	412	Based on Ac/At = 3.3
• ΔPinj, psi * (ox/fuel)	35/17	50	95	160	50/34	95	93	Based on chug criteria
• ΔP_{TCV}, psi	5	5	20	20	5	20	20	Typical for exist. engines
• ΔP_{lines}, psi	-	-	8	16	-	8	8	2% of Pc
• ΔP_{cj}, psi	32**	172	417***	176	38**	417***	81	
• ΣΔ P, psi	40/54	55/227	123/568	196/372	55/77	123/568	121/202	
• Interface or pump discharge pressure (ox/fuel) psia	143/157	209/381	535/980	1020/1196	209/231	535/960	533/614	
• ΔPinj/Pc	.35/.17	0.33	0.24	0.20	.33/.23	0.24	0.23	
• ΔP_{cj}/Pc	0.32	1.15	.03	0.22	0.25	.03	0.20	

NOTE: Pressure schedules for both OME and RCE engines are at the nominal operating Pc & MR.

*Pump-fed min. ΔP_{inj} = 0.2 x Pc

**Includes 15 psi for ΔP across heat/exchanger (nozzle)

***Supercritical fuel cooling. Actual ΔPc.j. = 10 - 13 psi. Remaining ΔP is achieved across a throttling valve.

Pressure-fed OME and RCE ΔP_{inj}:

- Liquid injection: min. ΔP_{inj}/Pc = 0.2
- Gas injection: min. ΔP_{inj}/Pc = 0.15

Note: min. ΔP_{inj}/Pc occurs at low Pc and high MR corner of operating box.

TABLE : PRESSURE SCHEDULE - RCE

PROPELLANTS	LO_2/C_3H_8			LO_2/CH_4				COMMENTS
Pc/F	100/870	150/870	300/870	150/870				
• Plenum Pc, psia	100	150	300	150				
• Face Pc, psia (ox/fuel)	103	154	309	154				Based on Ac/At = 3.3
• ΔPinj, psi *	54	28/82	163	30				Based on chug criteria
• ΔP_{TCV}, psi	20	20	20	20				Typical for exist engines
• ΔP_{lines}, psi	-	-	-	-				2% of Pc
• ΔP_{cj}, psi	-	-	-	-				
• ΣΔ P, psi	74	48/102	183	50				
• Interface or pump discharge pressure (ox/fuel) psia	177	202/256	492	204				
• ΔPinj/Pc	.54	.19/.55	.54	.20				
• $\bar{\Delta P}_{cj}/Pc$	-	-	-	-				

TABLE : CHAMBER THERMAL ANALYSIS - OME CONCEPTS

PROPELLANTS	LOX/C_3H_8	LOX/C_3H_8	LOX/C_3H_8	LO_2/C_3H_8	LOX/CH_4	LOX/CH_4	LOX/NH_3	COMMENTS
Pc/F	100/6K	150/.6K	400/10K	800/6K	150/6K	400/10K	400/10K	-
• Thrust, lbs	4500	4500	9000	5400	4500	9000	9000	
• Pc, psia	75	112.5	360	720	112.5	360	360	
• MR_{TCA}	3.30	2.14	2.94	3.15	4.08	3.68	1.28	
• MR_{Core}	3.30	3.30	2.94	3.15	4.08	3.68	1.47	
• $\dot{W}$ox, lb/sec	10.23	8.94	19.00	10.95	10.47	19.76	15.96	
• $\dot{W}_f$, lb/sec	3.10	4.17	6.43	3.47	2.56	5.37	10.86	
• No.of Regen Passes	1	1	1	1	1	1	1	
• ΔPc.j., psi	5	76	11	132.8	15	8	63	
Pc.j.-in.psia	131	197	1200	1080	197	1200	630	
• Pc.j.out, psia	126	121	1189	947.2	182	1192	567	
• Tc.j.-in.°F	90	-44	-40	-44	-160	-259	-28	
• Tc.j.-out, °F	379	28	112	202	641	-82	12	
• ΔTc.j.,°F	289	72	156	246	801	177	40	
• Regen ε	6.23	6.23	10.64	23.76	6.23	10.64	6.73	Radiation cooled nozzle attachment area ratio
• $\dot{W}_{ffc}$, lb/sec	-	1.46	-	-	-	-	1.41	
• %Fuel Film Coolant	-	35	-	-	-	-	13.0	
• T_{ffc-in}, °F	-	28	-	-	-	-	126	
• Twg max, °F	802	161	787	949	1000	782	747	

Note: All thermal analyses were performed at low Pc and high MR corner of operating box. This is the most severe operating point.

TABLE : CHAMBER THERMAL ANALYSIS - OME CONCEPTS (CONT.)

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PROPELLANTS	LOX/C_3H_8	LOX/C_3H_8	LOX/C_3H_8	LOX/C_3H_8	LOX/CH_4	LOX/CH_4	LOX/NH_3	COMMENTS
Pc/F	100/6K	150/6K	400/10K	800/6K	150/6K	400/10K	400/10K	-
• Tw_1 max	800	144	744	780	995	739	160	
• h_g^*, BTU/in^2-sec-°F	.000371	.00193	.00262	.00477	.000995	.00262	.00368	
• h_1^*, BTU/in^2-sec-°F	.00138	.00955	.0132	.0200	.00157	.0134	.0676	
• Tr^*, °F	5346	1515	5740	5909	5346	5740	2233	
• Q/A_g max, BTU/in^2-sec	1.81	2.64	13.51	23.64	4.55	14.14	5.47	
• Q/A_1 max, BTU/in^2-sec	.40	1.41	6.37	11.97	1.03	5.70	6.25	
• $\frac{Q/A}{Q/A_{BO}}$ max.	-	.77	-	-	-	-	.454	
• Q Total, BTU/sec	186	170	584	554	526	1059	463	
• V_c max, ft/sec	155	51.8	47.2	106.9	309	39.8	111	
• V_c (Mach No)^{max}	.181	-	.014	.058	.177	.025	-	0.3 max
• No of channels**	323	263	207	112	263	206	208	
• Min Ch Depth, in	.082	.030	.084	.030	.040	.099	.050	.030 in. min.
• Limiting Criteria	T_{wl}	$\frac{Q/A}{Q/A\ B.O.}$	None	T_{wl}	T_{wg}	None	T_{wg}	
• Chamber Contraction Ratio	2.0	2.0	3.3	3.3	2.0	3.3	3.3	
• % Fuel Regen. Cooling	40	100	100	100	40	100	100	

* @ max-flux.
**At throat land width = .030" and channel width = .0325"

TABLE : CHAMBER THERMAL ANALYSIS - RCE CONCEPTS

PROPELLANTS	LO_2/C_3H_8			LO_2/CH_4			COMMENTS
Pc	100	150	300	150			
• 870 Lb Thrusters							Chamber I.D. = 3.9"
• Thrust, lbs	520	518	527	520			
• Pc, psia	60	90	180	90			
• MR_{TCA}	3.13	3.13	3.06	3.48			
• MR_{core}	3.85	3.85	3.85	4.20			
• $\dot{W}_{ox}$, lb/sec	1.624	1.263	1.220	1.291			
• $\dot{W}_f$, lb/sec	.518	.404	.399	.371			
• $\dot{W}_{ffc}$, lb/sec	.096	.076	.082	.064			Saturated vapor injection.
• % Fuel Film Coolant (of W_f)	18.6	18.8	20.5	17.2			6% entrainment factor
• Taw Max, °F	2400	2400	2400	2400			2400°F maximum
• % Fuel Film Cool (of total flow)	4.5	4.5	5.0	3.8			
• 25 Lb Thrusters							Concept is simil to Lo_2/RP-1 igni which has duct f cooling. Core MR is 20:1 to reduc Twg. Selected ov all MR is at max Isp.
• Thrust, lbs							
• Pc, psia							
• MR_{TCA}							
• MR_{core}							
• $\dot{W}_{ox}$, lb/sec							
• $\dot{W}_f$, lb/seo							
• $\dot{W}_{ffc}$, lb/sec							
• %Fuel Film Coolant							
• Taw Max, °F							

PROPELLANTS	LO_2/C_3H_8		LO_2/CH_4	LO_2/NH_3	COMMENTS
Pc/F	400/10K	800/6K	400/10K	400/10K	
• Pumps					
• $\dot{W}_{ox}$, lbm/sec	20.9	12.1	22.0	16.9	
• $\dot{W}_f$, lbm/sec	7.9	4.3	6.4	14.0	
• $NPSP_{ox}$, psia	20.3	20.3	41.3	34.3	
• $NPSP_f$, psia	20.3	20.3	20.3	20.3	
• P_{iox}, psia	35.0	35.0	56.0	49.0	
• P_{if}, psia	35.0	35.0	35.0	35.0	
• P_{Dox}, psia	535	1020	535	533	
• P_{Df}, psia	980	1196	980	614	
• T_{sox}, °R	162.7	162.7	162.7	162.7	
• T_{s_f}, °R	416.2	416.2	217	432	
• Spec. Spd_{ox}	2920	1740	4920	4360	
• Spec SPd_f	1315	1140	1130	3150	
• Suct.Spec. Spd_{ox}	30K	30K	30K	30K	
• Suct.Spec Spd_f	30K	23K	36K	36K	
• No. of $Stages_{ox}$	1	1	1	1	
• No. of $Stages_f$	1	1	1	1	

TABLE : TPA AND GGA ANALYSIS - OME CONCEPTS

PROPELLANTS	LO_2/C_3H_8		LO_2/CH_4	LO_2/NH_3	COMMENTS
Pc/F	400/10K	800/6K	400/10K	400/10K	
• Imp. D_{ox}, in	1.48	1.46	1.00	0.96	
• Imp. D_f, in	1.60	1.40	1.80	1.30	
• η_{ox}, %	61.9	59.6	56.4	55.4	
• η_f, %	59.5	56.6	59.0	59.0	
• Turbines					
• Pin, psia	390	790	390	390	
• Pout, psia	39	79	39	39	
• Pr	10	10	10	10	
• $\dot{W}_{GG_{ox}}$, lbm/sec	0.22	0.24	0.15	0.14	for Ox TPA
• $\dot{W}_{GG_f}$, lbm/sec	0.26	0.19	0.23	0.24	for Fuel TPA
• T_{in}, °R	2000	2000	2000	2000	
• Tout, °R (ox, fuel)	1682/1603	1646/1628	1565/1579	1539/1515	
• ΔT, °R	318/397	354/372	435/421	461/485	
• Spec. Spd_{ox}	8	7.5	9.4	9.7	
• Spec Spd_f	19	10	17.0	13.3	
• No. of $Stages_{ox}$	1	1	1	1	
• No. of $Stages_f$	1	1	1	1	
• Tip D_{ox}, in	5.1	5.4	4.7	4.7	
• Tip D_f, in	4.1	3.9	3.9	4.4	
• η_f, %	69	65	62	68	
o η_{ox} %	55	62	64	64	

TABLE TPA AND GGA ANALYSIS - OME CONCEPTS

PROPELLANTS	LO_2/C_3H_8		LO_2/CH_4	LO_2/NH_3	COMMENTS
Pc/F	400/10K	800/6K	400/10K	400/10K	
• Gas Generator					
• Pc_{GG}, psia	400	800	400	400	
• $\dot{W}_{GG_{ox}}$, lbm/sec	0.22	0.24	0.15	0.14	
• $\dot{W}_{GG_f}$, lbm/sec	0.26	0.19	0.23	0.24	
• MR	0.36	0.36	1.2	0.57	
• Cp, BTU/lbm-°F	0.64	0.64	0.78	0.59	
• γ	1.18	1.18	1.23	1.25	
• MW	20	20	13.5	16.6	
• Tc, °R	2000	2000	2000	2000	
• Additional Data					
Pumps:					
• Oxid Flow, GPM	132	76	138	106	
• Fuel Flow, GPM	134	53	109	165	
• Oxid Speed, RPM	45,070	59,200	72,600	72,600	
• Fuel Speed, RPM	84,500	87,250	87,250	77,500	
• Impeller Tip Spd.					
• Oxid, ft/sec	291	377	317	304	
• Fuel, ft/sec	590	533	686	440	
• Shaft Power					
• Oxid, HP	64	76	71	54	
• Fuel, HP	94	66	106	95.4	
Turbine:					
• Blade Tip Sped (u)					
• Oxid, ft/sec	1,000	1400	1500	1500	
• Fuel, ft/sec	1,500	1500	1500	1500	
• Ratio u/spouting velocity (u/v)					
• Oxid	0.23	0.32	0.29	0.32	
• Fuel	0.34	0.34	0.29	0.32	

TABLE : PERFORMANCE ANALYSIS - OME CONCEPTS

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4		LO_2/NH_3	COMMENTS
Pc/F	100/6K	150/6K	400/10K	800/6K	150/6K	400/10K	400/10K	
• Engine Fv, lbf	6000	6000	10,068	6,061	6000	10,062	10,054	
• TCA Fv, lbf	6000	6000	10,000	6,000	6000	10,000	10,000	
• Engine MR	2.75	1.79	2.69	2.81	3.40	3.44	1.21	
• TCA MR	2.75	1.79	2.80	3.00	3.40	3.50	1.22	
• Core MR	2.75	2.75	2.80	3.00	3.40	3.50	1.40	Max ODK Isp MR
• Film Barrier MR	-		-	-	-	-		
• Turbine Ex. Fv, lbf	-	-	68	61	-	62	48	
• TCA $\dot{W}_{Tot}$, lbm/sec	17.80	18.37	28.15	16.01	17.33	27.88	30.41	
• TCA $\dot{W}_{ox}$, lbm/sec	13.05	11.79	20.74	12.01	13.39	21.68	16.71	
• TCA $\dot{W}_f$, lbm/sec	4.75	6.58	7.41	4.00	3.94	6.20	13.70	
• $\dot{W}_{turb}$, lbm/sec	-	-	.48	.43	-	.38	.38	
• % Fuel Film Coolar (of fuel flow)	0	35	0	0	0	0	13	
• Eng $\dot{W}_{ox}$, lbm/sec	13.05	11.79	20.87	12.12	13.39	21.89	16.85	
• Eng $\dot{W}_f$, lbm/sec	4.75	6.58	7.76	4.32	3.94	6.37	13.94	
• Eng Isp, Sec	337.0	326.7	351.7	368.7	346.2	356.0	326.5	
• TCA Isp, sec	337.0	326.7	355.2	374.7	346.2	358.7	328.8	
• Core Isp (ODK), sec	350.7		371.6	394.1	360.8	375.9		
• ISP_{turb}, sec	-	-	141.8	141.8	-	164.3	141.2	
• Ae/At	46	67	115	404	69	115	111	
• D_t, in.	6.34	5.26	4.02	2.14	5.12	4.00	4.14	
• D_e, in	43	43	43	43	43	43	43	
• % Fuel Film Coolan (of total flow)	0	12.5	0	0	0	0	5.8	
• Engine Total Flow Rate, lbm/sec	17.80	18.37	28.63	16.44	17.33	28.26	30.79	

TABLE : PERFORMANCE ANALYSIS - RCE CONCEPTS

PROPELLANTS	LO_2/C_3H_8			LO_2/CH_4			COMMENTS
Pc	100	150	300	150			
• 870 lbf Thrusters							
• TCA MR	2.24	2.23	2.19	2.48			Max ODK Isp MR
• Core MR	2.75	2.75	2.75	3.00			
• TCA $\dot{W}_{ox}$, lbm/sec	2.02	1.97	1.87	1.97			
• TCA $\dot{W}_f$, lbm/sec	.90	.88	.86	.80			
• % $\dot{W}_{ffc}$ (of fuel flow)	18.6	18.8	20.5	17.2			
• TCA Isp, Sec	297.9	305.6	318.8	313.9			
• Core Isp(ODK), sec							
• Ae/At	18	27	56	27			
• D_t, in	2.52	2.04	1.42	2.02			
• D_{ex}, in	10.6	10.6	10.6	10.6			
• 25 lbf Thrusters							
• TCA MR	2.75	2.75	2.75	3.00			
• Core MR	20	20	20	20			
• TCA $\dot{W}_{ox}$, lbm/sec	.082	.080	.077	.080			
• TCA $\dot{W}_f$, lbm/sec	.030	.029	.028	.026			
• % $\dot{W}_{ffc}$ (of fuel flow)	23	23	23	21			
• TCA Isp, sec	223.4	229.2	239.1	235.4			
• Core Isp(ODK), sec	-	-	-	-			
• Ae/At	18	27	56	27			
• D_t, in	.42	.35	.24	.35			
• D_{ex}, in	1.80	1.80	1.80	1.80			

*Film cooling as % of total flow

TABLE WEIGHT (LBM) - OME CONCEPTS

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PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4		LO_2/NH_3	COMMENTS
Pc/F	100/6K	150/6K	400/10K	800/6K	150/6K	400/10K	400/10K	
● TCA (each)								
. Injector	19.6	14.0	15.7	4.5	13.2	15.6	16.8	
. Chamber	77.6	52.9	65.3	36.8	58.2	57.4	47.9	
. Nozzle	79.5	78.0	79.5	85.5	77.1	79.1	81.6	
. Controls+TCA Instr.	18.3	18.3	19.3	19.3	18.3	19.3	19.3	Does not include TCA valve
	195.0	163.2	179.8	146.1	166.8	171.4	165.6	
● Thrust Structure Assy.	21.3	21.3	30.5	21.3	21.3	30.5	30.5	Scaled from OME
● Gimbal System	52.9	52.9	74.4	52.9	52.9	74.4	74.4	Scaled from OME
● Plumbing**	21.7	17.2	18.2	16.9	17.2	18.2	18.2	
● TPA (ox/fuel)*	-	-	10.2/6.5	11.8/5.5	-	7.6/6.5	7.6/6.9	
● Boost Pump(ox/fuel)	-	-	-	-	-	-	-	
● GGA (ox/fuel)*	-	-	2.4/2.6	2.3/2.3	-	2.4/2.6	2.4/2.4	
● Controls & Instr.								
. TCA Valve	21.0	21.0	21.0	21.0	21.0	21.0	21.0	
. Pneumatic Pack	7.6	7.6	7.6	7.6	7.6	7.6	7.6	
. Purge Valves	-	-	-	-	-	-	-	Not required
. Instr.*	2.6	2.6	9.6	9.6	2.6	9.6	9.6	
. GGA Valves*	-	-	7.8	7.8	-	7.8	7.8	
. TPA Controller*	-	-	22.8	22.8	-	22.8	22.8	
. Boost Pump* Circuit Valves	-	-	-	-	-	-	-	
	31.2	31.2	68.8	68.8	31.2	68.8	68.8	
Total, lbm	322.1	285.8	393.4	327.9	289.4	382.4	376.8	

*For two TPA's double these weights

**Plumbing weights are for TCA only. They do not include: Purge lines, GGA lines, or turbine exhaust/duct lines. These weights for pump-fed OME point designs previously supplied are: 2.6#, 2.0#, and 10.0# respectively.

TABLE : WEIGHT (LBM) - RCE CONCEPTS

PROPELLANTS	LO_2/C_3H_8			LO_2/CH_4			COMMENTS
Pc	100	150	300	150			
• 870-lbf Thruster							
• TCA (each)							
. Valves	2.5	3.4	2.5	5.3			
. Injector	6.4	5.3	3.9	5.3			
. Chamber/Nozzle	4.8	4.8	4.8	4.8			
. Insulation + Misc.	9.4	9.4	9.4	9.4			
	23.1	22.9	20.6	24.8			
• Propellant Conditioning							
. Heat Exchanger (ox/fuel)		26.3/-		26.3/26.5			
. GGA(ox/fuel)		5.5/-		5.5/4.5			
		31.8		62.8			
• Controls & Instr.							
. Pressure Reg. (ox/fuel)		6.2/2.0		6.2/6.2			
. Accumulator Valves (ox/fuel)		7.6/3.8		7.3/7.3			
. TPA GGA Valves		7.8		7.6			
. Prop.Cond GGA valves		3.3		10.9			
. Main Propellant valves(ox/fuel)		4.3/3.8		4.2/4.9			
. Instr.		14.4		19.2			
. TPA Controller		36.0		36.0			
		89.2		109.8			

TABLE WEIGHT (LBM) - RCE CONCEPTS (cont.)

PROPELLANTS	LOX/C_3H_8			LO_2/CH_4			COMMENTS
Pc	100	150	300	150			
• 25-lbf Thruster							
TCA (each)	W_B (~8.0 lbm)	W_B+0.5	W_B	W_B+1.0			W_B is notation for the basic 25-lbf thruster weight which is 5-10 lbm Deviations shown reflect valve weight differences

TABLE : ENVELOPE/SIZE - OME CONCEPTS

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PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4		LO_2/NH_3	COMMENTS
Pc/F	100/6K	150/6K	400/10K	800/6K	150/6K	400/10K	400/10K	
• TCA (each)	-	-	-	-	-	-	-	Same as existing TCA (approx 77" x 46")
. Length, in.								
. Nozzle Dia.in								
• TPA (ox/fuel)								
. Length, in.	-	-	6	6	-	6	6	
. Diameter, in.	-	-	8	8	-	8	8	
• GGA(ox/fuel)								
. Length, in.	-	-	10	10	-	10	10	
. Diameter, in	-	-	4	4	-	4	4	

TABLE : ENVELOPE/SIZE - RCE CONCEPTS

PROPELLANTS	LOX/C_3H_8			LO_2/CH_4			COMMENTS
Pc	100	150	300	150			
• 870 Lbf Thruster	-	-	-	-			Same as existing TCA's(approx. 19" x 11")
• 25 Lbf Thruster	-	-	-	-			Same as existing TCA's(approx. 11" x 6")
• Heat Exchangers (ox/fuel)							
. Length, in	-	20		20			
. Diameter, in	-	12	-	12			
• GGA's (ox/fuel)							
. Length, in	-	11.0/-	-	11.0/11.0			
. Diameter, in.	-	5.1/-	-	5.1/4.3			

TABLE : HEAT EXCHANGER AND GGA ANALYSES

PROPELLANTS	LOX/C_3H_8			LOX/CH_4			COMMENTS
Pc	100	150	300	150			
				OX	FUEL		
• $\dot{W}_C$, lbm/sec	21	21		21	6		
• T_{C_i}, °R		162		162	200		
• T_{C_o}, °R		310		310	380		
• P_{C_i}, psia		900		900	900		
• P_{C_o}, psia		800		800	800		Judgment
• $\dot{W}_H$, lbm/sec		3.4		2.9	1.6		
• T_{H_i}, °R		2000		2000	2000		Fuel rich GGA
• T_{H_o}, °R		800		800	800		Judgment
• P_{H_i}, psia		600		600	600		"
• P_{H_o}, psia		300		300	300		"
• ΔQ_C, Btu/sec		2,184		2,184	1,200		
• % η		80		80	80		Assumption
• $\dot{W}_H/\dot{W}_C$		0.16		0.14	0.27		

APPENDIX III

ADDED SCOPE OME POINT DESIGN
DATA DUMP

3.0 PRESSURE SCHEDULE

3.1 OME Concept

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4		
Pc/F	100/6K	400/6K	400/6K	400/6K	100/6K	400/6K	800/10K
• Plenum Pc, psia	100	400	400	400	100	400	800
• Face Pc, psia	103	412	412	412	103	412	824
• ΔP_{inj}, psi* (ox/fuel)	35/17	95	95	95	35/17	95	160
• ΔP_{TCV}, psi	5	20	20	20	5	20	20
• ΔP_{lines}, psi	-	8	8	8	-	8	16
• ΔP_{cj}, psi	36**	445***	445***	445***	25**	545***	288
• $\Sigma\Delta$ P, psi	40/58	123/568	123/568	123/568	49/47	123/668	484/196
• Interface or pump discharge pressure (ox/fuel) psia	143/161	535/980	535/980	535/980	143/150	535/1080	1308/1020
• $\Delta P_{inj}/Pc$	.35/.17	0.24	0.24	0.24	.35/.17	0.24	0.20
• $\Delta P_{cj}/Pc$	0.36	1.11	1.11	1.11	0.25	1.36	0.42
• Design Point Definition	Ni Chamber	Ni Chamber	Ni Cham. Flat Cg	CRES 304L Chamber	Ni Chamber	Ni Chamber	Zr-Cu Chamber Ox Regen.

Note: Pressure schedule is for the nominal operating point.

*Pump-fed minimum ΔP_{inj} = 0.2 Pc

Pressure-fed ΔP_{inj} =

- Liquid Injection: minimum $\Delta P_{inj}/Pc$ = 0.2
- Gas injection: minimum $\Delta P_{inj}/Pc$ = 0.15

  Note: min. $\Delta P_{inj}/Pc$ occurs at low Pc and high MR corner of operating box

**Includes 15 psi for ΔP across heat exchanger (nozzle)

***Supercritical fuel regen cooling. ΔPc.j. includes the regen. jacket and a throttling valve.

4.0 CHAMBER THERMAL ANALYSIS

4.1 OME Concepts

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4			COMMENTS
Pc/F	100/6K	400/6K	400/6K	400/6K	100/6K	400/6K	800/10K	
• Thrust, lbs	4500	5400	5400	5400	4500	5400	9000	
• Pc, psia	75	360	360	360	75	360	720	
• MR_{TCA}	3.30	2.85	2.94	2.21	3.60	3.68	3.68	
• MR_{Core}	3.30	2.94	2.94	2.94	3.60	3.68	3.68	
• $\dot{W}ox$, lb/sec	10.23	11.16	11.17	11.18	10.33	11.64	19.05	
• $\dot{W}_f$, lb/sec	3.10	3.92	3.80	5.05	2.87	3.16	5.18	
• No.of Regen Passes	1	1	1	1	1	1	1	
• ΔPc.j., psi	14	66	33	56	7	81	217	
• Pc.j.-in.psia	131	[illegible]	800	800	131	900	1260	
• Pc.j.out, psia	117	734	767	744	124	819	1043	
• Tc.j.-in.°F	90	-44	-44	-44	-180	-259	-286	
• Tc.j.-out,* °F	221	103	101	8	746	-79	-173	
• ΔTc.j.,°F	131	147	145	52	926	180	113	
• Regen ϵ	6.23	6.73	6.73	6.73	6.23	11.33	28.4	
• $\dot{W}_{fc}$, lb/sec	-	0.12	-	1.26	-	-	-	
• %Fuel Film Coolant	-	3	--	25	-	-	-	
• T_{ffc-in}, °F	-	145	-	145	-	-	-	
• Twg max., °F	636	776	659	693	1000	1000	660	

4.0 CHAMBER THERMAL ANALYSIS (cont.)

4.1 OME Concepts (cont.)

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4			COMMENTS
Pc/F	100/6K	400/6K	400/6K	400/6K	100/6K	400/6K	800/10K	
• % Regen Flow	95	100	100	100	20	100	100	
• Tw_1 max	600	582	569	431	989	880	609	
• h_g^*, BTU/in^2-sec-°F	.000371	.00261	.00151	.00448	.000368	.00276	.00462	
• h_l^*, BTU/in^2-sec-°F	.00189	.0280	.0222	.0238	.00184	.0362	.0995	
• Tr*, °F	5346	5179	5740	1687	5346	5740	5815	
• Q/A_g max, BTU/in^2 sec	1.78	11.57	7.72	4.85	1.65	14.21	25.55	
• Q/A_l max, BTU/in^2 sec	0.78	11.19	7.14	4.78	0.87	13.56	15.19	
• $\frac{Q/A}{Q/A_{Bo}}$ max	-	-	-	-	-	-	-	
• Q Total, BTU/sec	186	339	324	143	345	735	1001	
• V_c max, ft/sec	246	104	75	131	473	121	226	
• V_c (Mach No)^{max}	.297	.030	.021	.036	.250	.127	.113	
• No of channels	287	158	158	158	292	157	145	
• Min Ch Depth, in	.147	.030	.040	.030	.062	.032	.044	
• Limiting Criteria	Mach No.	TwL	TwL	Cycle Life	Mach No.	Cycle Life	Cycle Life	
• Description	Case #6 Nickel Ch Vapor Cooled	Case #1 Nickel Ch	Case #2 Nickel Ch Flat Cg	Case #3 CRES 304L Ch	Case #7 Vapor Cooled, Nickel Ch	Case #4 Nickel Ch	Case #5 ZR-cu Ch, LO_2 Cooled	
*at max gas-side flux **at max coolant-side flux								

*@ max. flux.

NOTE: Limit Tw_1 for C_3H_8 was reduced from 800°F to 600°F for compatibility reasons.

6.0 PERFORMANCE ANALYSIS

6.1 OME Concepts

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4			COMMENTS
Pc/F	100/6K	400/6K	400/6K	400/6K	100/6K	400/6K	800/10K	
• Engine Fv, lbf	6000	6054	6054	6055	6000	6051	10,104	
• TCA Fv, lbf	6000	6000	6000	6000	6000	6000	10,000	
• Engine MR	2.75	2.58	2.66	2.02	3.00	3.41	3.39	
• TCA MR	2.75	2.72	2.80	2.10	3.00	3.50	3.50	
• Core MR	2.75	2.80	2.80	2.80	3.00	3.50	3.50	Max ODK Isp MR
• Film Barrier MR								
• Turbine Ex. Fv, lbf	-	54	54	55	-	51	104	
• TCA $\dot{W}_{Tot}$, lbm/sec	17.80	16.68	16.66	17.25	17.48	16.46	27.03	
• TCA $\dot{W}_{ox}$, lbm/sec	13.05	12.20	12.28	11.69	13.11	12.80	21.02	
• TCA $\dot{W}_f$, lbm/sec	4.75	4.48	4.38	5.56	4.37	3.66	6.01	
• $\dot{W}_{turb}$, lbm/sec	-	~.38	~.38	~.39	-	~.31	~.63	
• % Fuel Film Coolant (of fuel flow)	Ø	3	Ø	25	Ø	Ø	Ø	
• Eng $\dot{W}_{ox}$, lbm/sec	13.05	12.30	12.38	11.79	13.11	12.97	21.36	
• Eng $\dot{W}_f$, lbm/sec	4.75	4.76	4.66	5.85	4.37	3.80	6.30	
• Eng Isp, Sec	337.0	354.9	355.3	343.3	343.2	360.8	365.3	
• TCA Isp, sec	337.0	359.7	360.1	347.9	343.2	364.5	369.9	
• Core Isp (ODK), sec	350.7	377.0	377.1	375.5	356.4	382.2	388.6	
• ISP_{turb}, sec	-	141.8	141.8	141.8	-	164.3	164.3	
• Ae/At	46	193	194	186	46	196	236	
• D_t, in.	6.34	3.10	3.08	3.16	6.36	3.08	2.80	
• D_e, in	43	43	43	43	43	43	43	
• % Fuel Film Coolant (of total flow)	Ø	0.8	Ø	7.9	Ø	Ø	Ø	
• Engine Total Flow Rate, lbm/sec	17.80	17.06	17.04	17.64	17.48	16.77	27.66	
• Description	Nickel Chamber	Nickel Chamber	Ni Cham Flat Cg	CRES 304L Chamber	Nickel Chamber	Nickel Chamber	Zr-Cu Ch. Ox Regen.	

7.0 WEIGHT (LBM)

7.1 OME Concepts

PROPELLANTS	LO_2/C_3H_8				LO_2/CH_4			
Pc/F	100/6K	400/6K	400/6K	400/6K	100/6K	400/6K	800/10K	
● TCA (each)								
. Injector	19.6	8.5	8.5	8.5	19.6	8.5	8.5	
. Chamber	73.0	45.0	45.0	42.0	73.0	45.0	65.1	
. Nozzle	80.0	80.0	80.0	80.0	80.0	80.0	80.0	
. Controls+TCA Instr.	18.3	19.3	19.3	19.3	18.3	19.3	19.3	Does not include TCA valve
	190.9	152.8	152.8	149.8	190.9	152.8	172.9	
● Thrust Structure Assy.	21.3	21.3	21.3	21.3	21.3	21.3	30.5	Scaled from OME
● Gimbal System	52.9	52.9	52.9	52.9	52.9	52.9	74.4	Scaled from OME
● Plumbing*	21.7	15.0	15.0	15.0	21.7	15.0	16.9	
● TPA (ox/fuel)*	-	7.7/5.2	7.7/5.2	7.7/5.2	-	6.5/5.0	7.6/6.0	See Note
● Boost Pump(ox/fu	-	-	-	-	-	-	-	
● GGA (ox/fuel)*	-	~4.6	~4.6	~4.6	-	~4.6	~5.0	
● Controls & Instr								
. TCA Valve	21.0	21.0	21.0	21.0	21.0	21.0	21.0	
. Pneumatic Pa	7.6	7.6	7.6	7.6	7.6	7.6	7.6	
. Purge Valves	-	-	-	-	-	-	-	Not required
. Instru*	2.6	9.6	9.6	9.6	2.6	9.6	9.6	
. GGA Valves*	-	7.8	7.8	7.8	-	7.8	7.8	
. TPA Controlle	-	22.8	22.8	22.8	-	22.8	22.8	
. Boost Pump*	-	-	-	-	-	-	-	
Circuit Valve								
	31.2	68.8	68.8	68.8	31.2	68.8	68.8	
Total, lbm	318.0	328.3	328.3	325.3	318.0	326.9	382.1	
● Design Point Definition	Ni Chamber	Ni Chamber	Ni Cham, Flat Cg	CRES 304L Chamber	Ni Chamber	Ni Chamber	Zr-Cu Chamber Ox Regen	

*For two TPA's double these weights

NOTE: TPA weights are estimtes for designs based on NPSP = 20 psia. Use attached graph to scale weights to other NPSP values.

APPENDIX IV

GROUNDRULES AND EVALUATION CRITERIA

GROUND RULES AND EVALUATION CRITERIA FOR OME ANALYSIS

The following ground rules and evaluation criteria are proposed for the analysis of the OME systems.

		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
A. Propellants, Propellant Storage Condition, Engine Operating Point			
1. Propellants, engine thrust and chamber pressure, propellant storage temperature.	1. Per Statement of Work and reiterated on Table I.	Specified by NASA-JSC.	--
2. Nominal operating point mixture ratio.	2. Optimum kinetic performance of core gases, additional film cooling as required.	Specified by NASA-JSC. Provides maximum Isp design.	--
3. Propellant storage pressure	3. Pressure-fed: as required by system pressure schedule; pump-fed: determined by turbopump requirements.	Specified by NASA-JSC.	--
B. Applicable Requirements of Procurement Specification (MC621-0009)			
1. Para 3.1.2 Interface Definition Para 3.1.2.1 Envelope	1. Will not exceed nozzle skirt envelope; forward envelope to accommodate required components.	Specified by NASA-JSC. Makes basic operational requirements the same as the existing OMS engine.	--
2. Para. 3.2.1.1 Life Requirements Para. 3.2.1.1.1 Operating Life Para. 3.2.1.1.2 Useful Life Para. 3.2.1.1.3 Shelf Life	2. 100 missions ten years, .00 missions ten years		
3. Para 3.2.1.8.1 Mission Profile	3. Thirty-day mission.		

			SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
4.	Para. 3.2.1.8.2 Engine Start Capability	4. 500 starts, times scatter factor of 4 per Para. 3.2.2.3.3 Fatigue		
5.	Para. 3.2.1.8.3 Minimum Engine Firing Time	5. 2 seconds		
6.	Para. 3.2.1.8.4 Minimum Engine Off Time	6. 240 seconds		
7.	Para. 3.2.1.8.5 Maximum Duration Firing	7. No design limit		
8.	Para. 3.2.1.8.6 Engine Duration Capability.	8. 15 hours		
9.	Para. 3.2.2.11.2 Chamber Cooling	9. 500 °F maximum exterior, no refractories or ablatives.		
10.	Para. 3.2.2.12.1 Nozzle Cooling	10. Radiation cooling		
11.	Para 3.2.2.14 Propellant Valve Assembly	11. 500 dry/5000 wet cycles		
C.	<u>Component Design Considerations</u>			
1.	Engine components:	1. Per Statement of Work:	Specified by NASA-JSC.	--
	Thrust chamber assembly:	chamber, nozzle, injector, igniter, valves		
	Turbopump Assembly:	Main pump, turbine, gas generator, exhaust nozzle; boost pump & drive in some concepts		
	Other:	Gimbal system; instrumentation; controls.		

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		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
2. Engine cycles	2. (a) Pressure-fed (b) Gas generator-driven turbopump: will consider only fuel-rich gas generator; wil investigate pumps separate and combined on single shaft; for separate pumps will consider turbines in parallel. (c) Hydraulically-driven boost pump (d) Expander cycle (methane only)	Specified by NASA-JSC.	--
3. Chamber Cooling	3. Up-pass and two-pass regenerative, with film cooling as required.	Specified by NASA-JSC	--
D. **Analytical Bases** Thrust Chamber Cooling Analysis			
1. General approach	1. Design for most severe O/F; indicated O/F variation is ± 20% pressure-fed, ± 5% pump-fed.	Specified by NASA-JSC	Wider operating Pc and MR ranges have a general tendency to drive the engine to a conservative desig at the nominal operating point. Hence, nominal engine performance and weight is compromized.
2. Gas-side heat transfer	2. Turbulent correlation; Cg profile rom LOX/RP-1 Program (NAS 3-21030).	Data is judged to be the best available.	If the RP-1 profile is not the same for C_3H_8, CH_4 or NH_3 the calculated film coolant flow rates and ΔPc.j. are not valid.
3. Coolant heat transfer	3. Propane, methane convection and boiling: empirical correlations developed previously in NAS 9-15958; ammonia: convection: Sieder-Tate; boiling: JPL/RMI data. C_2H_5OH: Convection: Hines	Data is judged to be the best available	Propane and methane correlations should be excellent. Validity of NH_3 coefficients should be good. NOTE: The thermal characteristics for the pressure-fed LO_2/C_3H_8 and LO_2/NH_3 point designs were controlled by the burnout heat flux limitation.
4. Burnout Safety Factor (BOSF)	4. BOSF = 1.3 for propane, methane-based on NAS 9-15958 data scatter, BOSF= 1.67 for ammonia-based on JPL/RMI data scatter. C_2H_5OH BOSF=1.3*	Data is judged to be the best available.	
5. Gaseous coolant Mach number	5. 0.3 maximum.	Good design practice	--
6. Coolant-side wall temperature.	6. Consistent with cycle life, creep, and carbon deposition considerations: 800°F max. for propane, based on coking: 1000°F max. for methane in Zr-Cu, based on cycle life and creep; 800°F max. for ammonia in 304L, based on cycle life. C_2H_5OH 800°F max. based on coking.	Data is judged to be the best available.	Will make results conservative or optimistic depending on the validity of these values. NOTE: The thermal characteristics for the pump-fed point designs for LO_2/C_3H_8 and LO_2/CH_4 were controlled by the coolant-side wall temperature limitation.

*AF Report No: FTD-MT-24-146-68
Translation Soviet Data, 1964

		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
7. Gas-side wall temperature.	7. Consistent with cycle life and creep consideration: 1000°F for Zr-Cu, 800°F for 304L.	Data is judged to be the best available.	Will make results conservative or optimistic depending on the validity of these values. NOTE: The thermal characteristics for the pump-fed point design for LO_2/NH_3 was controlled by the gas-side wall temperature limitation.
8. Coking on gas-side	8. Consider in overall bulk temperature rise but not local heat balance; factor: 0.42 for propane. 0.765 for methane. 0.5 for C_2H_5OH	This is a conservative approach that considers gas-side coking of walls is not immediate following initial engine start. Factors are based on results of previous studies.	Will make results conservative or optimistic depending on the validity of these values. Using higher factors would require additional film cooling for the pressure-fed LO_2/C_3H_8 point design; i.e., these coking factors favor LO_2/C_3H_8 concepts.
9. Channel configuration	9. Consistent with life requirements and fabricability; maximum depth-to-width ratio: 5 Minimum channel width: 0.0325 inch Minimum depth: 0.030 inch Minimum land width: .0325 inch	Based on current manufacturing limitations/capabilities without special processing.	Increased capability would decrease film coolant requirements of pressure-fed cases.
Performance Analysis			
1. Chamber contour	1. Not to exceed current exit diameter envelope; barrel and throat area scaled in proportion to thrust/chamber pressure.	Eliminates potential for redesigning Orbiter and/or Pod structure/aerodynamic surfaces.	Limits nozzle area ratio and biases results in favor of pump-fed concepts.
2. Performance prediction	2. Simplified JANNAF methodology	Provides good results for time and resources expended.	Performance comparisons are valid
3. Energy release efficiency	3. Same as current OME	State-of-the-art, precludes costly development programs.	--
4. Injector pressure drop	4. Consistent with chugging considerations.	Engine design parameter that must be considered.	--
5. Igniter design	5. Based on recent ALRC LOX/HC technology.	Data is judged to be the best available	No impact as the affect of igniters on system evaluation is essentially nil.

						SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
Turbopump Assembly Analysis							
1. Pump Design Limits	Oxygen	Ammonia	Methane	Propane	Ethyl Alcohol		
• Suction specific speed - maximum design value RPM GPM1/2/FT3/4	30,000	36,000	37,500	30,000	30,000	ALRC estimated state-of-the-art criteria derived from LO_2/LH_2 experience.	Based on results completed at this time, the TPA criteria has not resulted in any significant compromize in TPA designs.
• Thermodynamic suppression head - minimum design feet of liquid	3.5	6.6	7.1	3.5	[illegible]		
• Suction inlet velocity maximum design coefficient $2g\ NPSH/C_m^2$ (typical design criteria)	2.3	1.67	1.58	2.29	2.3		
• Minimum impeller diameter - inch		0.7					
• Minimum impeller exit blade width - inch		0.03					
• Minimum inducer inlet blade flow coefficient - fluid axial velocity/blade tangential velocity		0.06					
2. Turbine Design Limits							
• Minimum blade height-to-hub ratio blade height/hub diameter		0.05					
• Minimum blade height - inch		0.15					
• Minimum rotor hub-to-blade tip ratio		0.6					
• Minimum rotor diameter - inch		1.0					

	SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
3. Turbine Stress Design Criteria	ALRC estimated state-of-the-art criteria derived from LO_2/LH_2 experience.	Based on results completed at this time, the TPA criteria has not resulted in any significant compromise in TPA designs.

- Mean blade root centrifugal stress — Based on modified Goodman diagram relating alternating stress to steady-state stress.
- Design allowable blade root stress — Assumes 20% overspeed, 5% centrifugal bending, 10% gas bending stress

4. Bearing Design Limits

	Oxygen	Ammonia	Methane	Propane	Ethyl Alcohol
• Bearing DN limit - ID innerface in mm times speed in RPM x 10^{-6}	1.5	1.6	1.9	1.6	1.5
• Bearing size	15 to 40 mm				

5. Face Seal Design Limits

	Oxygen	Ammonia	Methane	Propane
• Face contact seal maximum fluid pressure differential times rubbing velocity (psig x ft/sec)	50,000	120,000	120,000	120,000

6. Turbopump efficiency, size, weight

- Efficiency — Empirical correlations
- Size, weight — Empirical correlations

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		Selection Rationale	Impact on Engine Design
Instrumentation, Valves, and Controls Analysis			
1. Design approach	1. Use current OME and Titan engine as basis.	Precludes costly and lengthy development programs	--
Engine Weight and Volume Analysis			
1. Component:			
Thrust chamber assembly	Scaled from current OME (will include redundant features)	Best data available	--
Turbopump assembly	Calculated separately	--	
Gimbal system	scaled from current OME	Best data available	--

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GROUND RULES AND EVALUATION CRITERIA FOR RCE ANALYSIS

The following ground rules and evaluation criteria are proposed for the analysis of the RCE systems.

		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
A. Propellants, Propellant Storage Condition, Engine Operating Point			
1. Propellants, engine thrust and chamber pressure, propellant storage temperature	1. Per Statement of Work and reiterated in Table II.	Specified by NASA-JSC	—
2. Nominal operating point mixture ratio.	2. Optimum kinetic performance of core gases, additional film cooling.	Specified by NASA-JSC. Provides maximum Isp design.	—
3. Propellant storage pressure.	3. Pressure-fed: as required by system pressure schedule; pump-fed (to accumulators): determined by turbopump requirements.	Specified by NASA-JSC	—
B. Applicable Requirements of Procurement Specification (MC647-0028, 29)			
1. Para. 3.1.1 Interface Definition Para. 3.1.1.1 Envelope	1. Chamber and nozzle envelope maintained, turbopump assembly envelope to be determined; accumulator configuration not addressed.	Specified by NASA-JSC. Makes basic operational requirements the same as the existing OMS engine.	—
2. Para 3.2.1.1 Life Reqmts. Para 3.2.1.1.1 Operating Life Para 3.2.1.1.3 Shelf Life	2. 50,000 cycles (vernier: 500,000) 20,000 seconds duration (vernier: 125,000) 10 years.		
3. Para 3.2.1.2.1 Nominal Duty Cycle	3. Steady-state duration 800 sec (vernier: 125) pulse widths from 0.080 to 0.960 seconds; minimum off-time 0.080 seconds; mission duty cycle less than 350 seconds or 1000 starts; mission duration		

		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
	7 days with design objective of 30 days.	Specified by NASA-JSC. Makes basic operational requirements the same as the existing OMS engine.	--
4. Para 3.2.1.4.6 Impulse Bit	4. 50 lbs-sec for 0.080 sec firing (vernier: 1.3 lb · sec)		--
5. Para 3.2.1.4.7.3 External Surfaces	5. Maximum external temperatures indicated, typically 500°F at forward end, 2200°F at aft end.		
C. <u>Component Design Considerations</u>			
1. Engine components: Thrust chamber assembly Turbopump assembly	1. Per Statement of Work: chamber, nozzle, igniter, valves pump and turbine; flow corresponding to 6000-10,000 lbf thrust with discharge pressures of 400-1200 psia	Specified by NASA-JSC	--
Propellant conditioning assembly	Heat exchanger, gas generator, exhaust nozzle		
Other	Low thrust (25 lbF) vernier engine		
2. Engine cycles	2. (a) Pressure-fed (b) Gas-generator-driven turbopump with accumulator tanks	Specified by NASA-JSC	
3. Chamber Cooling	3. Ducted and non-ducted film cooling	Specified by NASA-JSC	All 870-lbf thrusters were assumed to be non-ducted film cooling because for the existing RCE chamber configuration ducted film cooling advantage was not considered significant because of the short barrel length.

		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
D. Analytical Bases			
Thrust Chamber Cooling Analysis			
1. General approach	1. Design for most severe point in operating point box; indicated Pc and O/F variation ± 40% about nominal	Specified by NASA-JSC	Large variation in operating parameters results in relatively large TCA inlet pressures.
2. Film coolant temperature profile	2. Based on mixing model correlated with empirical data	Best approach available.	--
3. Wall temperature	3. Consistent with current engine: 2400°F maximum, to meet cycle life and exterior temperature requirements.	Specified by NASA-JSC	--
Performance Analysis			
1. Chamber contour	1. Not to exceed current exit diameter envelope; barrel and throat area scaled in proportion to 1/chamber pressure.	Precludes re-packaging/locating of RCE.	Limits nozzle area ratio
2. Performance prediction	2. Simplified JANNAF methodology	Provides good results for time and resources expended	Performance comparisons are valid
3. Energy release efficiency	3. Same as current RCE	State-of-the art, precludes costly development programs.	--
4. Injector pressure drop	4. Consistent with chugging considerations .	Engine design parameter that must be considered	--

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		SELECTION RATIONALE	IMPACT ON ENGINE DESIGN
Turbopump Assembly Analysis			
1. Design approach	1. Utilize applicable OME configuration	Specified by NASA-JSC	--
Heat Exchanger Analysis			
1. Size, pressure drops	1. Standardized charts for tube in shell	Existing data is adequate since heat exchanger characteristics are minimal and will not influence selection of RCS main issues.	--
. Weight	2. Simple calculation		
Vernier Thruster Analysis (vernier may double as igniter)			
1. Film coolant requirements	1. Calculate for one design, scale other in proportion to RCE requirements; base on recent ALRC LOX/HC igniter technology	A ROM approach to the 25-lbf thrusters is warranted since its characteristics are minimal and will not influence selection of RCS main issues.	--
2. Size, weight	2. Simple calculation		
Engine Weight and Volume Analysis			
1. Component:			
Thrust chamber assembly	Use current aft RCE as basis; adjust chamber and insulation weights for new contours.	Best data available	--
Turbopump assembly	Base on OME configuration.		

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APPENDIX V

CHAMBER THERMAL ANALYSIS REPORTS

NO. 9751:0738 AND 9751:0746

Aerojet
Liquid Rocket
Company

ROCKET
ENGINEERING

THERMODYNAMIC ANALYSIS REPORT	NUMBER: 9751: 0738 DATE: 15 October 1981
SUBJECT: REGENERATIVE AND FILM COOLING ANALYSES OF LOX/HYDROCARBON PROPELLANTS IN SHUTTLE OME/RCE APPLICATIONS	PAGE 1 OF 58 NO. OF TABLES 11 NO. OF FIGURES 10
ADDITIONAL INFORMATION AND WORK NOTES INCLUDED IN MICROFILM FILE CDN______	

PREPARED FOR: S. Hart

ABSTRACT

Thermal analyses were performed to assess the cooling capabilities of propane, methane and ammonia for shuttle OME/RCE applications. Six baseline and seven parametric design points for the OME and six baseline and four parametric design points for the RCE were evaluated. Engine sizing was based on nominal design point operating parameters while chamber cooling considered the most severe thermal condition for a specified variation in these parameters from the nominal.

For the OMS application, propane as a supercritical fluid and as a vapor was an acceptable regen-only coolant. As a subcritical liquid, however, film cooling (30-35% of total fuel flow) was required. Methane, analyzed as a gas-phase coolant at inlet pressures from 225 to 1200 psia, was an acceptable coolant

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	APPROVED BY: J. L. PIEPER, MANAGER J. L. Pieper

Abstract (cont.)

for the four cases studied; fuel bypass was required at low pressure. Ammonia as a regen coolant required film cooling (11-30%) for the three subcritical pressure cases analyzed.

The three fuels were analyzed as a vapor film coolant for the 870 lbf RCE. The film cooling flow rates, expressed as a fraction of total fuel flow, were comparable: propane (18.6 - 20.5%), methane (17.2 - 19%), and ammonia (20.4%).

Each coolant was assessed as providing the required cooling capability for the OMS and RCS applications. Each possesses inherent disadvantages - propane is subject to wall coking and recent evidence suggests an adverse reaction between propane and hot copper, methane requires vaporization prior to cooling the high flux regions, and ammonia is burnout limited. Each coolant must be considered in context with system factors for a proper assessment of its relative ranking.

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I INTRODUCTION

The cooling capabilities of propane, methane and ammonia for rocket engines directed towards meeting Shuttle OME (Orbit Maneuvering Engine) and RCE (Reaction Control Engine) duty requirements have been investigated analytically under Contract NAS 9-15958 for Johnson Space Center (JSC). The primary objectives of the study from an engine thermal analysis standpoint were: (1) to determine the feasibility of regenerative ("regen") cooling or regenerative plus film cooling with these propellants as coolants, based on the engine size envelope and mission duty requirements for the current OMS engine and (2) to determine similarly the cooling requirements for film-cooled designs based on the current RCS engine. Liquid oxygen is the oxidizer in all cases.

II SUMMARY

A. Engine Concepts and Analysis Basis

Thermal analyses were performed to assess the coolant characteristics of propane, methane, and ammonia as a propellant in engine concepts meeting the engine size envelopes and mission duty requirements of current OMS and RCS designs. Nominal thrusts for the OME application were 6K and 10K lbf; nominal chamber pressures varied from 100 to 800 psia with low chamber pressure engines (below 400 psia) pressure fed and high chamber pressure engines pump fed. RCS engines were evaluated for a nominal thrust of 870 lbf and nominal chamber pressures varying from 100 to 300 psia.

Six baseline and seven parametric design point cases were evaluated for the OME application. Propellants were NBP oxygen and hydrocarbon fuels at the following inlet conditions:

Coolant	Coolant Inlet State(s)
Propane	NBP Liquid (-44°F) and Vapor (90°F)
Methane	NBP Liquid (-259°F) and Vapor (-160 to -190°F)
Ammonia	NBP Liquid (-28°F)

Regeneratively-cooled chambers for LO_2/C_3H_8 and LO_2/CH_4 were Zirconium-copper alloy while stainless steel (CRES 304L) was assumed for the LO_2/NH_3 propellant combination.

The off-design condition for each nominal design point which resulted in the most severe thermal environment was analyzed to provide a cooling design meeting thermal criteria. The MR and Pc off-design operating points so selected were based on variations in mixture ratio and chamber pressure. For pressure-fed cases $\Delta MR = \pm 20\%$, $\Delta Pc = \pm 25\%$; for pump-fed cases, $\Delta MR = \pm 5\%$, $\Delta Pc = \pm 10\%$.

Six baseline and four parametric RCS operating points were analyzed for film cooling requirements for adiabatic wall operation at 2400°F. The MR and Pc off-design operating points were based on a variation of $\pm$ 40% in both parameters. The film coolant was assumed to be injected as a saturated vapor at the engine chamber pressures.

For both engine systems, the current OME and RCE packaging envelopes were maintained.

B. Analysis Results

Thirteen off-nominal design points were analyzed for the OMS application. Of these, five could not be regeneratively cooled and required film cooling. Design point iterations were not performed as a general rule, i.e., once an acceptable cooling concept was achieved, that case was considered completed. Acceptability was defined as determining that all parameters were within criteria constraints and limits. It follows that optimization of channel design would result in benefits in coolant flow requirements, pressure drop, inlet pressure, etc.

The more significant findings may be summarized as follows:

OME	Case	Coolant	Nom. Pc/F	Coolant State(1)	Cooling Mode(2)	Regen ΔP,psia	F FFC
	1.1	C_3H_8	100/6K	L	R + FFC	17	30
	1.2		150/6K	L	R + FFC	76	35
	1.3		100/6K	V	R	5	-
	1.11		800/10K	SCF	R	90	-
	1.12		400/10K	SCF	R	10.5	-
	1.13		800/6K	SCF	R	133	-
	2.1	CH_4	100/6K	V	R	4.4	-
	2.2		150/6K	V	R	14.5	-
	2.11		800/10K	SCF	R	146	-
	2.12		400/10K	SCF	R	8	-
	3.1	NH_3	100/6K	L	R + FFC	7	11
	3.11		800/10K	L	R + FFC	140	30
	3.12		400/10K	L	R + FFC	63	13
RCE	11.1	C_3H_8	150/870	V	FFC	-	18.6
	11.2		100/870	V	FFC	-	18.6
	11.3		300/870	V	FFC	-	20.5
	11.11		250/870	V	FFC	-	19.6
	11.12		150/870	V	FFC	-	18.75

II Summary (cont.)

12.1	CH_4	150/870	V	FFC	-	17.2
12.11		250/870	G	FFC	-	19.0
12.12		150/870	G	FFC	-	17.25
13.1	NH_3	150/870	V	FFC	-	20.4
13.11		250/870	V	FFC	-	20.4

(1) L = Liquid, G = Gas, V = Vapor, SCF = Supercritical Fluid

(2) R = Regen, FFC = Fuel Film Cooling

As thermal considerations are but one facet of the factors contributing to assessment of an engine concept, only the broadest of conclusions are appropriate. These results show that each coolant studied can be shown to be a viable candidate for OME propulsion. Ammonia is flux limited, methane has essentially no subcooling and has a relatively low density while propane is subject to wall coking.

Similarly, the ten RCS analyses performed demonstrated a coolant capability at each design point. The film cooling requirements for ammonia were somewhat greater than those for propane and methane. However, these results must be judged in context with other factors for a realistic assessment.

III DISCUSSION OF RESULTS

A. Results of Analysis

1. Study Basis

Thermal analyses have been performed for cooling of thrust chamber designs for engine concepts derived from the current Shuttle Orbiter Maneuvering System (OMS) and Reaction Control System (RCS). This effort is an integral part of the Preliminary Engine System Characterization

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study whose objective is the development of second generation OMS and RCS engines using LOX/Hydrocarbon type propellants. Propane, methane, and ammonia were the candidate fuels evaluated. For the OME designs, nominal thrust levels studied were 6000 and 10,000 lbf at nominal chamber pressures ranging from 100 to 800 psia. For the RCE, the nominal thrust was 870 lbf with nominal chamber pressures of 100 to 300 psia.

For each engine system the analyses were performed in two consecutive phases: (1) baseline point designs in which major system concepts (i.e., pump-fed vs pressure fed) were studied for each propellant combination, and (2) parametric point designs in which the effect of changes in significant operational parameters, such as chamber pressure and coolant inlet temperature, were considered. In both phases, performance parameters generated for the nominal operating point were utilized in the off-design engine cooling analysis. Operating point information for these OMS and RCS design concepts is shown in Tables I and II respectively. The analytical methodology employed and the assumptions made are discussed in Section III.B.

2. OMS Concepts

A primary constraint for OME concepts is the packaging envelope for the current shuttle design. This diameter-length limitation resulted in a range of nozzle expansion ratios as throat dimensions varied. Engine thermal analyses were performed for the low Pc-high MR corner of the "operating box", as tabulated in Table I, as this off-design operating point represents in most cases the most severe conditions for engine cooling. Coolant flow rate, whether regenerative only or regenerative plus film cooling, is lower; the decrease in flux resulting from the lower chamber pressure is offset by the larger relative reduction in flow rate and thus cooling capability.

The design philosophy of the Statement of Work (SOW) required that cooling modes be examined in the following sequence with each design point analysis being terminated when a feasible cooling method satisfying specified design criteria had been demonstrated.

- Regenerative Cooling, single up-pass
- Regenerative Cooling, single up-pass augmented with film cooling
- Regenerative Cooling, dual up-down passes
- Regenerative Cooling, dual up-down passes, augmented with film cooling

a. LO_2/Propane

Three pressure-fed design points were studied (Cases 1.1, 1.2 and 1.3 of Table I) with chamber pressure and coolant inlet temperature as the primary variables. For Case 1.3, in which the propane coolant is a hot (90°F) vapor, a regen-only design was developed. The other cases required 30-35% fuel film cooling in series flow to augment regenerative cooling in order to satisfy thermal criteria. Pertinent data for these cases are given in Table III. Note vapor cooling (Case 1.3) requires fuel flow bypass.

Three pump-fed cases, with inlet coolant at -44°F, varied thrust level and Pc (Cases 1.11, 1.12, and 1.13 of Table I). Coolant pressure in the cooling jacket was maintained above the critical (minimum P/P_{crit} = 1.54) and coolant outlet pressures were above the minimum P_{out}, calculated as 123% of Pc. Regenerative cooling designs were obtained for all three cases. Data for these analyses are given in Table IV. The design is constrained by the coking temperature limiting the coolant-side wall temperature for Cases 1.11 and 1.13. For Case 1.12 the channel design process is controlled by the differential between the gas-side and closeout wall temperatures. Both liquid-cooled design points (Cases 1.1 and 1.2) were limited as regeneratively cooled designs by the burnout safety factor criterion; Case 1.3, in which approximately 35-40% of the propane vapor flows in the cooling jacket, is constrained by the coolant Mach number criterion.

b. LO_2/Methane

Four cases, as noted in Table I, were analyzed for the LO_2/CH_4 propellant combination with the results as summarized in Table V. The pressure-fed nominal design points considered the 6K lbf thrust engine at chamber pressures of 100 and 150 psia while the pump-fed cases were for the nominal 10K lbf engine at Pc's of 800 and 400 psia. The low pressure engines at subcritical pressures were analyzed assuming methane as a superheated vapor. This assumption of vapor cooling from an area ratio of approximately 6:1 is predicated upon the very limited subcritical temperature range for CH_4 and the avoidance of two-phase flow in the high heat flux portions of the cooling jacket. Satisfactory regenerative cooling designs for Cases 2.1 and 2.2 were achieved by reducing the fraction of fuel vapor through the cooling jacket to 28 and 40% respectively. Cooling channel pressure losses were nominal and the gas-side wall temperature was the design-constraining variable.

Two cases, nos. 2.11 and 2.12, were analyzed at supercritical pressures with NBP methane. The gas-side heat flux at Pc of 800 psia was almost double that for the Pc of 400 psia case while the coolant flow rate was reduced about 5%. In addition, the radiation-cooled nozzle extension was attached at a larger area ratio for Case 2.11, giving a longer coolant flow path. Although regenerative cooling designs were obtained in both analyses, these factors contributed to a much higher pressure drop, 146 vs 8 psi, for the higher Pc case. The channel depth for this case approached the limiting value of 0.030 in. and wall temperatures were higher.

c. LO_2/Ammonia

One low chamber pressure and two higher chamber pressure cases were analyzed for the LO_2/NH_3 propellant combination. The high critical pressure of ammonia (1635.8 psia) results in all analyses being performed with a subcritical fluid where the burnout heat flux is often the limiting design parameter. The reactivity between copper and ammonia mandated the use of corrosion-resistant stainless steel for the chamber, minimizing the effectivity of the channel lands as heat conducting fins.

The results of the analyses are given in Table VI. Film cooling was required in each case to augment regenerative cooling, with fuel film cooling fractions ranging from 0.11 to 0.30. The film coolant flow was in parallel with cooling jacket flow. The channel design of the pressure-fed engine (Case 3.1) was constrained by the limit imposed by the burnout safety factor (BOSF). The pump-fed designs (Cases 3.11 and 3.12) were limited by the maximum gas-side wall temperature of 800°F (no allowance for creep) but it will be noted that Q/Q_{BO},max is approaching the BOSF limit for both cases. In each analysis the effect of low thermal conductivity for stainless steel is evident in the poor flux transformation.

3. RCS Concepts

The RCE is a film-cooled design; the objective of the analysis is to determine the film cooling requirements for the specified adiabatic wall temperature of 2400°F. A study of the various engine cycles under consideration indicated the most severe thermal requirement results from those concepts which include an accumulator upstream of the engine. The contained fluid can be subjected to a heat gain sufficient to vaporize the liquid, resulting in the injection of gaseous fuel as the film coolant. In this analysis, the coolant fuel was assumed to be a saturated vapor at the design point chamber pressure. Oxygen to the combustor was assumed either as a saturated vapor or as a gas preheated to 90°F. The operating points are summarized in Table II.

The basic analytical tool utilized in these analyses was the HOCOOL program, described in detail in Reference (1), which includes the ALRC entrainment gas film cooling model. The film coolant is assumed to be injected onto the thrust chamber at the injector. The methodology, design assumptions and analysis criteria are discussed in Section III.B.

a. LO_2/Propane

Selected parameters for the 870 lbf thrust RCE using oxygen and propane are given in Tables VII and VIII. For the five off-design operating points prescribed, thrust is degraded to 520 lbf. The required gas film coolant flow rates, expressed as a percentage of total fuel flow, range from 18.6 to 20.5% for an adiabatic wall temperature of 2400°F. Cases 11.1 and 11.2 for Pc's of 90 and 60 psia respectively require the same percentage fuel film coolant (18.6%). Increasing the chamber pressure to 180 psia (Case 11.3) results in about a 2 percent increase in percent fuel film coolant (20.5%). Operation with oxygen at an inlet temperature of 90°F results in a slight percentage increase in film cooling.

b. LO_2/Methane

Three analyses were performed for the O_2/CH_4 propellant combination with the results given in Table IX. Fuel film cooling percentages of total fuel flow ranged from 17.2 to 19.0%, slightly less than for propane. The increase in Pc from 90 psia (Case 12.1) to 150 psia (Case 12.11) resulted in a 1.8% increment in the film cooling requirement. Heated oxygen (Case 12.12) had no effect on film cooling requirements.

c. LO_2/Ammonia

Two analyses were performed for gas cooling with saturated ammonia vapor. Increasing the chamber pressure from 90 to 150 psia resulted in no effect on percentage film coolant requirement and a slight decrease in the absolute flow rate. Data are presented in Table X.

B. Analysis Methods and Assumptions

1. Regenerative Cooling Model (OME)

Regenerative cooling analysis of single-phase hydrocarbon fuels at supercritical pressures in the cooling channels is performed using a cooling channel design program which has been developed specifically for parametric design studies. With this program it is economically feasible to generate a relatively large number of parametric design points for selected fuels and still obtain a detailed, multi-station analysis of a rectangular channel at each design point. This technique provides an analytic modeling and comparison of fuels at realistic regenerative cooled engine conditions.

The program scales the chamber geometry and the local gas-side heat transfer coefficients and coolant heat loads from reference input to other thrust and chamber pressures. The coolant channel geometry parameters are prescribed together with channel material(s) and their temperature-dependent properties and the coolant-side heat transfer correlation(s). Two-dimensional heat conduction around the coolant channel is included, providing a fin effectivity which for high conductivity metals results in a transformation of the gas-side heat flux to a lower valued coolant-side flux. At each station the program iterates to determine the channel depth required to satisfy both (1) a gas-side wall temperature limit, which can be specified as a function of closeout wall temperature consistent with cycle life and creep criteria, and (2) an optional coolant-side wall temperature limit which can be specified as the decomposition, or "coking", temperature for the coolant. The only simplifying assumption is that gas-side wall temperature differences between the reference input and the scaled cases have a negligible effect on gas-side heat transfer coefficients and heat loads. Normally gas-side wall temperature limits are well known in advance, so that local reference gas-side heat transfer analyses can be run at appropriate wall temperatures using a cooling channel analysis program.

This design program was developed for forced-convection single-phase cooling. A modified version includes subcooled nucleate boiling characteristics and the burnout heat flux as parameters. Engine geometry and gas-side heat transfer are scaled but channel geometry and dimensions are input. A satisfactory design is achieved when the burnout safety factor criterion, as discussed in Section III.B.3, is met as well as gas-side wall temperature and pressure drop criteria.

a. Gas-Side Boundary Layer Regimes

The relatively low thrust and chamber pressures for the nominal study range for the OME application (F = 6K and 10K lbf, $100 \leq Pc \leq 800$ psia) required that the possibility of boundary layer laminarization on the gas-side be evaluated. A recommended definition of gas-side boundary layer regimes at the throat for this study is shown in Figure 1. It is based on ALRC experience, e.g., Reference (2), and work at NASA-LeRC, Reference (3). Conversion of critical throat Reynolds numbers for a convergence angle of 30° to products of thrust and chamber pressure yields the following limits:

Propellants	Thrust x Pc, lbf^2/in^2	
	Laminarized	Turbulent
O_2/C_3H_8, O_2/CH_4	$\leq$ 135,000	$\geq$ 640,000
O_2/NH_3	$\leq$ 96,500	$\geq$ 456,500

As shown in Figure 1, the nominal parametric range of interest for this study lies in the fully turbulent regime. However, when the off-nominal design points are considered, the three high Pc cases are fully-turbulent while one low Pc case is in the reverse transition zone and the other is roughly borderline.

The channel design program automatically considers boundary layer flow regimes in computing gas-side heat transfer. At high Reynolds numbers where the flow is fully turbulent, heat transfer coefficients can be calculated from the standard pipe-flow correlations, or, as in this study as noted below, using any desired C_g profile as input. At moderate Reynolds numbers, acceleration effects are strong enough for the boundary layer to start a transition to laminar flow but the transition is not completed. The non-turbulent cases are treated by employing a weighted average of the turbulent and laminar coefficients.

b. Gas-Side C_g Profile

In conventional analysis of fully-turbulent flow, heat transfer coefficients would be determined using a standard C_g profile as shown by the solid line of Figure 2. Recent ALRC experimental work on Contract NAS 3-21030 (High Density Fuel Combustion and Cooling Investigation), however, indicates an axial variation in heat transfer in the cylindrical and convergent sections of the engine which is not predicted with the standard C_g profile. The throat C_g in this high pressure LO_2/hydrocarbon study was greater than expected while the near-injector value conversely was less than predicted. ALRC analyses have shown these data to be highly hardware dependent with the axial heat transfer variation strongly influenced by film cooling from the injector. Incorporation of these data thus provides a conservative prediction for chamber convergent section and throat heat fluxes.

To achieve a similar conservation, the conventional pipe flow C_g profile was modified to include these experimental results, resulting in the dashed line shown in Figure 2 for the low Pc engine.

A similar curve was derived for the high Pc engine with a contraction ratio of 3.3.

The effect of this modified C_g profile is (1) to increase local heat fluxes in the convergent section and the adjacent portion of the cylindrical section and (2) to decrease the heat fluxes in the near-injector region. In computing the heat load to propane and methane the gas-side coefficient calculated from this modified input C_g profile was then reduced respectively to 42 and 76.5% of the clean wall coefficients. These factors are based upon the carbon deposition data of Pratt and Whitney, Reference 4. No such flux reduction was assumed for ammonia.

Channel design fluxes are clean-wall fluxes to allow for clean engine startup, carbon layer spalling, and uncertainty in the correlation.

c. Attachment Area Ratio for a Radiation-Cooled Nozzle Extension

The minimum area ratio at which a radiation-cooled nozzle extension can be attached was calculated based on the lower temperature-duration curve of Figure 3 for oxidation protection of a columbium alloy (FS-85 or C103) by a silicide coating (SYLCOR R512). Predicted wall temperatures were based on the simple energy balance:

$$hg\ (T_{aw} - T_{wg}) = \sigma\epsilon\ (1 + f_i)\ (T_{wg})^4$$

in which;

ϵ = coating emissivity; typical value is 0.85

f_i = internal view factor to end planes from an axisymmetric view factor program

The Statement of Work (SOW) firing duration of 15 hours results in a conservative wall temperature estimate of 3215°R for attachment of the radiation-cooled nozzle extension.

d. Chamber Contour Selection

The design criteria (Section III-B.3) specify contraction ratios of 3.3 and 2.0 respectively for the pump-fed and pressure-fed engines. The basic non-dimensional chamber contours used in the study are shown in Figure 4. The convergent section contours were selected to minimize boundary layer turbulence within the limits of standard design practice. This goal dictates the use of a large convergence angle with a conical section of sufficient length. Therefore, a 30° convergence angle was selected along with a radius of curvature at the start of convergence large enough to prevent flow separation and local perturbations in the local heat transfer coefficient.

e. Nozzle Contour Selection

The non-dimensional contour data for a 400:1 area ratio, 90% bell nozzle are shown on Table XI. The symbol R on this table is the ratio of the nozzle radius to the throat radius and Z is the ratio of the nozzle axial length (measured from the throat) to the throat radius. Packaging considerations limit the maximum diameter of the nozzle. For an OME application, the largest expansion ratio for a nozzle exit diameter of 43 in. is shown as a function of the thrust/chamber pressure ratio as the upper curve of Figure 5; the lower curve is for an RCS application in which the nozzle exit diameter is limited to 10.6 in.

2. Regenerative Cooling Model Augmented with Film Cooling (OME)

For those design points for which a one-pass regenerative cooling design could not be developed, film cooling augmentation was mandated by the SOW. It was assumed that the coolant injected was a subcooled liquid, either at the coolant jacket discharge temperature for series flow (as for propane) or at the cooling jacket inlet temperature for parallel flow (as for ammonia).

The following design points required film cooling:

		Nominal		Off-Design	
Case	Coolant	Pc	$MR_{(core)}$	Pc	$MR_{(core)}$
1.1	C_3H_8	100	2.75	75	3.30
1.2	C_3H_8	150	2.75	112.5	3.30
3.1	NH_3	100	1.40	75	1.68
3.11	NH_3	800	1.40	720	1.47
3.12	NH_3	400	1.40	360	1.47

Each case is characterized by subcritical coolant jacket inlet pressures and thus the ALRC liquid film cooling model was selected as best representing the physical phenomena occurring in the chamber. This film cooling model has been correlated with considerable firing data with storable propellants during the past five years. In this model, part of the injected liquid is entrained directly from the liquid film into the mixing layer due to surface instability of the former; the remainder is vaporized. Entrainment of mainstream combustion products into the mixing layer occurs along its entire length. This liquid film analysis is a modification of the work reported in Reference (5). The entire model is described in detail in Reference (6).

ALRC data for liquid film cooling indicate that a typical thrust chamber entrainment fraction, defined as the ratio of the core mass flux being entrained into the mixing layer to the local axial mass velocity of the core gas, is 0.03. In this study, however, the entrainment fraction was conservatively estimated as 0.06. Flow acceleration effects are accounted for explicitly. These data also indicate that the entrainment fraction is greatly reduced downstream of the throat. As a result of this reduction coupled with kinetic energy recovery effects, adiabatic wall temperatures in the nozzle are generally lower than at the throat.

3. Film Cooling Model (RCE)

a. Coolant State

Cooling of the high-thrust (870 lbf) RCS engines was analyzed as coolant injection from the injector for an adiabatic wall temperature of 2400°F. A review of the various engine cycles proposed indicates that the most difficult cooling concept is that in which the injected coolant is assumed to be a saturated vapor, i.e., the coolant stored in accumulators has been environmentally heated prior to injection. In some cases it was assumed that the fuel, oxidizer or both were preheated to 90°F to assess the effect of added enthalpy on film cooling requirements.

b. Analytical Methodology

The basic analytical tools employed were (1) a computer program to determine thermodynamic and transport properties at the specified chamber pressures and mixture ratios and (2) the HOCOOL program, Reference (1), utilizing the gas film cooling adiabatic wall option. The end result is the film cooling fraction required to maintain the specified adiabatic throat wall temperature for an assumed entrainment rate of 6 percent.

The film coolant is assumed to be injected onto the thrust chamber walls through an annular slot and in a direction which is parallel to the core gas flow. In this study the coolant injection point was assumed at the injector.

This entrainment model is basically a two stream mixing model in which core gases at a specified injector mixture ratio are considered to be entrained by and to mix with the film coolant gases from the injector periphery. Mixing is assumed to occur in an annular mixing layer by entraining core gases. An entrainment fraction of 0.06 was assumed, based on ALRC test data and suitable conservatism.

The analysis logic consisted of the following sequential steps (See Reference (1) for the development of the analytical model).

(1) Engine free-stream (core) enthalpies were determined for the off-design chamber pressure and core mixture ratio, based on propellants as saturated vapors at the chamber pressure or as heated gases.

(2) Wall enthalpy as a function of wall mixture ratio was then determined for the engine throat pressure and an adiabatic wall temperature of 2400°F.

(3) The adiabatic wall enthalpy from the film cooling model was then determined as a function of wall mixture ratio and inlet propellant enthalpies, including an approximation for flow acceleration in terms of the conventional recovery factor.

(4) The wall mixture ratio and adiabatic wall enthalpy which satisfy the functions developed in steps (2) and (3) were then determined for the specified adiabatic wall temperature. The engine geometry, core flow data and free stream and film coolant properties were input to the gas film cooling model for an assumed range of film coolant fractions to obtain the wall mixture ratio at the throat as a function film cooling flow rate.

(5) A plot of wall mixture ratio for the range of assumed film cooling fractions permits the direct determination of the film cooling fraction characterized by the throat wall mixture ratio determined in Step (4).

c. Engine Geometry

The analysis maintained the overall dimensional envelope for the current RCS engine. Changes in chamber pressure were accommodated by varying the throat flow area and thus the contraction and expansion ratios. The radii of curvature were maintained at the normalized values for the current design; the convergence angle was 45 degrees.

4. Analysis Criteria

The design criteria upon which the study was based are summarized in this section.

a. Regenerative Channel Design Constraints

The basic coolant channel configuration for regeneratively-cooled designs with propane and methane as coolants is that of rectangular coolant passages milled in a zirconium-copper (aged at 1100°F) liner with an electroformed nickel closeout. This type of construction minimizes cooling problems at higher chamber pressures. For ammonia as a coolant, the Cu-NH_3 reaction made it necessary to employ the chemically-resistant stainless steel for the liner. These channel-walled chambers were considered to extend normally to the area ratio (ϵ_A) at which a radiation-cooled nozzle can be utilized since these attachment area ratios are relatively low. At low chamber pressures, however, ϵ_A approached the throat. For these cases the nozzle channel design was considered to extend to an area ratio of approximately 6:1 at which point fabricability methods for joining the nozzle extension to the cooled chambers are straightforward.

(1) Creep and Cycle Life Considerations

The service life of 500 thermal cycles times a safety factor of four and an accumulated run time of 15 hours limits (a) the maximum channel wall temperature to 1000°F, and (b) the temperature differential between the gas-side and closeout walls to the relationship shown in Figure 6 for Zirconium copper. The equivalent data for stainless steel is given in Figure 7. Note that creep allowance is not included for the steel, limiting the gas-side wall temperature to 800°F.

(2) Channel Dimensions and Geometry

Manufacturing considerations based on conventional machining technology suggest a minimum land width of 0.030 in. and a minimum channel width of 0.0325 in. for the high flux (throat) section. In order to minimize maldistribution of flow considering typical dimensional tolerances, a minimum allowable channel depth of 0.030 in. was selected. A maximum channel depth-to-width aspect ratio of 5:1 further constrained the design process. A typical channel layout is given in Figure 8. Ideally, each set of input parameters (e.g., inlet pressure, bulk temperature, coolant state, etc.) requires an iterative optimization of station channel and land dimensions to minimize pressure drop and provide the most effective cooling. Such an optimization was beyond the scope of this study.

The allowable gas side channel width-to-wall thickness ratio requirements for Zr-Cu and stainless steel are shown in Figures 9 and 10. The design program utilizes the input wall thickness unless the gas-side wall temperature requires a thicker wall to maintain the ℓ/t ratio at or below the value shown. The strength data shown for a differential pressure of 1000 psi were input to the channel design programs for the reference case hot wall aspect ratios at the cold and hot conditions. In this context "cold" refers to that point in the engine operating cycle where the channel fluid is at design inlet pressure, the engine gas-side pressure is zero, and the gas-side wall is at 500°F or lower. Similarly, "hot" refers to coolant at the local calculated pressure, the nozzle gas at the calculated gas pressure, and the gas-side wall is at the computed wall temperature - all at steady-state operating conditions. The actual pressure differentials across the hot wall at its calculated temperature were used in the program to determine the required wall thickness at each station.

(3) Coolant Flow Arrangement

A single-pass, upflow (i.e., coolant flow towards the injector) regenerative-cooling only configuration was the preferred cooling concept. If this arrangement could not be shown analytically as feasible, a single-pass upflow regenerative system augmented with film cooling, a two-pass, up-down flow regen system configuration, and a two-pass up-down flow regen system augmented with film cooling were to be examined in that sequence to achieve a workable design.

(4) Coolant State

Fluid at the coolant channel outlet is single phase.

(5) Coolant Velocity

The limiting channel fluid velocity for propane and ammonia as subcooled liquids at subcritical pressures was 200 ft/sec. For propane and methane at supercritical pressures or as a superheated vapor at subcritical pressures, the maximum acceptable fluid velocity was equivalent to a local Mach number of 0.3.

(6 Coolant Channel Outlet Pressure

Coolant outlet pressures were assumed equal to or greater than 1.23 Pc.

(7) Coolant Channel Inlet Pressure

No criteria were established for coolant inlet pressures. A combination of initial engineering estimates and run iterations were used to determine inlet pressures sufficient to result in the desired design point outlet pressures.

b. Coolant Properties

Earlier analyses under Task I of the contract had resulted in the establishment of property data files containing the necessary thermodynamic and transport properties over the pressure-temperature range of interest. Particulars are discussed in Reference (7).

c. Coolant Heat Transfer Correlations

Coolant heat transfer correlations for single-phase fluids in forced convection are semi-empirical and the usual caveat regarding their use beyond the range of supporting test data must be considered. The critical points, normal boiling points, and typical inlet temperatures are presented below for the coolants of interest:

	Propane	Methane	Ammonia
Critical Pressure, psia	615	667	1635.8
Critical Temperature, °F (°R)	206 (666)	-117 (343)	270 (730)
Normal Boiling Point, °F (°R)	-44 (416)	-259 (201)	-28 (432)
Typical Inlet Temperature, °F (°R)	-44 (416)	-259 (201)	-28 (432)

Correlations utilized were:

(1) Propane

Heat transfer to propane at supercritical and near critical pressures was studied under Task I of the contract with the results given in Reference (7). The correlation developed is:

$$Nu_b = 0.00545\, Re_b^{0.90}\, Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-0.11} \left(\frac{k_b}{k_w}\right)^{0.27} \left(\frac{\overline{C}_p}{C_{p_b}}\right)^{0.53} \left(\frac{\mu_b}{\mu_w}\right)^{0.23} \left(1 + \frac{2}{\ell/D}\right)$$

At subcritical pressures with propane as a vapor the film or Colburn equation, in which properties are evaluated at the film temperature, was utilized:

$$Nu_f = 0.027\, Re_f^{0.8}\, Pr_f^{0.4}$$

As a subcooled liquid, the forced convection heat transfer characteristics of propane were predicted by the Hines equation of Reference (8) as developed for water, RP-1 and dielhylcyclohexane (DECH):

$$Nu_b = 0.0055\, Re_b^{0.95}\, Pr_b^{0.4}$$

(2) Methane

Heat transfer to methane at supercritical pressures was assumed to be characterized by the above equation for its homolog propane. The film equation was employed when considering heat transfer to a vapor or low pressure superheated gas.

(3) Ammonia

The Hines equation was used to predict the forced convection heat transfer coefficient for ammonia.

d. Burnout and Burnout Safety Factor

Burnout heat flux data for fluids in nucleate boiling are conventionally correlated to fluid velocity and subcooling by a relationship of the following form:

$$\phi_{BO} = K_1 + K_2 V (T_{sat} - T_b) = K_1 + K_2 \quad V\Delta T_{sub}$$

where:

ϕ_{BO}	=	burnout heat flux, Btu/in^2-sec
V	=	local coolant velocity, ft/sec
T_{sat}	=	local coolant saturation temperature, °F
T_b	=	local coolant bulk temperature, °F
ΔT_{sub}	=	$T_{sat} - T_b$, °F
$\left.\begin{matrix} K_1 \\ K_2 \end{matrix}\right\}$	=	empirical constants

The wall superheat for nucleate boiling was conservatively estimated, i.e., the nucleate boiling mechanism was initiated when the local wall temperature exceeded the fluid saturation temperature at the local pressure by 25°F. Nucleate boiling heat transfer coefficients, defined as the slope of the curve relating boiling heat fluxes to wall temperature, ranging from 0.05 to 3 Btu/in^2-sec-°R were evaluated as no forced-flow boiling data were available.

(1) Propane

Burnout heat flux correlations for propane, based on the work reported in Reference (9), were derived at ALRC as:

a. Where $V\Delta T_{sub} \leq 1000$

$$\phi_{BO} = 0.3 + 0.0004\ V\Delta T_{sub}$$

b. Where $V\Delta T_{sub} > 1000$

$$\phi_{BO} = 0.58 + 0.00012\ V\Delta T_{sub}$$

These correlations are supported by test data to a $V\Delta T_{sub}$ value of about 3500 ft °R/sec, with a data spread of ± 25%.

(2) Methane

Methane has a very limited subcritical temperature range. Further, the proximity of the boiling points of methane and oxygen precludes any significant subcooling of methane. As the burnout correlation given above for propane is also applicable to methane, heat transfer by nucleate boiling of methane in the nozzle high heat flux region presents design problems unresolvable with current analytical methods. It was concluded that channel design studies were meaningful only for supercritical methane and for subcritical superheated vapor.

(3) Ammonia

The burnout heat flux correlation is of the form discussed for propane. Based on test data of JPL and RMI (References (10) and (11), the equations derived by ALRC are, with notation as employed earlier:

a. Where $V\Delta T_{sub} \leq$ 4000 ft °R/sec

$$\phi_{BO} = 2.15 + 0.00086\ V\Delta T_{sub}$$

b. Where $V\Delta T_{sub} >$ 4000

$$\phi_{BO} = 3.3 + 0.000587\ V\Delta T_{sub}$$

These equations are supported by test data to a $V\Delta T_{sub}$ value of about 14,000 ft °R/sec with a data spread of ± 30%.

The burnout safety factor (BOSF) is expressed in simplified form as:

$$BOSF = \frac{\phi_{BO}}{\phi_c} = \frac{1}{1 - \frac{S_{BO}}{100}}$$

where:

ϕ_c = local coolant heat flux, Btu/in^2 sec

S_{BO} = data scatter in ϕ_{BO} correlation

Burnout safety factors utilized for each fuel were:

Fuel	BOSF
Propane	1.30
Methane	1.30
Ammonia	1.67

Application of this criterion thus requires that at each chamber station

$$\frac{\phi_{BO}}{\phi_c} \geq \text{BOSF}$$

Since definitive information on ultimate heat flux limits at supercritical pressures was not found, this factor was not included as a criterion.

e. Coolant Side Coking

Thermal degradation, or coking, of carbon-containing coolants on hot metal walls has been studied for a number of compounds but is not yet well defined. Experimental tests have usually taken the form of flowing the coolant under specified conditions over a metal surface whose surface temperature is increased in discrete steps. The coking temperature is defined as that initial surface which exhibits a subsequent continuing rise while power is maintained at a constant flux level, indicating an increasing thermal resistance at the wall.

The coking temperature for propane, 800°F, is sufficiently low to become a design-constraining limit in some analyses. The equivalent methane limit of 1300°F is above the limit imposed for copper and thus does not constrain the design. Similarly, the cracking temperature of ammonia is sufficiently high that thermal decomposition was not considered.

f. Thrust Chamber Geometry Definition

(1) OME/RCE Design Guidelines

For the OME application, the design study maintained constant:

- Engine length of 72.0 in.
- Nozzle exit diameter of 43.09 in.
- Energy Release Efficiency (ERE) of 97.5%
- Diverging nozzle is an 85%-Bell, truncating at the area ratio required by the limiting nozzle exit diameter.

For the RCE, the current RCE envelope was maintained with throat diameter and associated convergent section radii adjusted for each case. Maintained constant were:

- Injector diameter of 3.90 in.
- Nozzle exit diameter of 10.586 in.
- TCA L' of 2.46 in.
- Divergent nozzle length of 12.0 in.
- Convergent nozzle angle of 45°
- ERE of 93%

(2) Chamber Length and Contraction Ratio

A detailed ALRC study for various propellants covering a broad spectrum of thrust levels and chamber pressures has been completed recently, Reference (12). This study related these parameters together with fluid injection state to give a predictive correlation of the form

$$L' = L_1 + K\,(F/Pc)^{0.23}$$

where:

L_1	=	parameter to provide additional length over that predicted by the second term
K	=	injection state constant
	=	7.079 for liquid/liquid
	=	4.178 for liquid/gas

The L' values for the baseline analyses of the study were not specified by the initial SOW design criteria. These baseline calculations thus utilized L' values approximated from the above formulation. Prior to performing the parametric analyses, however, L' criteria were specified and were utilized in the later studies. These L' values are significantly less than those computed earlier, based on 92% (current OME) of liquid/liquid injection K.

Pc, psia	Criteria L', in.	Calculated (Nominal) L', in.
100-150	11.0	16.7
400	9.0	13.6
800	7.0	11.6

The impact of the longer engine L' values in the baseline analyses is felt primarily in greater pressure drops and higher outlet bulk temperature. Channel design parameters at the higher flux region are not affected.

The contraction ratio for the OME-based design was selected as 3.3:1 for nominal chamber pressures of 400 psia or greater and 2:1 for lower chamber pressures.

(3) Nozzle Extension

The radiation-cooled nozzle extension extended to a maximum exit diameter of 43 in. For this nozzle, FS-85 columbium with a silicide coating was selected because of its high temperature capability. This material has been found to be suitable for use in nozzles where pressure levels are low.

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TABLE I

OPERATING POINTS FOR OMS CONCEPTS

			Coolant Inlet State		Nom. Operating Point			Oper. Pt. Degradation		Off Design Operating Point		
Case	Propellants	Anal Type(1)	Supply Method	Temp. (°F)	F (lbf)	Pc (psia)	MR(2)	Pc (%)	MR (%)	F (lbf)(3)	Pc (psia)	MR
1.1	LO_2/C_3H_8	B	Pressure-Fed	NBP (-44°)	6K	100	2.75	-25	+20	4400	75	3.30
1.2		P	Pressure-Fed	NBP (-44°)	6K	150	2.75	-25	+20	4500	112.5	3.30
1.3		P	Pressure-Fed	Ambient (90°)	6K	100	2.75	-25	+20	4500	75	3.30
1.11		B	Pump-Fed	NBP (-44°)	10K	800	3.00	-10	+5	9058	720	3.15
1.12		P	Pump-Fed	NBP (-44°)	10K	400	2.80	-10	+5	9000	360	2.94
1.13		P	Pump-Fed	NBP (-44°)	6K	800	3.00	-10	+5	5400	720	3.15
2.1	LO_2/CH_4	B	Pressure-Fed	Vapor (-190°)	6K	100	3.00	-25	+20	4350	75	3.60
2.2		P	Pressure-Fed	Vapor (-160°)	6K	150	3.40	-25	+20	4500	112.5	4.08
2.11		B	Pump-Fed	NBP (-259°)	10K	800	3.50	-10	+5	8867	720	3.68
2.12		P	Pump-Fed	NBP (-259°)	10K	400	3.50	-10	+5	9000	360	3.68
3.1	LO_2/NH_3	B	Pressure-Fed	NBP (-28°)	6K	100	1.40	-25	+20	4550	75	1.68
3.11		B	Pump-Fed	NBP (-28°)	10K	800	1.40	-10	+5	8905	720	1.47
3.12		P	Pump-Fed	NBP (-28°)	10K	400	1.40	-10	+5	9000	360	1.47

(1) Analysis Type: B - Baseline Point Design, P - Parametric Point Design

(2) Mixture ratio as determined by performance analysis without film cooling

(3) Off-design thrust based on nominal performance parameters

TABLE II

OPERATING POINTS FOR RCS (870 LBF) CONCEPTS

Code	Propellants	Anal.(1) Type	Inlet State	Nom. Operating Point F (lbf)	Pc (psia)	$MR_{(core)}$	Oper. Pt. Degradation Pc (%)	MR (%)	Off-Design Operating Point F (lbf)	Pc (psia)	$MR_{(core)}$
11.1	LO_2/C_3H_8	B	Both Satd Vapors at Pc	870	150	2.75	-40	+40	520	90	3.85
11.2		P	Both Satd Vapors at Pc	870	100	2.75	-40	+40	520	60	3.85
11.3		P	Both Satd Vapors at Pc	870	300	2.75	-40	+40	520	180	3.85
11.11		B	0 gas at 90°F F Satd. Vapor at Pc	870	250	2.75	-40	+40	520	150	3.85
11.12		P	0 gas at 90°F F Satd. Vapor at Pc	870	150	2.75	-40	+40	520	90	3.85
12.1	LO_2/CH_4	B	Both Satd Vapors at Pc	870	150	3.00	-40	+40	520	90	4.20
12.11		B	Both Gases at 90°F	870	250	3.00	-40	+40	520	150	4.41
12.12		P	Both Gases at 90°F	870	150	3.00	-40	+40	520	90	4.20
13.1	LO_2/NH_3	B	Both Satd Vapors at Pc	870	150	1.40	-40	+40	520	90	1.96
13.11		B	0 gas at 90°F F Satd. Vapor at Pc	870	250	1.40	-40	+40	520	150	1.96

(1) B - Baseline Point Design
P - Parametric Point Design

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TABLE III
PRESSURE-FED OME CONCEPTS
WITH LO_2/C_3H_8 PROPELLANTS

PROPELLANTS	LO_2/C_3H_8			COMMENTS
NOMINAL Pc/F	100/6K	150/6K	100/6K	
CASE NO.	1.1	1.2	1.3	
• Thrust, lbf	4400	4500	4500	
• Pc, psia	75	112.5	75	
• Throat Radius, in.	3.230	2.570	3.168	
• Contraction Ratio	2.0	2.0	2.0	
• L', in.	15.9	11.1	11.0	
• MR (TCA)	2.31	2.14	3.30	
• MR (Core)	3.30	3.30	3.30	
• $\dot{W}_{ox}$, lbm/sec	9.62	8.94	10.23	
• $\dot{W}_f$, lbm/sec	4.16	4.17	3.10	
• $\dot{W}_{c.j.}$, lbm/sec	4.16	4.17	1.24	
• % Fuel Regen	100	100	40	(e)
• No. Regen Passes	1	1	1	
• $\Delta P_{c.j.}$, psi	17	76	5	
• $P_{c.j.}$-in, psia	150	197	131	
• $P_{c.j.}$-out, psia	133	121	126	
• $\Delta T_{c.j.}$, °F	72	72	289	
• $T_{c.j.}$-in, °F	-44	-44	90	
• $T_{c.j.}$-out, °F	28	28	379	
• Regen ε	6.23	6.23	6.23	
• $\dot{W}_{ffc}$, lbm/sec	1.25	1.46	0	
• % Fuel Film Coolant	30	35	0	(a)
• T_{ffc-in}, °F	28	28	-	
• Coolant State	Liquid	Liquid	Vapor	

Sheet 1 of 2

TABLE III (cont.)

PROPELLANTS	LO_2/C_3H_8			COMMENTS
NOMINAL Pc/F	100/6K	150/6K	100/6K	
CASE NO.	1.1	1.2	1.3	
• T_{wg},max, °F • $T_{w\ell}$,max, °F	222 157	161 144	801 802	
• h_g,max,Btu/in^2-sec °F • h_ℓ,max,Btu/in^2-sec °F	.00133 .00654	.00193 .00955	.000371 .00138	(b) (c)
• Q/A_g max,Btu/in^2-sec • Q/A_ℓ max,Btu/in^2-sec	1.74 1.02	2.64 1.41	1.81 .40	
• Q_{total}, Btu/sec • Q/Q_{BO} - max, -	160 .77	170 .77	186 -	 (d)
• T_r, °F • Wall Thickness, in.	1510 .125	1515 .050	5346 .050	
• $V_{c.j.-max}$, ft/sec • $M_{c.j.-max}$, -	38.3 -	51.8 -	155 .18	 (e)
• No. Channels • Min. Chan. Depth, in.	350 .038	263 .030	323 .082	
• $\Delta P_{c.j.}/P_c$, - • Limiting Criterion	.23 BOSF	.67 BOSF	.07 Mach No.	 (d), (e)
• Coolant Correlation • Run Ident.	Hines -	Hines OMPRLC/ 6-30/5	Film OMPRVP/ 6-19/1	

• NOTES:

(a) Percentage of total fuel flow
(b) At maximum gas-side heat flux
(c) At maximum coolant-side heat flux
(d) Limit set by burnout safety factor (BOSF) = 0.77
(e) Case 1.3, Mach no. with 30% of fuel through cooling jacket > 0.3

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TABLE IV
PUMP-FED OME CONCEPTS
WITH LO_2/C_3H_8 PROPELLANTS

PROPELLANTS	LO_2/C_3H_8			COMMENTS
NOMINAL Pc/F	800/10K	400/10K	800/6K	
CASE NO.	1.11	1.12	1.13	
• Thrust, lbf • Pc, psia	9058 720	9000 360	5400 720	
• Throat Radius, in. • Contraction Ratio • L', in.	1.415 3.3 11.7	2.007 3.3 9.0	1.070 3.3 7.0	
• MR (TCA) • MR (Core)	3.15 3.15	2.94 2.94	3.15 3.15	
• $\dot{W}_{ox}$, lbm/sec • $\dot{W}_f$, lbm/sec	18.84 5.98	18.91 6.43	10.95 3.47	
• $\dot{W}_{c.j.}$, lbm/sec • % Fuel Regen	5.98 100	6.43 100	3.47 100	
• No. Regen Passes • $\Delta P_{c.j.}$, psi	1 90	1 10.5	1 133	
• $P_{c.j.}$-in, psia • $P_{c.j.}$-out, psia	1080 990	1200 1189	1080 947	
• $\Delta T_{c.j.}$, °F • $T_{c.j.}$-in, °F	230 -44	156 -44	246 -44	
• $T_{c.j.}$-out, °F • Regen ϵ	186 23.8	112 10.6	202 23.7	
• $\dot{W}_{ffc}$, lbm/sec • % Fuel Film Coolant	0 0	0 0	0 0	
• T_{ffc-in}, °F • Coolant State	- Supercritical Fluid	- Supercritical Fluid	- Supercritical Fluid	

Sheet 1 of 2

TABLE IV (cont.)

PROPELLANTS	LO_2/C_3H_8			COMMENTS
NOMINAL Pc/F	800/10K	400/10K	800/6K	
CASE NO.	1.11	1.12	1.13	
• T_{wg},max, °F	835	787	949	
• $T_{w\ell}$,max, °F	793	744	780	
• h_g,max,Btu/in^2-sec °F	.00493	.00262	.00477	(a)
• h_ℓ,max,Btu/in^2-sec °F	.0321	.0132	.0200	(b)
• Q/A_g max,Btu/in^2-sec	26.3	13.51	23.64	
• Q/A_ℓ max,Btu/in^2-sec	16.9	6.42	11.97	
• Q_{total}, Btu/sec	869	584	554	
• Q/Q_{BO} - max, -	-	-	-	
• T_r, °F	5960	5740	5909	
• Wall Thickness, in.	-	.050	.050	
• $V_{c.j.-max}$, ft/sec	136	47.2	107	
• $M_{c.j.-max}$, -	.044	-	.058	
• No. Channels	145	207	112	
• Min. Chan. Depth, in.	.040	.084	.030	
• $\Delta P_{c.j.}/P_c$, -	.12	.029	.185	
• Limiting Criterion	$T_{w\ell}$	T_{wg}-$T_{closeout}$	$T_{w\ell}$	
• Coolant Correlation	ALRC Propane	ALRC Propane	ALRC Propane	
• Run Ident.	OMPRSC/ 2-25/1b	OMPRSC/ 6-18/1	OMPRLC/ 7-15/1	

• NOTES:

(a) At maximum gas-side heat flux

(b) At maximum coolant-side heat flux

TABLE V

OME CONCEPTS WITH LO_2/CH_4 PROPELLANTS

FEED MODE	PRESSURE FED		PUMP FED	
NOMINAL Pc/F	100/6K	150/6K	800/10K	400/10K
CASE NO.	2.1	2.2	2.11	2.12
• Thrust, lbf	4500	4500	8867	9000
• Pc, psia	75	112.5	720	360
• Throat Radius, in.	3.230	2.567	1.400	2.001
• Contraction Ratio	2.0	2.0	3.3	3.3
• L', in.	16.1	11.0	11.6	9.0
• MR (TCA)	3.60	4.08	3.68	3.68
• MR (Core)	3.60	4.08	3.68	3.68
• $\dot{W}_{ox}$, lbm/sec	10.33	10.47	18.79	19.76
• $\dot{W}_f$, lbm/sec	2.87	2.56	5.11	5.37
• $\dot{W}_{c.j.}$, lbm/sec	.80	1.03	5.11	5.37
• % Fuel Regen	28	40	100	100
• No. Regen Passes	1	1	1	1
• $\Delta P_{c.j.}$, psi	4.4	14.5	146	8
• $P_{c.j.}$-in, psia	225	197	1080	1200
• $P_{c.j.}$-out, psia	220.6	182.5	934	1192
• $\Delta T_{c.j.}$, °F	760	801	247	-82
• $T_{c.j.}$-in, °F	-180	-160	-259	-259
• $T_{c.j.}$-out, °F	580	641	-12	177
• Regen ϵ	6.23	6.23	23.59	10.64
• $\dot{W}_{ffc}$, lbm/sec	0	0	0	0
• % Fuel Film Coolant	0	0	0	0
• T_{ffc-in}, °F	-	-	-	-
• Coolant State	Vapor	Vapor	Supercritical Fluid	Supercritical Fluid

TABLE V (cont.)

FEED MODE	PRESSURE-FED		PUMP-FED	
NOMINAL Pc/F	100/6K	150/6K	800/10K	400/10K
CASE NO.	2.1	2.2	2.11	2.12
• T_{wg},max, °F • $T_{w\ell}$,max, °F	1000 994	1000 995	890 852	782 739
• h_g,max,Btu/in²-sec F • h_ℓ,max,Btu/in²-sec °F	.000356 .000931	.000995 .00157	.00490 .0404	.00262 .0134
• Q/A_g max,Btu/in²-sec • Q/A_ℓ max,Btu/in²-sec	1.64 .37	4.55 1.03	26.63 14.57	14.14 5.70
• Q_{total}, Btu/sec • Q/Q_{BO} - max, -	389 -	526 -	1450 -	1059 -
• T_r, °F • Wall Thickness, in.	5452 .250	5346 .050	5840 .025	5740 .050
• $V_{c.j.-max}$, ft/sec • $M_{c.j.-max}$, -	120 .05	309 .177	242 .234	39.8 .025
• No. Channels • Min. Chan. Depth, in.	350 .121	263 .040	143 .036	206 .099
• $\Delta P_{c.j.}/P_c$, - • Limiting Criterion	.059 T_{wg}	.129 T_{wg}	.0203 $T_{wg}-T_{closeout}$	.022 $T_{wg}-T_{closeout}$
• Coolant Correlation • Run Ident.	Film OMMESC/ 5-18/3	Film OMMEVP/ 6-19	ALRC Propane OMMESC/ 4-21/1a	ALRC Propane OMMESC/ 6-18
• COMMENTS:				

TABLE VI

OME CONCEPTS WITH LO_2/NH_3 PROPELLANTS

FEED MODE	PRESSURE-FED	PUMP-FED		COMMENTS
NOMINAL Pc/F	100/6K	800/10K	400/10K	
CASE NO.	3.1	3.11	3.12	
• Thrust, lbf	4550	8905	9000	
• Pc, psia	75	720	360	
• Throat Radius, in.	3.230	1.403	2.041	
• Contraction Ratio	2.0	3.3	3.3	
• L', in.	15.9	11.6	9.0	
• MR (TCA)	1.495	1.13	1.28	
• MR (Core)	1.68	1.47	1.47	
• $\dot{W}_{ox}$, lbm/sec	8.66	14.52	15.96	
• $\dot{W}_f$, lbm/sec	5.79	9.88	10.86	
• $\dot{W}_{c.j.}$, lbm/sec	5.79	9.88	10.86	
• % Fuel Regen	100	100	100	
• No. Regen Passes	1	1	1	
• $\Delta P_{c.j.}$, psi	7	140	63	
• $P_{c.j.}$-in, psia	150	1080	630	
• $P_{c.j.}$-out, psia	143	940	567	
• $\Delta T_{c.j.}$, °F	62	58	40	
• $T_{c.j.}$-in, °F	-28	-28	-28	
• $T_{c.j.}$-out, °F	34	30	12	(a)
• Regen ϵ	6.2	30.93	6.73	
• $\dot{W}_{ffc}$, lbm/sec	.64	2.96	1.41	
• % Fuel Film Coolant	11	30	13	
• T_{ffc-in}, °F	-28	-28	-28	(a)
• Coolant State	Liquid	Liquid	Liquid	

Sheet 1 of 2

TABLE VI (cont.)

FEED MODE	PRESSURE-FED	PUMP-FED		COMMENTS
NOMINAL Pc/F	100/6K	800/10K	400/10K	
CASE NO.	3.1	3.11	3.12	
• T_{wg},max, °F • $T_{w\ell}$,max, °F	440 152	878 201	747 160	(d)
• h_g,max,Btu/in²-sec °F • h_ℓ,max,Btu/in²-sec °F	.00094 .0186	.00791 .111 or .0947	.00368 .0676	
• Q/A_g max,Btu/in²-sec • Q/A_ℓ max,Btu/in²-sec	2.20 2.32	6.90 7.53	5.47 6.25	
• Q_{total}, Btu/sec • Q/Q_{BO} - max, -	345 .59	617 .559	463 .454	(b)
• T_r, °F • Wall Thickness, in.	2773 .030	1728 .025	2233 .025	(c)
• $V_{c.j.-max}$, ft/sec • $M_{c.j.-max}$, -	28.1 -	180 -	111 -	
• No. Channels • Min. Chan. Depth, in.	328 .060	144 .041	208 .050	
• $\Delta P_{c.j.}/P_c$, - • Limiting Criterion	.09 BOSF	.194 T_{wg}	.175 T_{wg}	
• Coolant Correlation • Run Ident.	Hines -	Hines OMAMLC/ 3-6/2b	Hines OMAMLC/ 6-23	

• NOTES:

(a) Cooling jacket and film cooling flows in parallel for all cases.
(b) Limit set by burnout safety factor (BOSF) = 0.60
(c) Chamber walls are 347 stainless steel
(d) Temperature limit (no creep) is 800°F

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TABLE VII

RCE CONCEPTS WITH O_2/C_3H_8 PROPELLANTS

(Propellants as Saturated Vapors at Pc)

PROPELLANTS	O_2/C_3H_8			
NOMINAL Pc/F	150/870	100/870	300/870	
CASE NO.	11.1	11.2	11.3	
• Thrust, lb	520	520	520	
• Pc, psia	90	60	180	
• Throat Radius, in.	1.021	1.240	.700	
• Contraction Ratio	3.65	2.47	7.76	
• MR (TCA)	3.14	3.13	3.06	
• MR (Core)	3.85	3.85	3.85	
• $\dot{W}_{ox}$, lbm/sec	1.386	1.624	1.220	
• $\dot{W}_f$, lbm/sec	.442	.518	.399	
• W_{ffc}, lbm/sec	.082	.096	.082	
• T_{ffc} - in, °F	48	25	96	
• Coolant State	Satd Vapor	Satd Vapor	Satd Vapor	
• T_{aw}, max, °F	2400	2400	2400	
• % Fuel Film Coolant (of $\dot{W}_f$)	18.6	18.6	20.5	
• % Full Film Coolant (of Total Flow)	4.5	4.5	5.0	

NOTES:

TABLE VIII

RCE CONCEPTS WITH O_2/C_3H_8 PROPELLANTS

(Fuel as Saturated Vapor at Pc; Oxidizer as Gas at 90°F)

PROPELLANTS	O_2/C_3H_8			
NOMINAL Pc/F	250/870	150/870		
CASE NO.	11.11	11.12		
• Thrust, lb	520	520		
• Pc, psia	150	90		
• Throat Radius, in.	.780	1.000		
• Contraction Ratio	6.25	3.80		
• MR (TCA)	3.09	3.13		
• MR (Core)	3.85	3.85		
• $\dot{W}_{ox}$, lbm/sec	1.335	1.263		
• $\dot{W}_f$, lbm/sec	.431	.404		
• W_{ffc}, lbm/sec	.084	.076		
• T_{ffc} - in, °F	82	48		
• Coolant State	Sat'd Vapor	Sat'd Vapor		
• T_{aw}, max, °F	2400	2400		
• % Fuel Film Coolant (of $\dot{W}_f$)	19.6	18.75		
• % Full Film Coolant (of Total Flow)	4.8	4.5		

NOTES:

TABLE IX

RCE CONCEPTS WITH O_2/CH_4 PROPELLANTS

PROPELLANTS	O_2/CH_4			
NOMINAL Pc/F	150/870	250/870	150/870	
CASE NO.	12.1 [a]	12.11 [b]	12.12 [b]	
• Thrust, lb	870	870	870	
• Pc, psia	90	150	90	
• Throat Radius, in.	1.021	.780	1.010	
• Contraction Ratio	3.65	6.25	3.73	
• MR (TCA)	3.475	3.57	3.48	
• MR (Core)	4.20	4.41	4.20	
• $\dot{W}_{ox}$, lbm/sec	1.390	1.361	1.291	
• $\dot{W}_f$, lbm/sec	.400	.381	.371	
• $\dot{W}_{ffc}$, lbm/sec	.069	.073	.064	
• T_{ffc} - in, °F	-209	90	90	
• Coolant State	Sat'd Vapor	Gas	Gas	
• T_{aw}, max, °F	2400	2400	2400	
• % Fuel Film Coolant (of $\dot{W}_f$)	17.2	19.0	17.25	
• % Full Film Coolant (of Total Flow)	3.8	4.2	3.8	

NOTES:

(a) Both propellants inlet as saturated vapors at Pc

(b) Both propellants inlet as gases at 90°F

TABLE X

RCE CONCEPTS WITH O_2/NH_3 PROPELLANTS

PROPELLANTS	O_2/NH_3			
NOMINAL Pc/F	150/870	250/870		
CASE NO.	13.1	13.11		
• Thrust, lb • Pc, psia	870 90	870 150		
• Throat Radius, in. • Contraction Ratio	1.021 3.65	.780 6.25		
• MR (TCA) • MR (Core)	1.56 1.96	1.56 1.96		
• $\dot{W}_{ox}$, lbm/sec • $\dot{W}_f$, lbm/sec	1.123 .719	1.088 .697		
• $\dot{W}_{ffc}$, lbm/sec • T_{ffc} - in, °F	.147 50	.142 79		
• Coolant State • T_{aw}, max, °F	Sat'd Vapor 2400	Sat'd Vapor 2400		
• % Fuel Film Coolant (of $\dot{W}_f$)	20.4	20.4		
• % Full Film Coolant (of Total Flow)	8.0	8.0		

NOTES:

Table XI

Nozzle Contour (ε = 400:1)

EXPANSION COEFFICIENT = 1.20000
THROAT RADIUS = 1.25400
UPSTREAM WALL RADIUS OF CURVATURE = 1.00000
DOWNSTREAM WALL RADIUS OF CURVATURE = .34300
NOZZLE LENGTH = 79.39482
NOZZLE EXPANSION RATIO = 399.96822

WALL CONDITIONS

	R	Z	MACH	THETA	PRESSURE	RHO	TEMP	EPSILON	RHU*V
THROAT PT	1.000000	.000000	1.2033	.000	.44024+00	.50857+00	5619.32	1.0000	8.3451
TANGENT PT	1.074962	.214021	2.3386	38.606	.74040-01	.11427+00	4168.67	1.1555	3.1270
	1.078679	.218673	2.3331	38.642	.73713-01	.11384+00	4165.54	1.1635	3.1183
	1.121356	.271704	2.3629	38.978	.69834-01	.10882+00	4128.18	1.2574	3.0054
	1.168447	.324663	2.3923	39.285	.66184-01	.10406+00	4091.40	1.3559	2.8967
	1.208608	.377626	2.4218	39.556	.62716-01	.99499-01	4054.87	1.4593	2.7912
	1.252215	.430960	2.4494	39.703	.59619-01	.95386-01	4020.78	1.5680	2.6949
	1.297143	.484794	2.4766	39.942	.56767-01	.91486-01	3987.37	1.6826	2.6027
	1.342934	.539240	2.5040	40.097	.53914-01	.87717-01	3953.94	1.8034	2.5124
	1.389586	.594549	2.5306	40.211	.51323-01	.84190-01	3921.62	1.9309	2.4270
	1.437358	.650945	2.5568	40.298	.48891-01	.80852-01	3890.02	2.0659	2.3454
	1.486277	.708571	2.5834	40.367	.46538-01	.77596-01	3858.17	2.2090	2.2650
	1.536596	.767607	2.6097	40.406	.44314-01	.74494-01	3826.82	2.3611	2.1877
	1.588333	.828494	2.6358	40.422	.42219-01	.71534-01	3795.91	2.5228	2.1132
	1.641550	.891244	2.6624	40.422	.40159-01	.68625-01	3764.52	2.6957	2.0392
	1.697162	.956209	2.6889	40.394	.38217-01	.65869-01	3733.56	2.8804	1.9681
	1.754463	1.023607	2.7156	40.349	.36352-01	.63160-01	3702.56	3.0781	1.8985
	1.814200	1.093903	2.7431	40.284	.34526-01	.60505-01	3670.90	3.2913	1.8292
	1.876190	1.167255	2.7703	40.196	.32804-01	.57979-01	3639.71	3.5201	1.7627
	1.941043	1.244079	2.7984	40.092	.31115-01	.55480-01	3607.79	3.7676	1.6964
	2.008603	1.324750	2.8267	39.965	.29444-01	.53062-01	3575.77	4.0353	1.6316
	2.079612	1.409698	2.8558	39.820	.27923-01	.50696-01	3543.30	4.3256	1.5677
	2.154413	1.499387	2.8854	39.653	.26402-01	.48384-01	3510.37	4.6415	1.5047
	2.232827	1.594317	2.9157	39.466	.24932-01	.46128-01	3477.01	4.9855	1.4427
	2.315484	1.695058	2.9468	39.257	.23569-01	.43923-01	3443.12	5.3615	1.3816
	2.402781	1.802239	2.9787	39.027	.22129-01	.41764-01	3408.59	5.7734	1.3212
	2.494928	1.916506	3.0114	38.774	.20802-01	.39667-01	3373.64	6.2246	1.2621
	2.592729	2.038718	3.0452	38.498	.19512-01	.37605-01	3337.83	6.7222	1.2035
	2.696276	2.169639	3.0798	38.198	.18272-01	.35603-01	3301.50	7.2699	1.1461
	2.806275	2.310283	3.1155	37.873	.17076-01	.33651-01	3264.47	7.8752	1.0896
	2.923410	2.461789	3.1524	37.522	.15922-01	.31744-01	3226.61	8.5463	1.0340
	3.048285	2.625405	3.1905	37.144	.14811-01	.29887-01	3187.95	9.2920	.9794
	3.181551	2.802501	3.2299	36.738	.13745-01	.28083-01	3148.50	10.1223	.9258
	3.324009	2.994946	3.2706	36.303	.12723-01	.26332-01	3108.21	11.0498	.8734
	3.476787	3.204651	3.3129	35.837	.11745-01	.24631-01	3066.96	12.0000	.8220

Table XI (continued)

	3.641100	3.434196	3.3569	35.337	.10804-01	.22978-01	3024.66	13.2576	.7717
	3.818360	5.080703	3.4027	34.601	.99060-02	.21375-01	2981.24	14.5799	.7228
	4.010280	3.965900	3.4506	34.226	.90476-02	.19820-01	2936.54	16.0823	.6742
	4.773120	5.134808	3.6264	32.019	.64960-02	.15030-01	2778.78	22.7827	.5229
	5.158993	5.704248	3.7061	30.943	.55931-02	.13275-01	2710.33	26.6152	.4659
	5.561955	6.449190	3.7859	29.935	.48177-02	.11723-01	2643.75	30.9353	.4151
	5.856794	6.968772	3.8408	29.208	.43488-02	.10764-01	2599.01	34.3020	.3834
	6.409135	8.098173	3.9502	27.731	.35496-02	.90884-02	2512.53	41.8497	.3273
	6.896093	8.927160	4.0216	26.703	.31109-02	.81822-02	2457.89	47.5549	.2953
	7.254567	9.649966	4.0795	25.972	.27961-02	.74495-02	2414.57	52.6258	.2717
	7.761117	10.715476	4.1577	24.905	.24228-02	.66109-02	2357.57	60.2349	.2428
	8.318020	11.946631	4.2401	23.717	.20848-02	.58328-02	2299.27	69.1895	.2157
	8.864393	13.221904	4.3176	22.716	.18109-02	.51866-02	2245.92	78.6464	.1931
	9.304722	14.491012	4.3880	21.757	.15952-02	.46608-02	2198.96	88.0730	.1747
	9.835077	15.644436	4.4473	20.951	.14341-02	.42706-02	2160.29	96.7405	.1606
	10.251239	16.750708	4.5006	20.227	.13038-02	.39446-02	2126.25	105.0878	.1489
	10.615281	17.755404	4.5401	19.611	.12021-02	.36866-02	2097.68	112.6842	.1376
	10.946078	18.699800	4.5869	19.001	.11182-02	.34708-02	2072.53	119.8297	.1318
	11.552185	20.481044	4.6571	18.117	.98756-03	.31795-02	2030.05	132.9913	.1195
	11.907686	21.663908	4.7105	17.403	.89849-03	.29938-02	1998.51	143.7094	.1109
	12.666617	24.168477	4.7904	16.341	.78170-03	.25755-02	1952.48	160.9553	.0992
	13.409987	26.723749	4.8709	15.278	.67962-03	.22920-02	1907.47	179.8278	.0887
	13.771959	28.072815	4.9104	14.759	.63466-03	.21650-02	1885.83	189.6667	.0840
	14.119600	29.415893	4.9490	14.207	.59479-03	.20510-02	1865.55	199.3631	.0797
	14.448640	30.723466	4.9831	13.811	.56002-03	.19506-02	1846.92	208.7002	.0760
	14.759145	32.016191	5.0103	13.379	.52897-03	.18601-02	1829.44	217.8324	.0726
	15.350696	34.584076	5.0786	12.579	.47559-03	.17025-02	1797.30	235.6439	.0667
	15.878107	37.614649	5.1337	11.867	.43317-03	.15748-02	1769.53	252.1143	.0619
	16.344547	39.300351	5.1820	11.249	.39918-03	.14711-02	1745.59	267.1442	.0579
	16.755670	41.426358	5.2245	10.708	.37158-03	.13858-02	1724.87	280.7927	.0547
	17.119536	43.387750	5.2618	10.235	.34902-03	.13154-02	1706.96	293.0721	.0520
	17.437969	45.169905	5.2946	9.821	.33039-03	.12566-02	1691.42	304.0835	.0498
	17.996077	48.543856	5.3527	9.097	.30019-03	.11601-02	1664.61	323.8789	.0461
	18.413903	51.231324	5.3953	8.557	.27946-03	.10930-02	1644.88	339.0718	.0435
	19.037556	55.587415	5.4603	7.740	.25103-03	.99948-03	1615.73	362.4285	.0399
	19.643481	60.269658	5.5244	6.950	.22596-03	.91559-03	1587.65	385.8663	.0367
EXIT PT	19.999206	63.313251	5.5627	6.476	.21226-03	.86907-03	1571.18	399.9682	.0349

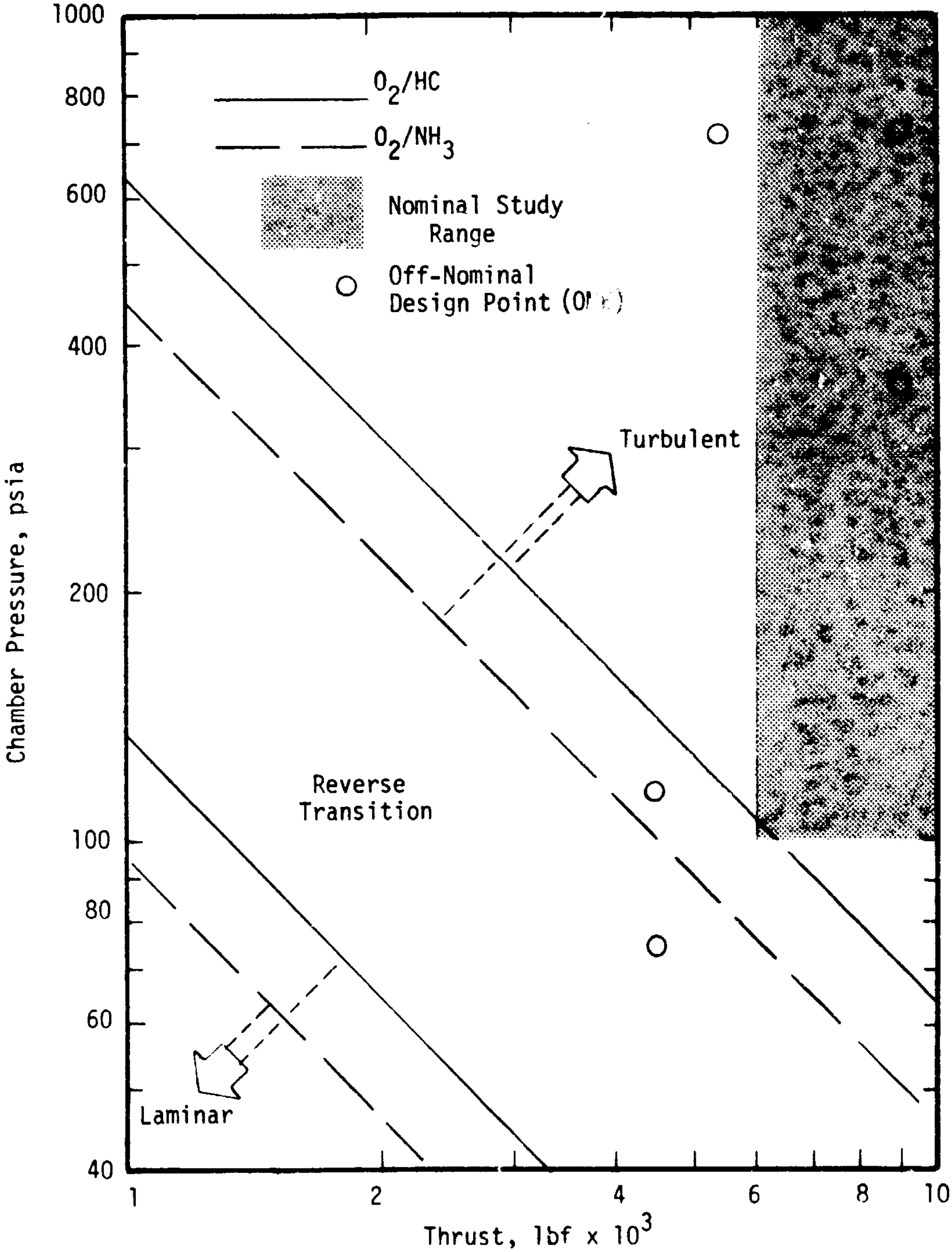

Figure 1 . Gas-Side Boundary Layer Flow Regimes

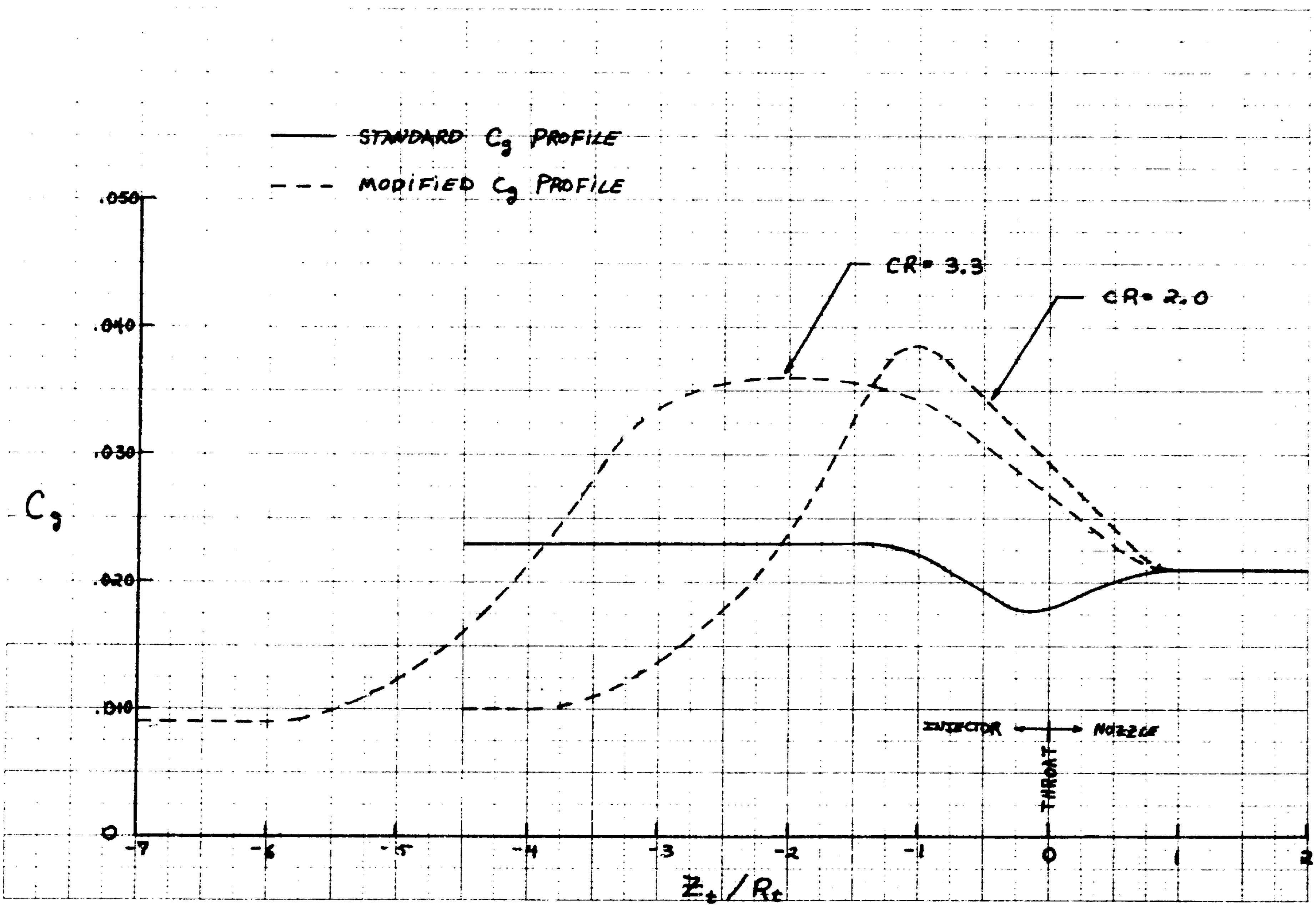

FIGURE 2. STANDARD AND MODIFIED C_g PROFILES

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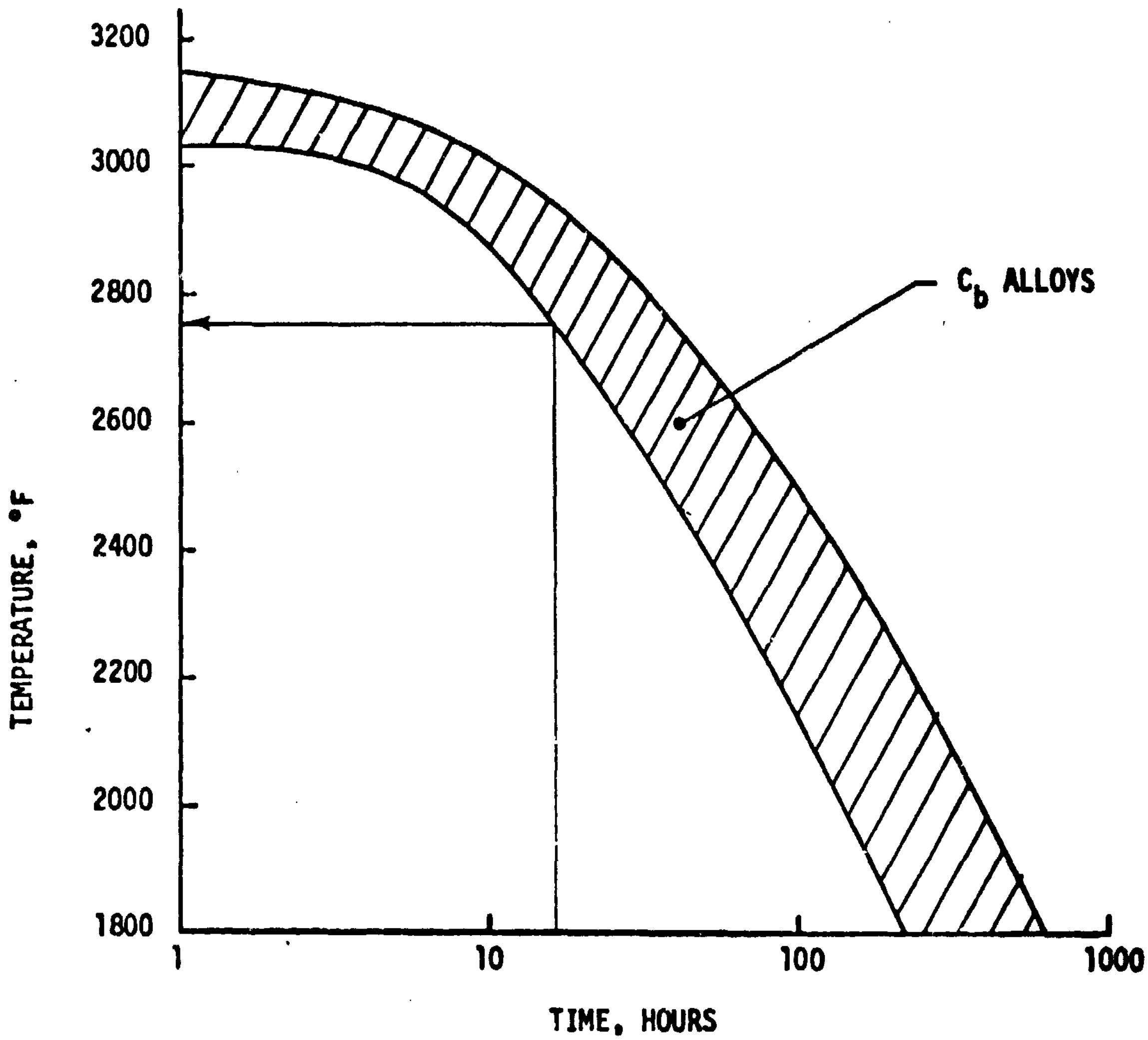
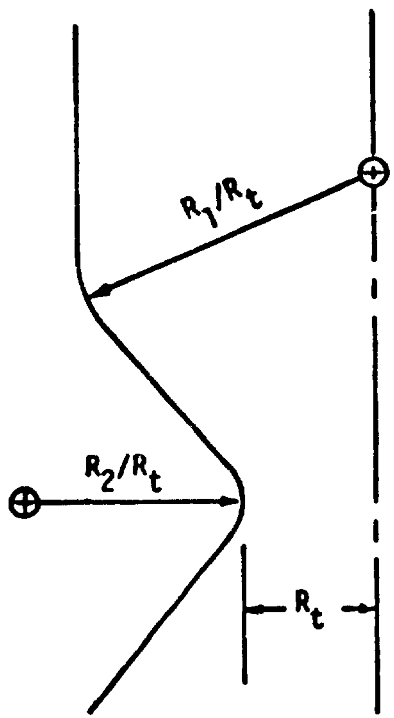
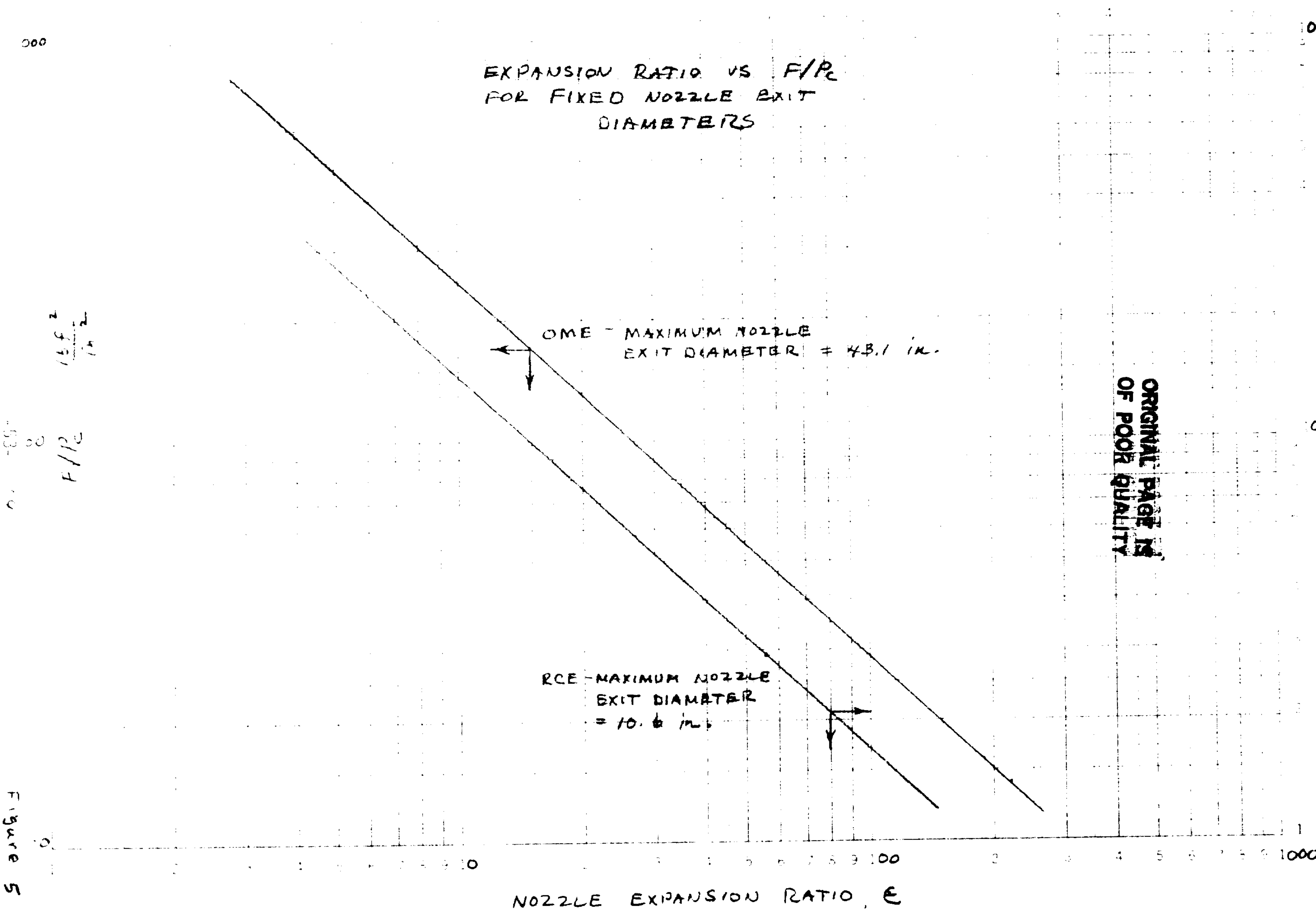


FIGURE 3. R512 SILICIDE COATING OXIDATION PROTECTION, TEMPERATURE VS DURATION



Contraction Ratio	3.3	2.0
Convergence Angle	30°	30°
R_1/R_t	2.5	1.4
R_2/R_t	1.0	.4

FIGURE 4. RECOMMENDED CHAMBER CONTOUR FOR OME APPLICATIONS


EXPANSION RATIO VS F/Pc
FOR FIXED NOZZLE EXIT
DIAMETERS
OME - MAXIMUM NOZZLE
EXIT DIAMETER = 43.1 in.
RCE - MAXIMUM NOZZLE
EXIT DIAMETER
= 10.6 in.
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10
100
1000
NOZZLE EXPANSION RATIO, ε

Figure 5

ALLOWABLE HOT GAS SIDE WALL TEMPERATURE
VS
CLOSEOUT WALL TEMPERATURE
FOR
N_f = 2000 cycles & t = 15 hours

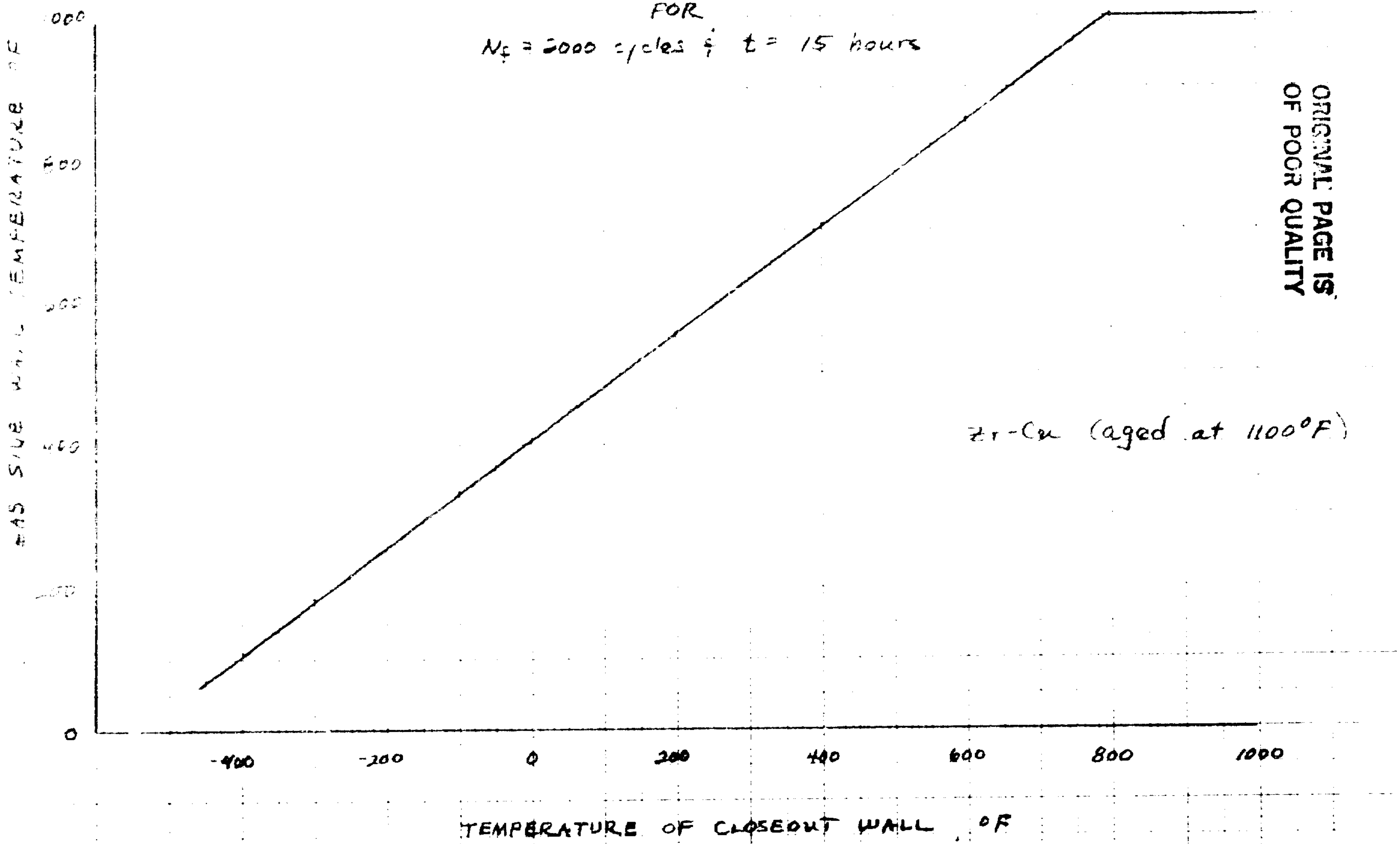

Figure 6

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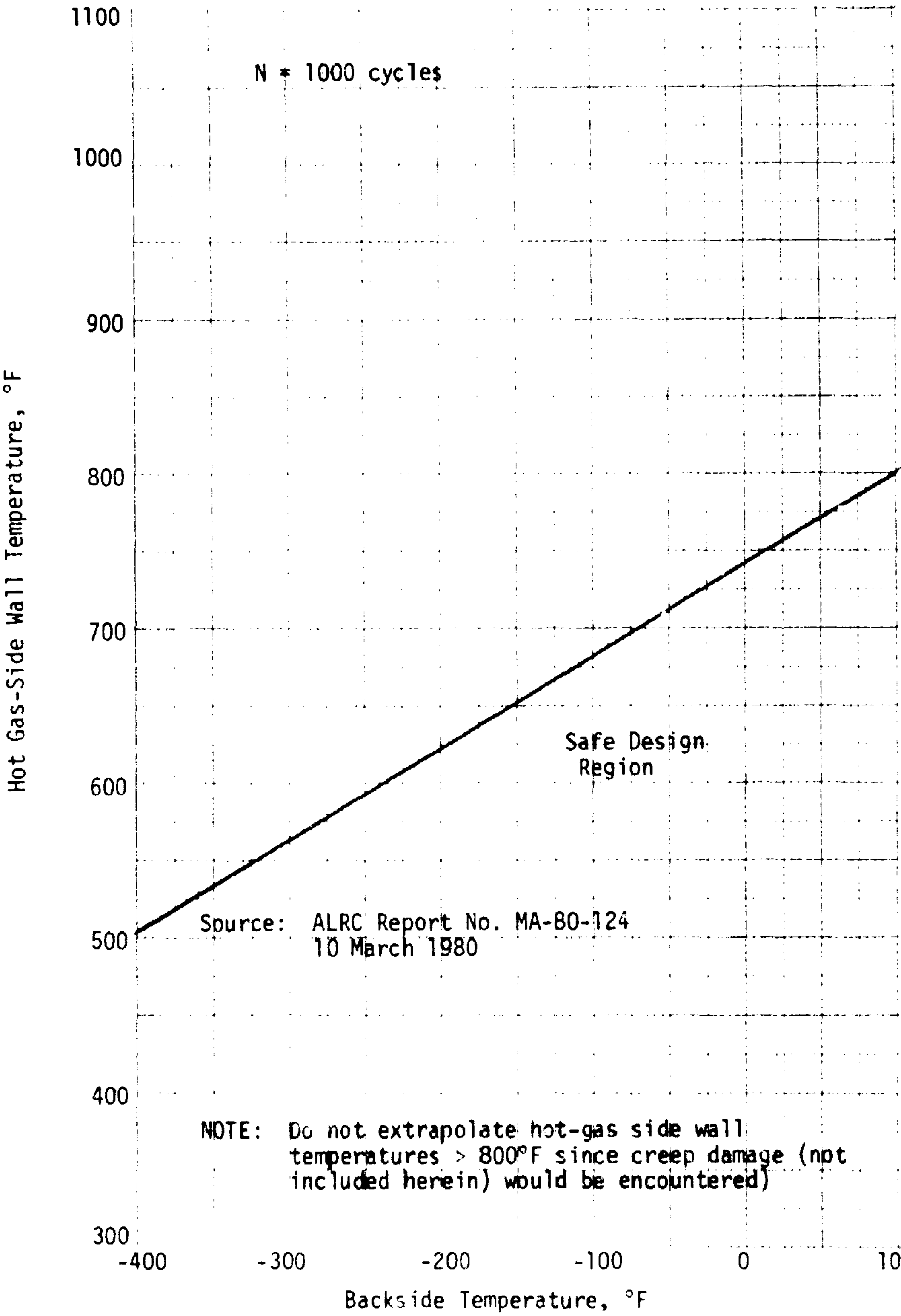

Figure 7. 304L Stainless Steel Design Envelope

CHANNEL LAYOUT

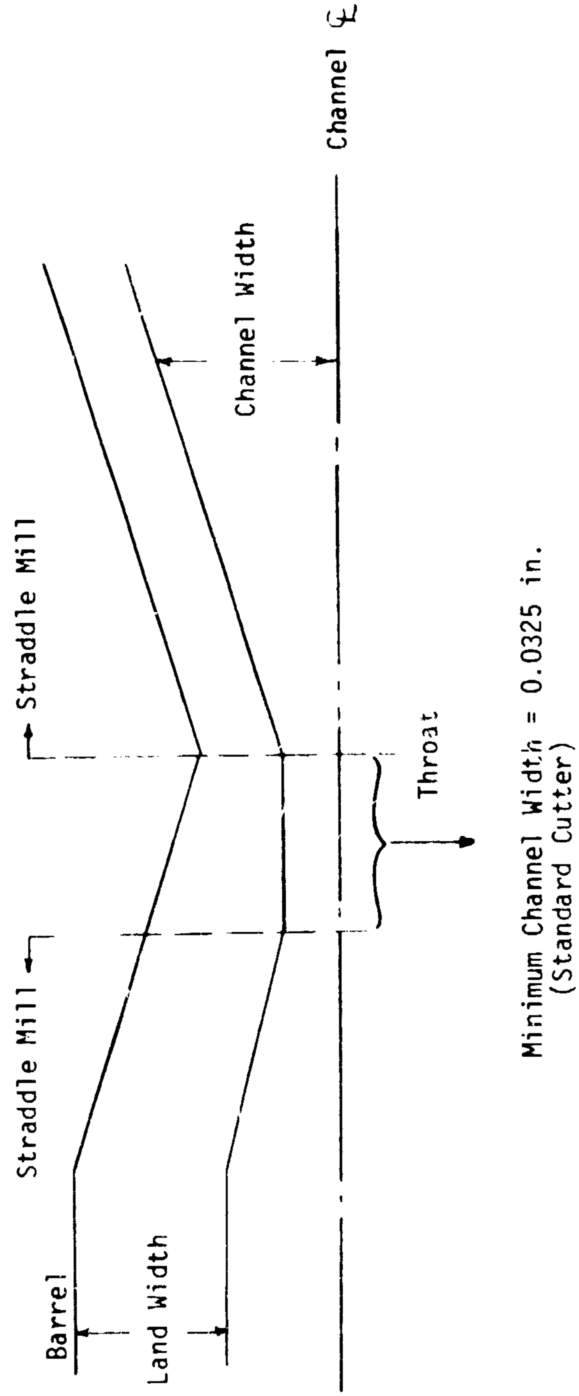

Figure 8

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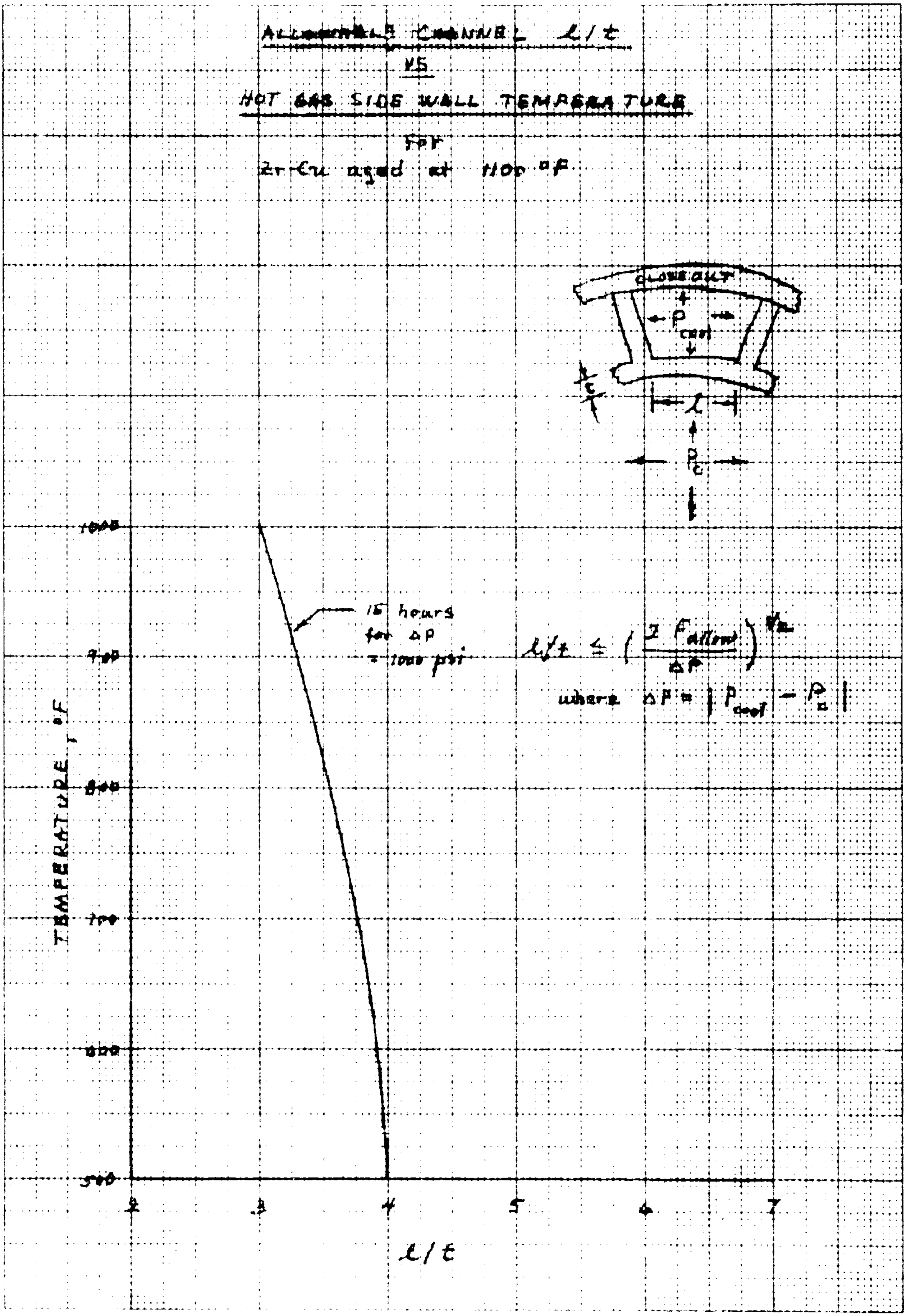

Figure 9

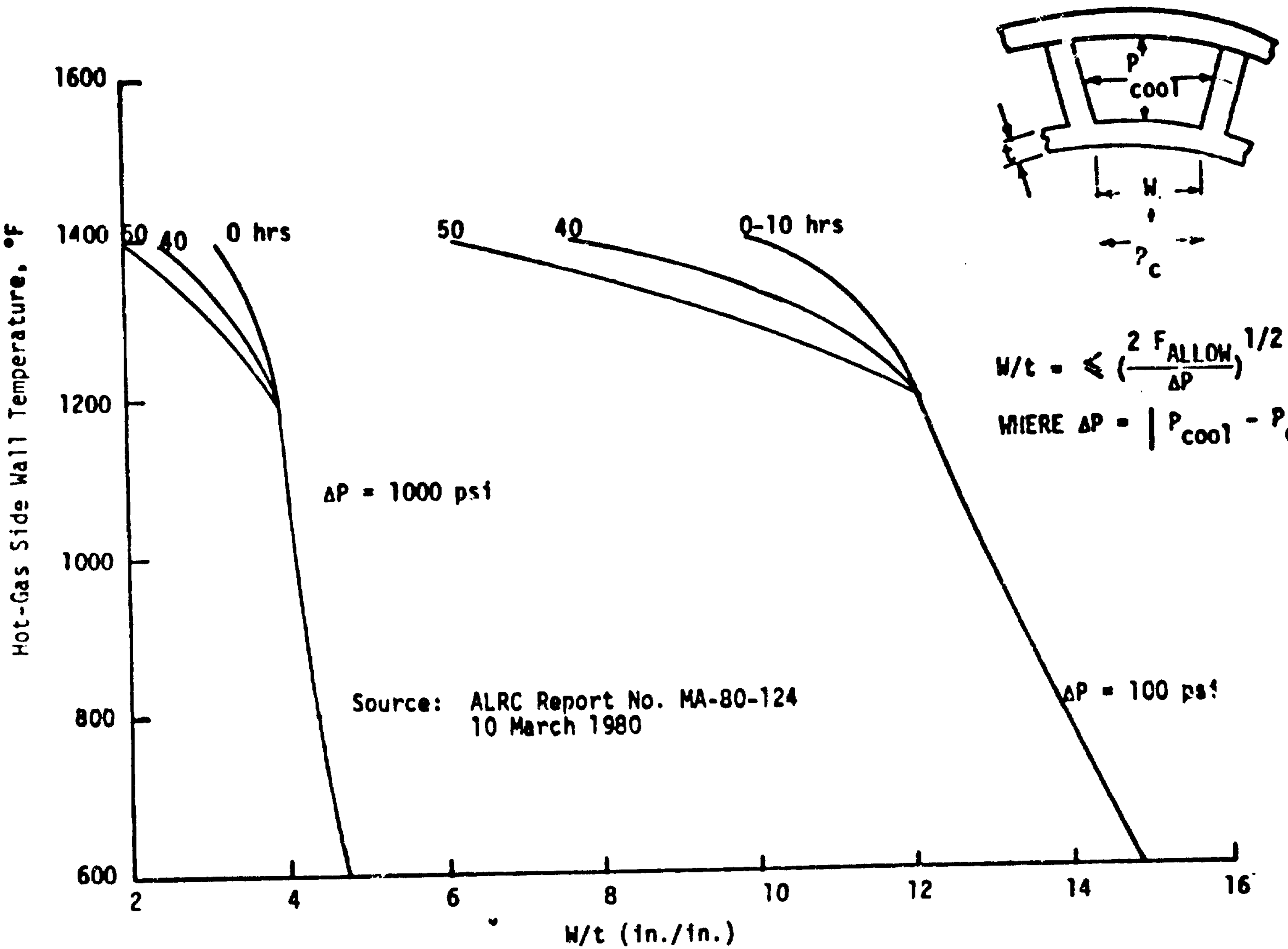

Figure 10. 304L Stainless Steel Wall Thickness Requirements

Aerojet
Liquid Rocket
Company

PROPULSION
ENGINEERING

THERMODYNAMIC ANALYSIS REPORT	NUMBER: 9751:0746 DATE: 19 October 1981
SUBJECT: REGENERATIVE AND FILM COOLING ANALYSIS OF LO_2/HC ENGINES FOR OME APPLICATIONS	PAGE 1 OF 7 NO. OF TABLES 3 NO. OF FIGURES 7
ADDITIONAL INFORMATION AND WORK NOTES INCLUDED IN MICROFILM FILE CDN______	

PREPARED FOR: S. Hart

SUMMARY

Thermal analyses were performed to assess the cooling characteristics of propane (C_3H_8), methane (CH_4), and oxygen (O_2) in three different chamber liners (Zr-Cu, CRES 304L, and N_i). Seven nominal OMS engine design points were analyzed; five pump-fed and two pressure-fed cases (See Table 1). Each nominal design point was degraded to the most severe thermal condition as follows:

	ΔMR	ΔPc
Pump-Fed	± 5%	± 10%
Pressure-Fed	+ 20%	+ 25%

Three nominal 6K/400 Pc propane cooled engines were analyzed. Using the modified ALRC C_g profile, the nickel and CRES 304L liner chambers required

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D. L. Kors, G. Meagner, J. Mellish, R. Michel, J. Salmon, L. Schoenman, R. Walker, 9751 File	PREPARED BY: S. BUCCELLA *S. Buccella*
	REVIEWED BY: W. R. THOMPSON *WR Thompson*
	APPROVED BY: *J. L. Pieper* J. L. PIEPER, MANAGER

Summary (cont.)

3% and 25% fuel film cooling, respectively (Cases 1 and 3). Using the standard flat C_g profile, a regeneratively cooled nickel chamber was designed (Case 2). A regeneratively cooled nickel chamber was designed for a nominal 6K/400 Pc methane cooled engine using the modified C_g profile (Case 4). A regeneratively cooled Zr-Cu chamber was designed for a nominal 10K/800 Pc oxygen cooled engine using the modified C_g profile (Case 5). All five pump-fed cases were cooled with supercritical propellants.

Two nominal 6K/100 Pc vapor cooled engines using the modified C_g profile were analyzed. A regeneratively cooled nickel chamber was designed for both propane and methane using 95% and 20% of the total fuel flow for coolant (Cases 6 and 7).

DISCUSSION OF RESULTS

Five pump-fed and two pressure-fed design cases as specified in Table 1 were analyzed. Each nominal operating point was degraded for the most severe thermal environment. These operating points are summarized in Table 2. Each operating point was analyzed to determine the feasibility of single up-pass regenerative cooling. If specified design criteria described in Section (c) were not met, film cooling was added. Detailed results of the thermal analyses are presented in Table 3.

(a) Pump-Fed Cooling Analysis

Five pump-fed off-nominal design points (Cases 1-5) were analyzed; four 6K lb nominal thrust engines with a nominal chamber pressure

Discussion of Results (cont.)

of 400 psia and one 10K nominal thrust engine with a nominal chamber pressure of 800 psia. All pump-fed cases were 3.3 contraction ratio engines cooled with supercritical coolant.

Case 1 was a nominal 6K lb thrust (degraded to 5400 lb thrust) engine with a nominal chamber pressure of 400 psia (degraded to 360 psia). The coolant was supercritical propane at an inlet pressure and temperature of 800 psia and -44°F, respectively. The chamber was a slotted nickel liner with an electroformed nickel closeout. The modified ALRC C_g profile (dashed line in Figure 1) was used. Due to the high heat flux in the convergent section of the chamber associated with this C_g profile, a regeneratively cooled design was not feasible due to cycle life constraints in that region. A minimum of 3% fuel film cooling was required to satisfy all design criteria. A 158 channel design with a pressure drop of 66 psia and a bulk temperature rise of 147°F was developed.

Case 2 was identical to Case 1, except the standard flat C_g profile (solid line in Figure 1) was used. The lower heat flux in the convergent section allowed a regeneratively cooled design to be developed. The 158 channel design resulted in a chamber pressure drop of 33 psia and a bulk temperature rise of 145°F.

Case 3 was identical to Case 1, except a slotted CRES 304L liner with an electroformed nickel closeout was used. A regeneratively cooled design could not be developed due to cycle life constraints resulting from poor liner thermal conductivity. A minimum of 25% film cooling was

Discussion of Results (cont.)

required. The 158 channel design resulted in a chamber pressure drop of 56 psia and a bulk temperature rise of 52°F.

Case 4 was identical to Case 1, except the coolant was supercritical methane at an inlet pressure and temperature of 900 psia and -259°F, respectively. The lower inlet temperature of methane compared to propane allowed a regeneratively cooled design to be developed. The 157 channel design resulted in a chamber pressure drop of 81 psia and a bulk temperature rise of 180°F.

Case 5 was a nominal 10K lb thrust (degraded to 9K lb thrust) engine with a nominal chamber pressure of 800 psia (degraded to 720 psia). The coolant was supercritical oxygen at an inlet pressure and temperature of 1260 psia and -286°F. The chamber was a slotted Zr-Cu liner with an electroformed nickel closeout. The modified ALRC C_g profile was used. A 145 channel regeneratively cooled design was developed with a chamber pressure drop of 227 psia and a bulk temperature rise of 114°F. The high inlet pressure was necessary to maintain the oxygen supercritical throughout the entire chamber.

(b) Pressure-Fed Cooling Analysis

Two pressure-fed off-nominal design points (Cases 6 and 7) were analyzed; both were 6K lb thrust (degraded to 4500 lb thrust) engines with a nominal chamber pressure of 100 psia (degraded to 75 psia). Both cases were 2.0 contraction ratio engines constructed of a slotted nickel liner with an electroformed nickel closeout. The modified ALRC C_g profile was used for both cases.

Discussion of Results (cont.)

Case 6 was cooled with propane vapor at an inlet pressure and temperature of 131 psia and 90°F, respectively. A 285 channel regeneratively cooled design was developed with a pressure drop of 14 psia and a bulk temperature rise of 131°F. The minimum coolant flow fraction was 95% due to the Mach number constraint on vapor flow ($M \leq 0.3$).

Case 7 was cooled with methane vapor at an inlet pressure and temperature of 131 psia and -180°F, respectively. A 292 channel design was developed with a chamber pressure drop of 7 psia and a bulk temperature rise of 926°F. The minimum coolant flow fraction was 20% due to Mach number constraints.

(c) Analysis Methods and Assumptions

Two cooling models were used to perform the OMS cooling analysis: the SCALER program for regeneratively cooled engines and the HEAT program for regeneratively cooled engines augmented by film cooling. Both models provided a detailed multi-station analysis of a rectangular channel. Further details on the SCALER and HEAT programs are given in References (a) and (b), respectively.

The analysis methods and assumptions used in the current study are the same as in Reference (a), except for the range of the gas-side boundary regimes. The conversion of the critical throat Reynolds number for a conversion angle of 30° to a product of thrust and chamber pressure for O_2/CH_4 and O_2/C_3H_8 engine are:

Discussion of Results (cont.)

	Thrust x Pc, lbF^2/in^2	
Propellants	Laminarized	Turbulent
O_2/CH_4 or C_3H_8	$\leq$ 111,000	$\geq$ 522,000

The gas-side C_g profiles used in the current analysis are discussed in Reference (a). Figure 1 illustrates the standard flat C_g profile (solid line) and the modified C_g profile (dash line) derived from ALRC experimental work.

(d) Analysis Criteria

The analysis criteria used in the current study are the same as in Reference (b), except the current study considers three different chamber liners(Zr-Cu, CRES 304L and Ni) and three different coolants (C_3H_8, CH_4 and O_2).

The service life of 500 thermal cycles times a safety factor of four and an accumulated run time of 15 hours limits (a) the maximum wall temperature to 1000°F* for Zr-Cu and Ni and 800°F* for CRES 304L and (b) the temperature differential between the gas-side and back-side wall temperature to the relationships shown in Figures 2 through 4. The allowable gas-side channel width-to-wall thickness requirement for Zr-Cu, CRES 304L and Ni are presented in Figures 5 through 7 respectively. The thermal conductivity of CRES 347 was used in place of CRES 304L due to availability.

---

*Cycle life criteria for Zr-Cu included allowance for creep and the 1000°F temperature upper limit is set by stress allowables. The criteria for 304L does not include allowance for creep and the 800°F limit is that set by increasing significance of creep.

Discussion of Results (cont.)

The forced convection characteristics of supercritical propane and methane were represented by the ALRC propane correlation:

$$Nu_b = .00545 \; Re_b^{0.9} \; Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-0.11} \left(\frac{\mu_b}{\mu_w}\right)^{0.23} \left(\frac{K_b}{K_w}\right)^{0.27} \left(\frac{\bar{C}_p}{C_{p_b}}\right)^{0.53}$$

The forced convection characteristics of supercritical oxygen were represented by the ALRC LOX correlation:

$$Nu_b = 0.0025 \; Re_b \; Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-0.5} \left(\frac{K_b}{K_w}\right)^{0.5} \left(\frac{\bar{C}_p}{C_{p_b}}\right)^{0.67} \left(\frac{P}{P_{crit}}\right)^{-0.2} \left(1 + \frac{2}{\ell/D}\right)$$

The forced convection characteristics of vapor propane and methane were represented with a film correlation:

$$Nu_b = .027 \; Re_F^{0.8} \; Pr_F^{0.4}$$

The allowable liquid-side wall temperature for carbon-containing coolants is controlled by the coolant coking temperature. The coking temperature used for propane was 600°F. The coking temperature for methane is greater than the maximum allowable wall temperature imposed by creep and cycle life considerations and therefore not applicable. The maximum allowable liquid-side wall temperature of oxygen in Zr-Cu is 600°F due to copper oxidization.

REFERENCES

(a) IOM 9751:0730, W. R. Thompson to S. Hart, dated 18 September 1981. Subject: Regenerative and Film Cooling Analysis of Ethanol as Coolant in OME/RCE Applications

(b) IOM 9751:0738, W. R. Thompson to S. Hart, dated 30 October 1981. Subject: Regenerative and Film Cooling Analyses of LOX/Hydrocarbon Propellants in Shuttle OME/RCE Applications

TABLE I - MDAC LO_2/HC OME STUDY PHASE II POINT DESIGNS

CASE NO.	PROPELLANTS	FV	PC	MAX Twg, °F	MAX Twl, °F	CHAMBER MATERIAL	ANALYSIS SPECIFICATIONS COMMENTS
1.	LO_2/C_3H_8	6K	400(1)	1000	600	Nickel	Max Twg based on cycle life of a nickel chamber needs to be defined. Conduct TPA analysis on this point design.
2.	LO_2/C_3H_8	6K	"	"	"	"	Use a flat Cg profile instead of that derived from NAS3-21030.
3.	LO_2/C_3H_8	6K	"	800	600	CRES 304L	Use established cycle life temperature criteria for CRES 304L
4.	LO_2/CH_4	6K	"	1000	-	Nickel	Use nickel chamber cycle life criteria.
5.	LO_2/CH_4	10K	800	1000	600	Zr-Cu	Regeneratively cool chamber with LO_2
6.	LO_2/C_3H_8	6K	100	1000	600	Nickel	Use fuel vapor cooling. Determine fraction of fuel required.
7.	LO_2/CH_4	6K	100	1000	-	Nickel	" " " " "

(1) Use supercritical fuel for chamber cooling.

TABLE II

DESIGN POINT PARAMETERS FOR OMS ENGINES

	Case 1	2	3	4	5	6	7
Feed Mode	Pump	Pump	Pump	Pump	Pump	Pressure	Pressure
Fuel Temperature, °F	-44	-44	-44	-259	-286	90	-180
Nominal Design Point							
Thrust, lbf	6K	6K	6K	6K	10K	6K	6K
Chamber Pressure, psia	400	400	400	400	800	100	100
Mixture Ratio, O/F*	2.80	2.80	2.80	3.50	3.50	2.75	3.00
Off-Design Variation Range							
Chamber Pressure, %	±10	±10	±10	±10	±10	±25	±25
Mixture Ratio, %	±5	±5	±5	±5	±5	±20	±20
Off-Design Variation Selected							
Chamber Pressure, %	-10	-10	-10	-10	-10	-25	-25
Mixture Ratio, %	+5	+5	+5	+5	-5	+20	+20
Off-Design Point							
Chamber Pressure, psia	360	360	360	360	720	75	75
Mixture Ratio, O/F	2.94	2.94	2.94	3.68	3.32	3.30	3.60

*Not specified in SOW. Determined by performance analysis for maximum Isp.

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CHAMBER THERMAL ANALYSIS TABLE III

CASE NO.	1	2	3	4	5	6	7
PROPELLANTS	LO_2/C_3H_8	LO_2/C_3H_8	LO_2/C_3H_8	LO_2/CH_4	LO_2/CH_4	LO_2/C_3H_8	LO_2/CH_4
Pc/F	400/6K	400/6K	400/6K	400/6K	800/10K	100/6K	100/6K
• C.R.	3.3	3.3	3.3	3.3	3.3	2.0	2.0
Thrust, lbs	5400	5400	5400	5400	9000	4500	4500
• Pc, psia	360	360	360	360	720	75	75
MR_{TCA}	2.85	2.94	2.21	3.68	3.32	3.30	3.60
MR_{Core}	2.94	2.94	2.94	3.68	3.32	3.30	3.60
• $\dot{W}ox$, lb/sec	11.16	11.17	11.18	11.64	18.62	10.23	10.33
• $\dot{W}_f$, lb/sec	3.92	3.80	5.05	3.16	5.61	3.10	2.87
No.of Regen Passe	1	1	1	1	1	1	1
• ΔPc.j., psi	66	33	56	81	227	14	7
Pc.j.-in.psia	800	800	800	900	1260	131	131
• Pc.j.out, psia	734	767	744	819	1033	117	124
Tc.j.-in.°F	-44	-44	-44	-259	-286	90	-180
Tc.j.-out, °F	103	101	8	-79	-172	221	746
• ΔTc.j., °F	147	145	52	180	114	131	926
Regen ε	6.73	6.73	6.73	11.33	28.4	6.23	6.23
• $\dot{W}_{ffc}$, lb/sec	0.12	-	1.26	-	-	-	-
%Fuel Film Coolant	3	-	25	-	-	-	-
T_{ffc-in}, °F	-44	-	-44	-	-	-	-
• Twg max, °F	776	659	693	1002	662	636	1000
• $\dot{W}_c$, lb/sec	3.92	3.80	5.05	3.16	18.62	2.95	0.57
• Coolant Flow Fraction	100	100	100	100	100	95	20

CHAMBER THERMAL ANALYSIS (cont.)

TABLE III

CASE NO.	1	2	3	4	5	6	7
PROPELLANTS	LO_2/C_3H_8	LO_2/C_3H_8	LO_2/C_3H_8	LO_2/CH_4	LO_2/CH_4	LO_2/C_3H_8	LO_2/CH_4
Pc/F	400/6K	400/6K	400/6K	400/6K	800/10K	100/6K	100/6K
Tw_1 max	582	569	431	880	610	600	989
h_g, BTU/in^2-sec-°F *	.00261	.00151	.00448	.00276	.00462	.000371	.000368
h_ℓ, BTU/in^2-sec-°F **	.0280	.0222	.0238	.0362	.0964	.00189	.00184
Tr, °F *	5179	5740	1687	5740	5815	5346	5346
Q/A_g max, BTU/in^2-sec	11.57	7.72	4.85	14.21	25.53	1.78	1.65
Q/A_ℓ max, BTU/in^2-sec	11.19	7.14	4.78	13.56	15.18	0.78	0.87
Q/Q_{Bo} max, BTU/in^2-sec	-	-	-	-	-	-	-
Q Total, BTU/sec	339	324	143	735	1096	186	345
V_c max, ft/sec	104	75	131	121	227	246	473
V_c (Mach No)^{max}	.030	.021	.036	.127	.115	.297	.250
No of channels	158	158	158	157	145	287	292
Min Ch Depth, in	.030	.040	.030	.032	.043	.147	.062

* at max gas-side flux

** at max coolant-side flux

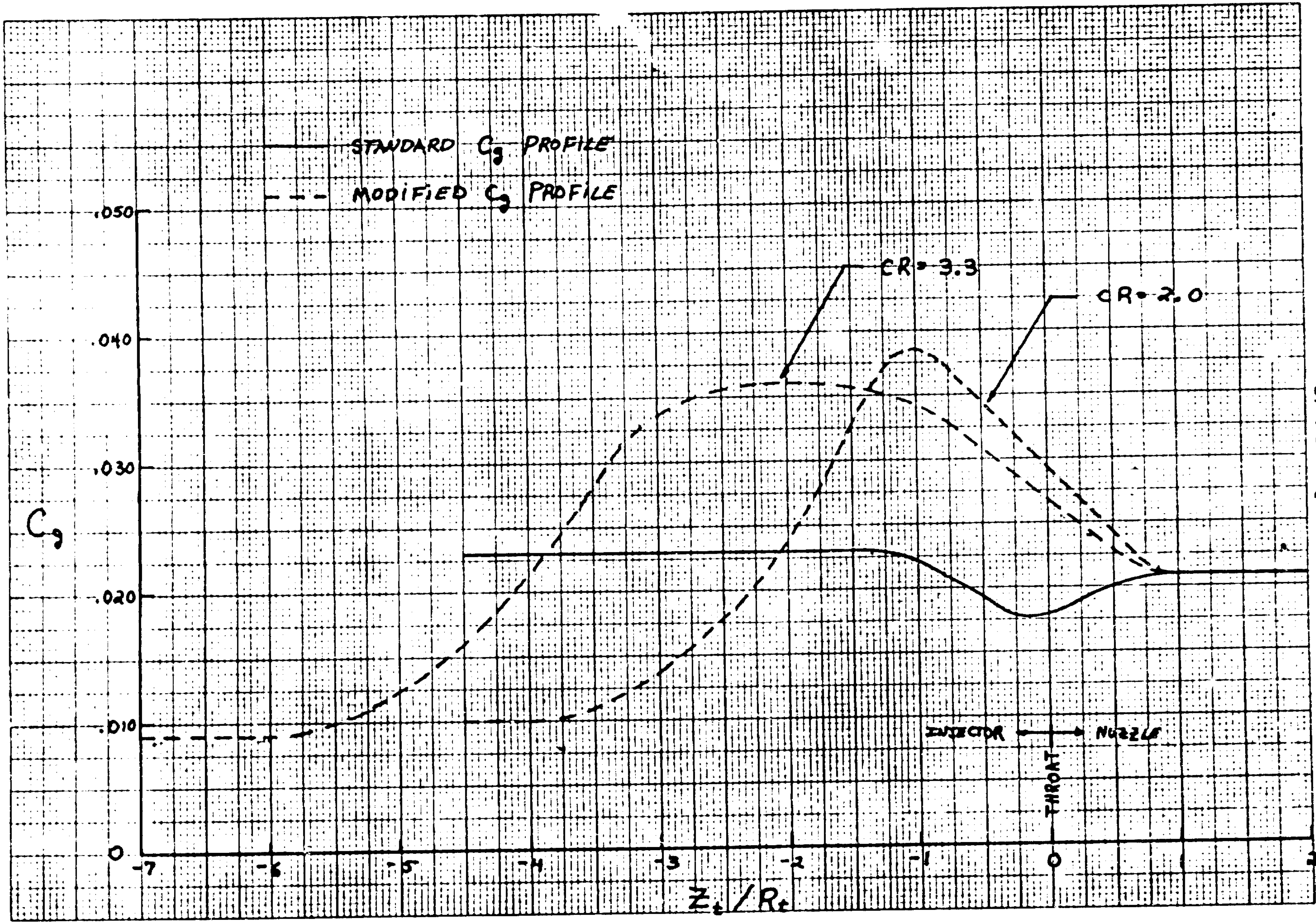

FIGURE 1. STANDARD AND MODIFIED C_g PROFILES

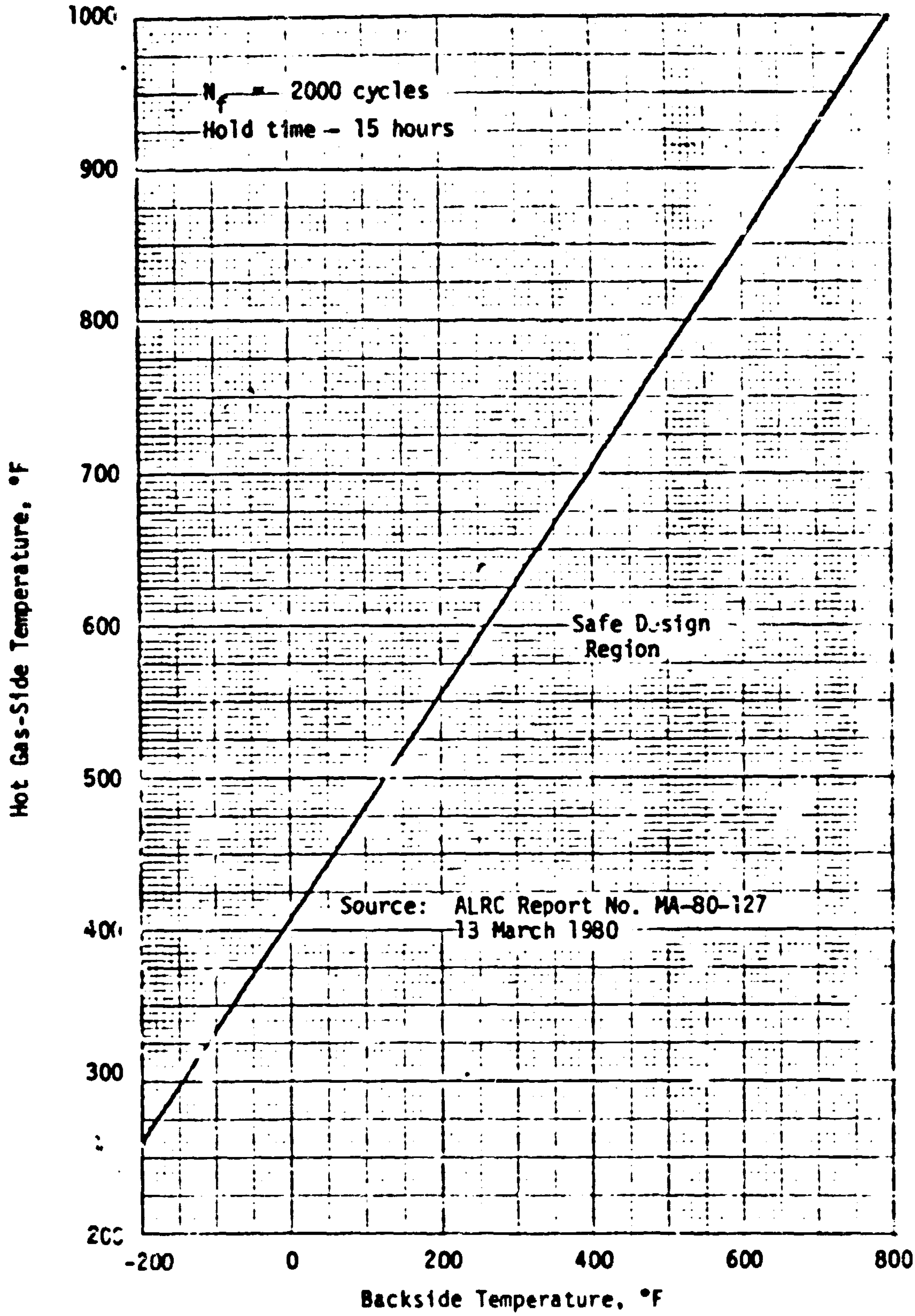

Figure 2. Zirconium-Copper Design Envelope (Solution Treated and Aged at 1100°F)

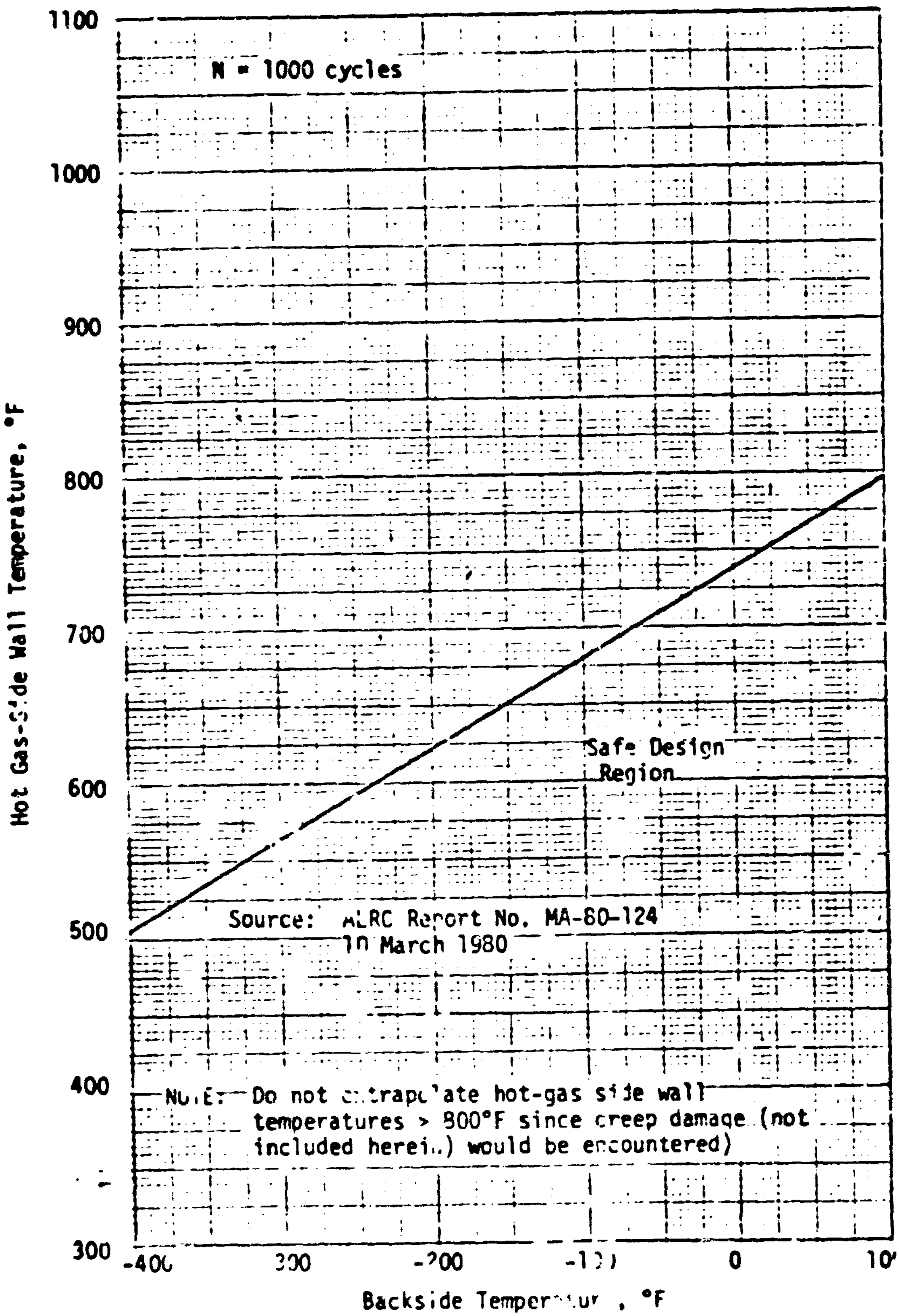

Figure 5. 304L Stainless Steel Design Envelope

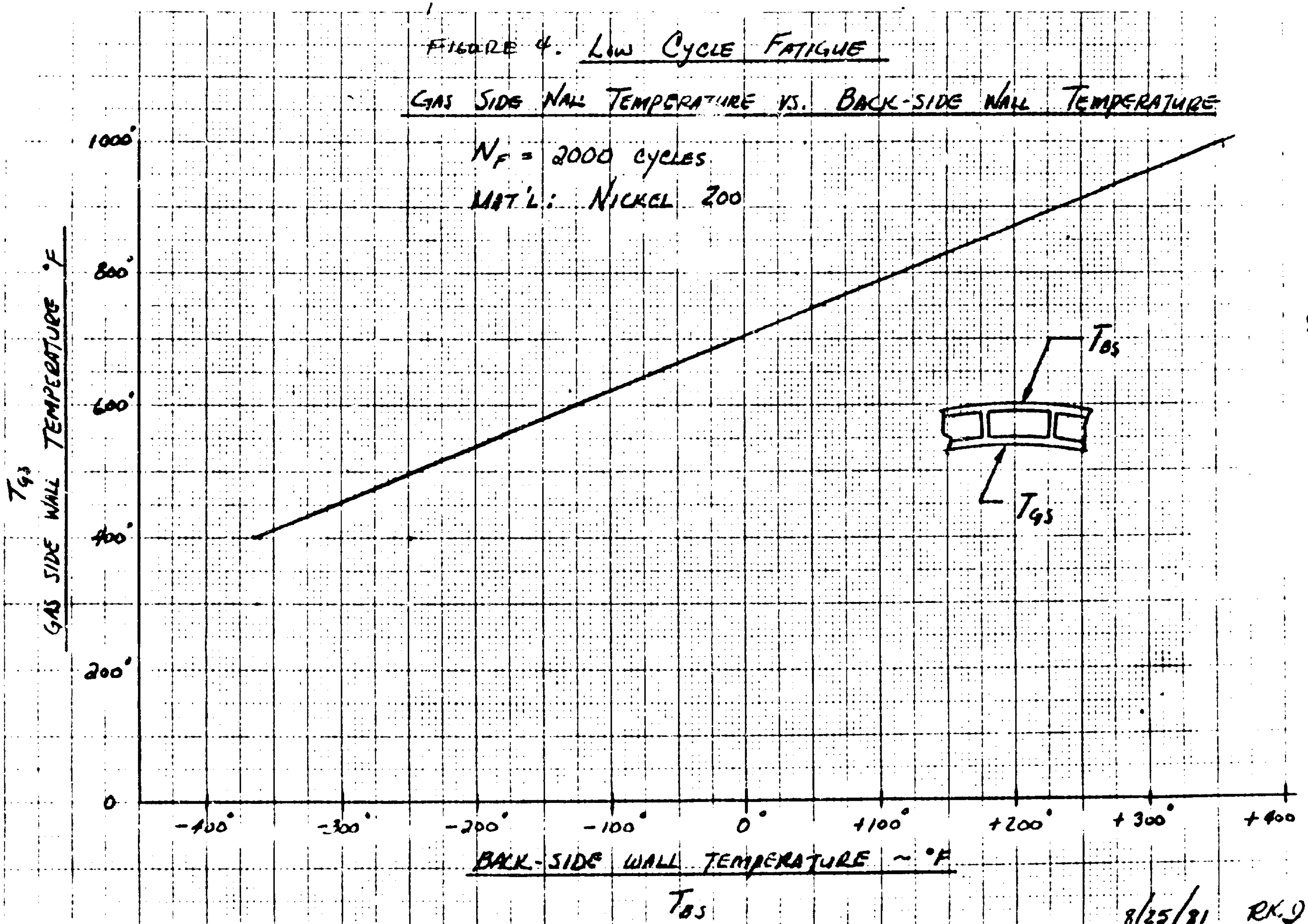
FIGURE 4. LOW CYCLE FATIGUE
GAS SIDE WALL TEMPERATURE VS. BACK-SIDE WALL TEMPERATURE
N_F = 2000 cycles
MAT'L: NICKEL 200
GAS SIDE WALL TEMPERATURE °F
T_{GS}
1000°
800°
600°
400°
200°
0
-400°
-300°
-200°
-100°
0°
+100°
+200°
+300°
+400
BACK-SIDE WALL TEMPERATURE ~ °F
T_{BS}
T_{BS}
T_{GS}
8/25/81
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Figure 4.

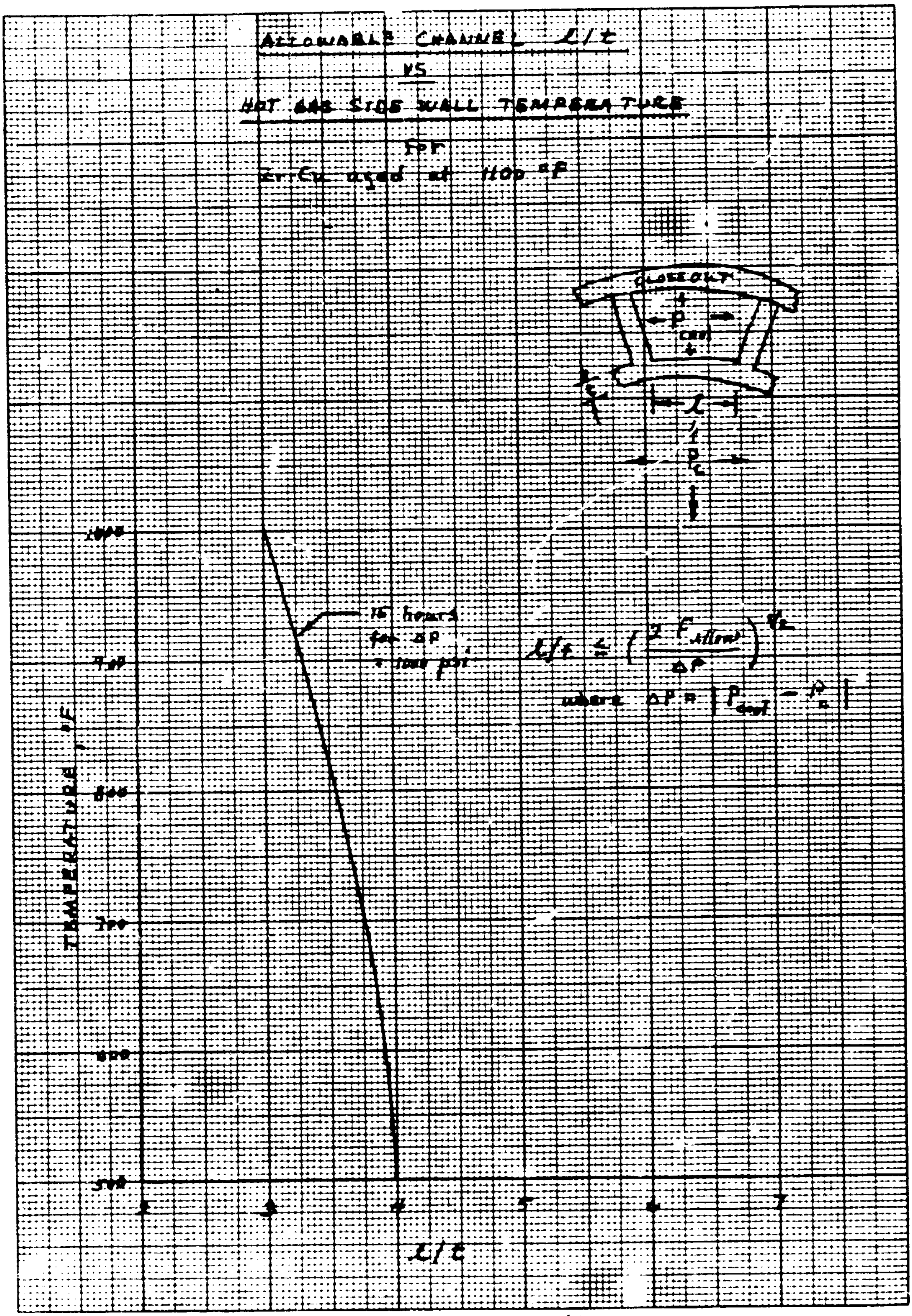

ALLOWABLE CHANNEL ℓ/t
VS
HOT GAS SIDE WALL TEMPERATURE
for
Zr-Cu aged at 1100 °F
CLOSEOUT
P
16 hours
for ΔP
= 1000 psi
$\ell/t \leq \left(\frac{2 F_{allow}}{\Delta P}\right)^{1/2}$
where $\Delta P = \left|P_{cool} - P_c\right|$
TEMPERATURE, °F
1000
900
800
700
600
500
2
3
4
5
6
7
ℓ/t

Figure 5

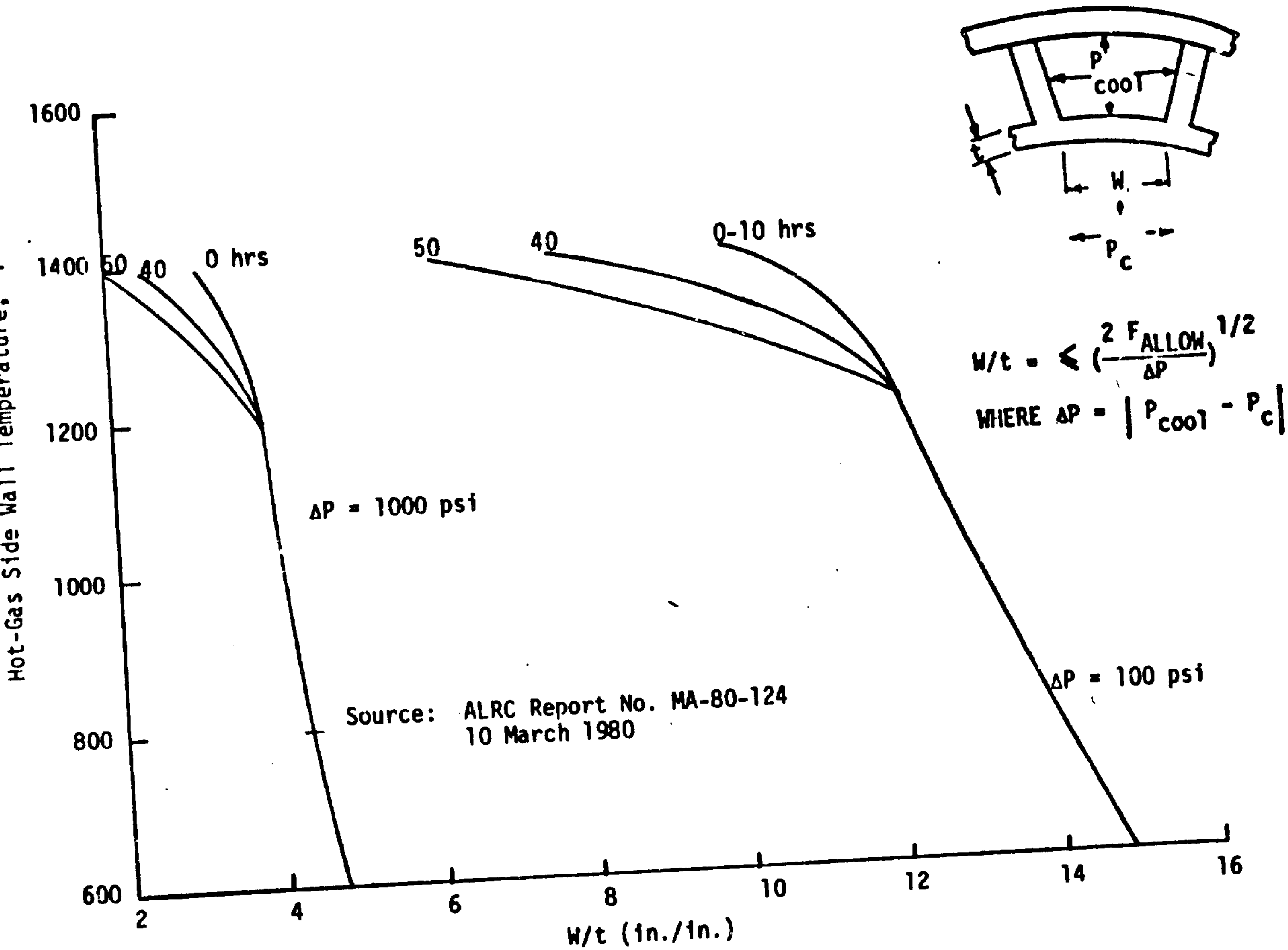
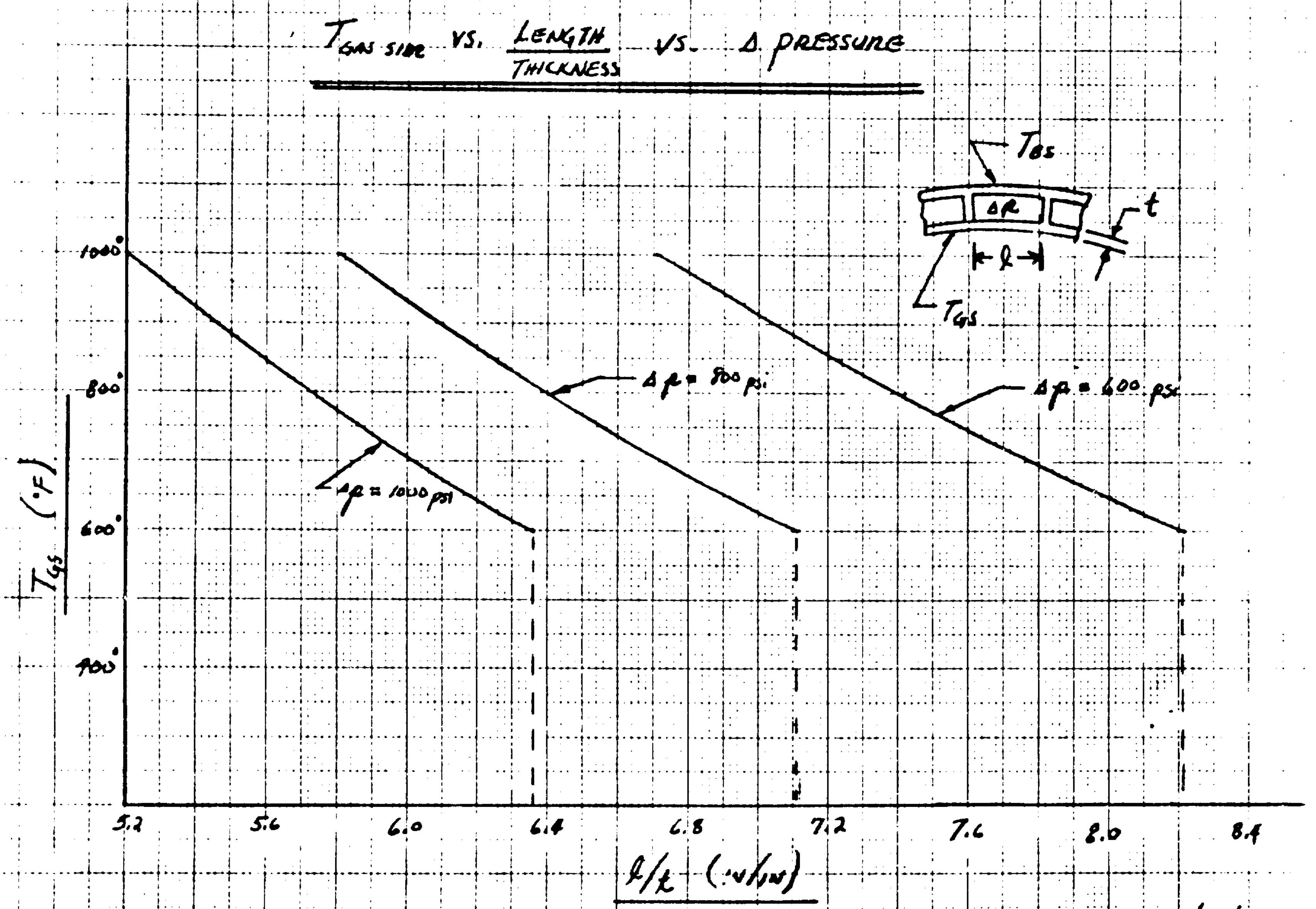


Figure 6. 304L Stainless Steel Wall Thickness Requirements

Fig. 2.0

Encl. (2)



8/25/81 RK9

Figure 7

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APPENDIX VI

PERFORMANCE REPORTS

NO. 9751:0694 AND 9751:0737

Aerojet
Liquid Rocket
Company

ROCKET
ENGINEERING

THERMODYNAMIC ANALYSIS REPORT	NUMBER: 9751:0694 DATE: 14 July 1981
SUBJECT: PERFORMANCE PREDICTIONS FOR ORBIT MANEUVERING (OME) AND REACTION CONTROL ENGINES (RCE) USING LOX/ HYDROCARBON AND AMMONIA PROPELLANTS	PAGE 1 OF 4 NO. OF TABLES 5 NO. OF FIGURES 0
ADDITIONAL INFORMATION AND WORK NOTES INCLUDED IN MICROFILM FILE CDN 7272	

PREPARED FOR: S. Hart

RESULTS

OME and RCE performance was predicted for the LOX/Hydrocarbon APS study (NAS 9-15958) parametric operating points. These operating points and guidelines are listed in Table I. The predicted OME and RCE performances are listed in Tables II and III respectively.

For a specific operating point (Thrust/Pc), methane yields the highest performance and ammonia the lowest. At the 10K/400 operating point, methane performance is predicted to be 1% greater than propane and approximately 9% greater than ammonia. The low performance with ammonia is a result of lower kinetic Isp and the fuel film cooling loss associated with the 13% film cooling. (The methane and propane engines are regeneratively cooled.) For the 6K/150 operating point the methane performance is approximately 6% better than the propane. This difference is primarily caused by the film cooling loss with 35% fuel film cooling for the propane engine. At the 870/150 operating point, the methane RCE had approximately 3% better

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D. Kors, R. Michel, J. Salmon, L. Schoenman, C. Teague, 9751 File J. Ito, W. Thompson, R. Ewen	PREPARED BY: G. M. MEAGHER *G.M. Meagher*
	REVIEWED BY: R. E. WALKER *R E Walker*
	APPROVED BY: J. L. PIEPER, MANAGER *J. L. Pieper*

Results (cont.)

performance than the propane engine, primarily a result of the higher kinetic Isp with methane. The RCE parametrics showed that for propane, increasing chamber pressure increased the amount of fuel film coolant required but resulted in a net engine performance increase due to an increase in allowable expansion ratios as chamber pressure was increased.*

Previously in this study (Reference (a)), OME and RCE performances were predicted for twelve baseline operating points. Recent performance model changes have resulted in the revised performance predictions shown on Tables IV and V for the OME and RCE cases respectively. Three performance model improvements were made subsequent to the original baseline analysis: (1) a modification to the turbulent boundary layer loss calculation was incorporated (based on BLIMP analyses) to account for laminar boundary layers at low Reynolds numbers (Reference (d)), (2) an improved kinetic loss correction for chamber throat size influences was incorporated, and (3) Isp vs mixture ratio data tables were expanded in order to better characterize the specific impulse contribution at film coolant mixture ratios. The net result of these changes was to change the predicted specific impulse for the OME baseline units by 1% or less, and to increase the predicted Isp for the baseline RCE engines by 2.5 to 3.5%. The more significant performance increase for the RCE engines was primarily the result of the boundary layer loss reduction for the smaller engines. For the 6K/100 propane baseline point, the propane was assumed to be liquid, but for the parametric point the propane was assumed to be gaseous. This change in state resulted in an increase in TCA performance of approximately 4%. This increase in performance resulted from being able to operate the gaseous propane engine using regenerative cooling while the liquid propane engine required film cooling.

*Detailed performance program output for the 10 cases, 6 OME and 4 RCE is contained in the appendix.

ANALYSIS

The predicted delivered specific impulse (Isp_{DEL}) is obtained by calculating the influence of the known loss mechanisms that degrade the ideal (Isp_{ODE}) performance. For this analysis these efficiencies/loss mechanisms were divided into five major categories: energy release efficiency (η_{ERE}), reaction kinetics efficiency (η_K), two-dimensional divergence efficiency (η_{2D}), loss due to the thrust decrement within the boundary layer, and loss due to film cooling.

A computer program was previously developed to help facilitate parametric analysis by representing each loss mechanism in a subroutine with the appropriate data base. For this study the energy release efficiencies were specified as 97.5% for the OME cases and 93% for the RCE cases; equal to the current values with earth storable propellants. Isp_{ODE} and Isp_{ODK} data were obtained using the Two-Dimensional Kinetics Program (TDK), Reference (b), and tabulated over a range of conditions that would encompass those desired for this analysis.

The kinetic efficiency was obtained by comparing the one-dimensional kinetic specific impulse (Isp_{ODK}) to the Isp_{ODE} ($\eta_K = Isp_{ODK}/Isp_{ODE}$). The two-dimensional efficiency was obtained from charts which gave the η_{2D} for optimum Rao nozzles as described in Reference (c). These charts were tabularized to facilitate their use in the performance program. The boundary loss was obtained by implementing the turbulent boundary layer chart procedures also given in Reference (c). The boundary layer efficiency was calculated assuming an adiabatic wall and propellants at the tank enthalpy. Past analysis have shown this approach to be quicker and results in the same efficiency as the more rigorous method of calculating the enthalpy loss to the regen coolant then finding a new Isp_{ODE} using the increased propellant enthalpy.

Analysis (cont.)

When film cooling was required, the efficiency was calculated by ratioing the mass weighted performance for the core and coolant stream tubes by the performance at the TCA MR. The low mixture ratio coolant stream tube performance is kinetically limited and model accuracy would be improved if empirical data were generated in order to anchor the predicted performance. For this study the low mixture ratio performance was estimated based on deviations from the ODE values obtained from the LOX/RP-1 fuel rich preburner data*.

---

*NASA Contract NAS 3-26647

REFERENCES

a. LOX/Hydrocarbon APS Study: Baseline Point Design Results, Contract NAS 9-15958, 15 May 1981

b. Nickerson, G. R., et. al., "The Two Dimensional Kinetic (TDK) Rocket Nozzle Analysis Reference Computer Program, Dec. 1973

c. Pieper, J. L., "ICRPG Liquid Propellant Thrust Chamber Performance Evaluation Manual", CPIA 178, September 1968

d. ALRC Memorandum 9751:0618, G. M. Meagher to L. Schoenman, "Low Thrust Isp Sensitivity Study: Comparison Between BLIMP and TBL Chart Predicted Performance Loss Due to Boundary Layer Influences

TABLE I

PARAMETRIC OPERATING POINTS
AND GUIDELINES

A. OPERATING POINTS

ENGINE	PROPELLANT	THRUST	CHAMBER PRESSURE
OME	O_2/NH_3	10K	400
OME	O_2/CH_4	10K	400
OME	O_2/CH_4	6K	150
OME	O_2/C_3H_8	10K	400
OME	O_2/C_3H_8	6K	100
OME	O_2/C_3H_8	6K	150
RCE	O_2/C_3H_8	870	100
RCE	O_2/C_3H_8	870	150
RCE	O_2/C_3H_8	870	300
RCE	O_2/CH_4	870	150

B. GUIDELINES

Mixture Ratio - Selected for maximum injector core kinetic Isp

Area Ratio - Based on current OMS and RCE exit diameter and length constraints.

	OME	RCE
Engine Length	75"	23"
Exit Diameter	43"	10.6"

TABLE II

OME PREDICTED PERFORMANCE

	O_2/NH_3	O_2/CH_4		O_2/C_3H_8		
Thrust (lbf)	10K	10K	6K	10K	6K	6K
Pc (psia)	400	400	150	400	100	150
TCA MR	1.28	3.5	3.4	2.80	2.75	1.99
$\dot{W}_{ox}$ (lbm/sec)	17.1	21.7	13.4	20.7	13.1	12.2
$\dot{W}_F$ (lbm/sec)	13.3	6.2	3.9	7.4	4.7	6.1
% Fuel Film Coolant (WF_C/WF)	13	-	-	-	-	35
ε	111	115	69.0	115	46	67
D_{Throat} (in)	4.14	4.0	5.12	4.02	6.34	5.26
D_{Exit} (in)	43.5	42.8	42.6	43	43	43.0
TCA Isp (sec)	328.8	358.7	346.2	355.2	337.0	326.7
Case No.	1	2	3	4	5	6

TABLE III

RCE PREDICTED PERFORMANCE

	O_2/CH_4	O_2/C_3H_8		
Thrust (lbf)	870	870	870	870
Pc (psia)	150	150	100	300
TCA MR	2.7	2.75	2.44	2.40
$\dot{W}_{ox}$ (lbm/sec)	2.0	2.0	2.1	2.7
$\dot{W}_F$ (lbm/sec)	.7	.8	.8	.8
% Fuel Film Coolant (WF_C/WF)	17.3	18.8	18.6	20.5
ε	27	27	18	56
D_{Throat} (in)	2.02	2.04	2.52	1.42
D_{Exit} (in)	10.5	10.6	10.7	10.6
TCA Isp (sec)	313.9	305.6	297.9	318.8
Case No.	7	8	9	10

TABLE IV

OME BASELINE OPERATING POINTS

PROPELLANTS	LOX/C_3H_8			LOX/CH_4		LOX/NH_3		
Pc/F	100/6K	800/10K	800/10K (Bst Pumps)	100/6K	800/10K	100/6K	800/10K	
• Engine Fv, lbf	6000	10,116	10,128	-	10,084	6000	10,107	
• TCA Fv, lbf	6000	10,000	10,000		10,000	6000	10,000	
• Engine MR	1.92	2.77	2.75		3.43	1.30	.93	
• TCA MR	2.11	3.0	3.0		3.5	1.30	1.06	
• Core MR	2.75	3.0	3.0		3.5	1.40	1.40	Max. ODK Isp MR
• Film Barrier MR	0.61	-	-		-	0.50	0.38	
• Turbine Ex. Fv, lbf	-	116	128		84	-	107	
• TCA $\dot{W}_{Tot}$, lbm/sec	18.50	27.07	27.07		27.12	18.82	31.09	
• TCA $\dot{W}_{ox}$, lbm/sec	12.55	20.3	20.3		21.02	10.64	16.02	
• TCA $\dot{W}_f$, lbm/sec	5.95	6.77	6.77		6.01	8.18	15.07	
• $\dot{W}_{turb}$, lbm/sec	-	0.82	0.90		0.51	-	0.76	
• % $W_{FFC} = (\dot{W}_{FFC}/\dot{W}_F)$	30	0	0		0	11	33	
• Eng $\dot{W}_{ox}$, lbm/sec	12.55	20.5	20.5		21.40	10.64	16.30	
• Eng $\dot{W}_f$, lbm/sec	5.95	7.37	7.47		6.24	8.18	15.55	
• Eng Isp, Sec	324.3	363.1	362.2		366.0	318.8	317.4	
• TCA Isp, sec	324.3	369.5	369.5		369.9	318.8	321.7	
• Core Isp (ODK), sec	350.1	387.7	387.7		388.6	337.8	362.7	
• ISP_{turb}, sec	-	141.8	141.8		164.3	-	141.2	
• Ae/At	44	240	240		236	44	224	
• D_t, in.	6.48	2.78	2.78		2.80	6.48	2.95	
• D_e, in	43	43.1	43		43	43	43.76	
• Fuel Film Coolant of Total Flow)	10.3	-	-		-	4.9	17	
• Engine Total Flow Rate, lbm/sec	18.50	27.87	27.97		27.64	18.82	31.85	

TABLE V

PREDICTED RCE BASELINE PERFORMANCE

PROPELLANTS	LOX/C_3H_8		LOX/CH_4		LOX/NH_3	
Pc	150	250	150	250	150	250
• <u>870 lbf Thrusters</u>						
• TCA MR	2.39	2.37	2.66	2.62	1.20	1.20
• Core MR	2.75	2.75	3.0	3.0	1.40	1.40
• TCA $\dot{W}_{ox}$, lbm/sec	2.01	1.94	2.02	1.95	1.62	1.57
• TCA $\dot{W}_f$, lbm/sec	0.84	.82	.76	.75	1.36	1.31
• $\dot{W}_{ffc}$,	19 (5.9)*	20 (6.3)	17 (4.9)	19 (5.5)	20 (9.4)	20 (9.4)
• TCA Isp, Sec	305.4	315.5	313.7	322.2	292.4	301.6
• Core Isp(ODK), sec	339.7	351.7	347.1	358.1	337.3	337.0
• Ae/At	27	46	27	46	25	46
• D_t, in	2.04	1.56	2.02	1.56	2.06	1.56
• D_{ex}, in	10.6	10.6	10.6	10.6	10.3	10.6
• <u>25 lbf Thrusters</u>						
• TCA MR	2.75	2.75	3.0	3.0	1.4	1.4
• Core MR	20	20	20	20	20	20
• TCA $\dot{W}_{ox}$, lbm/sec	.082	.080	.082	.080	.068	.066
• TCA $\dot{W}_f$, lbm/sec	.030	.029	.027	.027	.049	.047
• $\dot{W}_{ffc}$, lbm/sec	23	23	21	21	39	39
• TCA Isp, sec	222.4	229.8	228.7	233.6	214.0	219.5
• Core Isp(ODK), sec	-	-	-	-	-	-
• Ae/At	25	43	26	43	25	43
• D_t, in	0.36	0.27	0.35	0.27	0.36	0.27
• D_{ex}, in	1.80	1.80	1.80	1.80	1.80	1.80

APPENDIX

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A E R O J E T L I Q U I D R O C K E T C O M P A N Y

JSC OME APS STUDY

CASE NUMBER 1

PROPELLANTS LOX - AMMONIA

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	10000.	LBF	44482.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5585.	FT/SEC	1702.	M/SEC
GAMMA	1.213		1.213	
ISP ODE VACUUM	354.3	LBF-SEC/LBM	3474.6	N-SEC/KG
ISP DELIVERED VAC	328.8	LBF-SEC/LBM	3224.5	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9867		.9867	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9859		.9859	
DIVERGENCE	.9941		.9941	
COOLING	.9851		.9851	
OVERALL VACUUM	.9279		.9279	
GEOMETRY				
AREA RATIO	111.0		111.0	
EXIT RADIUS	21.76	INCH	.5527	M
THROAT RADIUS	2.07	INCH	.0525	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	62.5	INCH	1.59	M
L* LIQ/LIQ	13.66	INCH	.3469	M
L* LIQ/GAS	8.1	INCH	.2047	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	17.068	LBM/SEC	7.749	KG/SEC
FUEL	13.345	LBM/SEC	6.058	KG/SEC
TOTAL	30.413	LBM/SEC	13.795	KG/SEC
MIXTURE RATIO (TCA)	1.28		1.28	
(CORE)	1.40		1.40	
(BARRIER)	.47		.47	
(INJECTOR)	1.47		1.47	
% FILM COOLING	13.00		13.00	

A E R O J E T L I Q U I D R O C K E T C O M P A N Y

JSC OME APS STUDY

CASE NUMBER 2

PROPELLANTS LOX - METHANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	10000.	LBF	44482.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5792.	FT/SEC	1768.	M/SEC
GAMMA	1.200		1.200	
ISP ODE VACUUM	384.9	LBF-SEC/LBM	3774.8	N-SEC/KG
ISP DELIVERED VAC	358.7	LBF-SEC/LBM	3518.[illegible]	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9766		.9766	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9855		.9855	
DIVERGENCE	.9939		.9939	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9320		.9320	
GEOMETRY				
AREA RATIO	115.0		115.0	
EXIT RADIUS	21.44	INCH	.5446	M
THROAT RADIUS	2.00	INCH	.0508	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	61.7	INCH	1.57	M
L* LIQ/LIQ	13.66	INCH	.3469	M
L* LIQ/GAS	8.1	INCH	.2047	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	21.681	LBM/SEC	9.841	KG/SEC
FUEL	6.195	LBM/SEC	2.812	KG/SEC
TOTAL	27.876	LBM/SEC	12.644	KG/SEC
MIXTURE RATIO (TCA)	3.50		3.50	

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USC OME APS STUDY

CASE NUMBER 3

PROPELLANTS LOX - METHANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	150.	PSI	1034213.	PA
C-STAR DELIVERED	5749.	FT/SEC	1752.	M/SEC
GAMMA	1.202		1.202	
ISP ODE VACUUM	373.4	LBF-SEC/LBM	3661.4	N-SEC/KG
ISP DELIVERED VAC	346.2	LBF-SEC/LBM	3395.1	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9662		.9662	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9917		.9917	
DIVERGENCE	.9930		.9930	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9273		.9273	
GEOMETRY				
AREA RATIO	69.0		69.0	
EXIT RADIUS	21.29	INCH	.5409	M
THROAT RADIUS	2.56	INCH	.0651	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	59.4	INCH	1.51	M
L* LIQ/LIQ	15.21	INCH	.3865	M
L* LIQ/GAS	9.0	INCH	.2281	M
CONTRACTION RATIO	2.00		2.00	
FLOW RATES :				
OXIDIZER	13.392	LBM/SEC	6.079	KG/SEC
FUEL	3.939	LBM/SEC	1.788	KG/SEC
TOTAL	17.331	LBM/SEC	7.861	KG/SEC
MIXTURE RATIO (TCA)	3.40		3.40	

A E R O J E T   L I Q U I D   R O C K E T   C O M P A N Y

JSC OME APS STUDY

CASE NUMBER 4

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	10000.	LBF	44482.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5785.	FT/SEC	1763.	M/SEC
GAMMA	1.225		1.225	
ISP ODE VACUUM	377.7	LBF-SEC/LBM	3704.0	N-SEC/KG
ISP DELIVERED VAC	355.2	LBF-SEC/LBM	3483.2	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9839		.9839	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9869		.9869	
DIVERGENCE	.9939		.9939	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9404		.9404	
GEOMETRY				
AREA RATIO	115.0		115.0	
EXIT RADIUS	21.52	INCH	.5467	M
THROAT RADIUS	2.01	INCH	.0510	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	61.9	INCH	1.57	M
L* LIQ/LIQ	13.66	INCH	.3469	M
L' LIQ/GAS	8.1	INCH	.2047	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	20.745	LBM/SEC	9.417	KG/SEC
FUEL	7.409	LBM/SEC	3.363	KG/SEC
TOTAL	28.154	LBM/SEC	12.771	KG/SEC
MIXTURE RATIO (TCA)	2.80		2.80	

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AEROJET LIQUID ROCKET COMPANY

JSC OME APS STUDY

CASE NUMBER 5

PROPELLANTS LOX - PROPANE

	ENGLISH UNITS		S.I. UNITS	
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	100.	PSIA	689475.	PA
C-STAR DELIVERED	5705.	FT/SEC	1739.	M/SEC
GAMMA	1.230		1.230	
ISP ODE VACUUM	360.4	LBF-SEC/LBM	3534.6	N-SEC/KG
ISP DELIVERED VAC	337.0	LBF-SEC/LBM	3304.8	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9730		.9730	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9938		.9938	
DIVERGENCE	.9920		.9920	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9350		.9350	
GEOMETRY				
AREA RATIO	46.0		46.0	
EXIT RADIUS	21.50	INCH	.5461	M
THROAT RADIUS	3.17	INCH	.0805	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	58.1	INCH	1.48	M
L* LIQ/LIQ	16.70	INCH	.4242	M
L* LIQ/GAS	9.9	INCH	.2504	M
CONTRACTION RATIO	2.00		2.00	
FLOW RATES :				
OXIDIZER	13.057	LBM/SEC	5.927	KG/SEC
FUEL	4.749	LBM/SEC	2.155	KG/SEC
TOTAL	17.804	LBM/SEC	8.076	KG/SEC
MIXTURE RATIO (TCA)	2.75		2.75	

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A E R O J E T   L I Q U I D   R O C K E T   C O M P A N Y

JSC OME APS STUDY PARAMETRIC POINTS

CASE NUMBER 6

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	150.	PSIA	1034213.	PA
C-STAR DELIVERED	5746.	FT/SEC	1751.	M/SEC
GAMMA	1.237		1.237	
ISP ODE VACUUM	343.3	LBF-SEC/LBM	3366.5	N-SEC/KG
ISP DELIVERED VAC	326.7	LBF-SEC/LBM	3203.9	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9835		.9835	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9921		.9921	
DIVERGENCE	.9931		.9931	
COOLING	1.0075		1.0075	
OVERALL VACUUM	.9517		.9517	
GEOMETRY				
AREA RATIO	67.0		67.0	
EXIT RADIUS	21.51	INCH	.5465	M
THROAT RADIUS	[illegible].63	INCH	.0668	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	59.9	INCH	1.52	M
L* LIQ/LIQ	15.21	INCH	.3865	M
L* LIQ/GAS	9.0	INCH	.2281	M
CONTRACTION RATIO	2.00		2.00	
FLOW RATES :				
OXIDIZER	12.215	LBM/SEC	5.545	KG/SEC
FUEL	6.150	LBM/SEC	2.792	KG/SEC
TOTAL	18.365	LBM/SEC	8.330	KG/SEC
MIXTURE RATIO (TCA)	1.99		1.99	
(CORE)	2.75		2.75	
(BARRIER)	.57		.57	
(INJECTOR)	3.06		3.06	
% FILM COOLING	35.00		35.00	

A E R O J E T   L I Q U I D   R O C K E T   C O M P A N Y

USC RCE APS STUDY FUEL FILM COOLING

CASE NUMBER 7

PROPELLANTS LOX - METHANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	870.	LBF	3870.	N
CHAMBER PRESSURE	150.	PSIA	1034213.	PA
C-STAR DELIVERED	5561.	FT/SEC	1695.	M/SEC
GAMMA	1.208		1.208	
ISP ODE VACUUM	350.6	LBF-SEC/LBM	3438.0	N-SEC/KG
ISP DELIVERED VAC	313.9	LBF-SEC/LBM	3078.7	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9867		.9867	
ENERGY RELEASE	.9300		.9300	
BOUNDARY LAYER	.9944		.9944	
DIVERGENCE	.9913		.9913	
COOLING	.9908		.9908	
OVERALL VACUUM	.8955		.8955	
GEOMETRY				
AREA RATIO	27.0		27.0	
EXIT RADIUS	5.26	INCH	.1337	M
THROAT RADIUS	1.01	INCH	.0257	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	13.5	INCH	.34	M
L* LIQ/LIQ	9.76	INCH	.2479	M
L* LIQ/GAS	5.8	INCH	.1463	M
CONTRACTION RATIO	3.01		3.01	
FLOW RATES :				
OXIDIZER	2.022	LBM/SEC	.918	KG/SEC
FUEL	.749	LBM/SEC	.340	KG/SEC
TOTAL	2.771	LBM/SEC	1.257	KG/SEC
MIXTURE RATIO (TCA)	2.70		2.70	
(CORE)	3.00		3.00	
(BARRIER)	1.27		1.27	
(INJECTOR)	3.26		3.26	
% FILM COOLING	17.25		17.25	

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AEROJET LIQUID ROCKET COMPANY

JSC RCE APS STUDY FUEL FILM COOLING

CASE NUMBER 8

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	[illegible]	LBF	3870.	N
CHAMBER PRESSURE	[illegible]	PSIA	1034213.	PA
C-STAR DELIVERED	5[illegible]9.	FT/SEC	1652.	M/SEC
GAMMA	1.233		1.233	
ISP ODE VACUUM	3[illegible].1	LBF-SEC/LBM	3404.1	N-SEC/K[illegible]
ISP DELIVERED VAC	[illegible].6	LBF-SEC/LBM	2996.6	N-SEC/K[illegible]
EFFICIENCIES :				
KINETIC	.9776		.9776	
ENERGY RELEASE	.9300		.9300	
BOUNDARY LAYER	.9950		.9950	
DIVERGENCE	.9913		.9913	
COOLING	.9825		.9825	
OVERALL VACUUM	.8803		.8803	
GEOMETRY				
AREA RATIO	27.0		27.0	
EXIT RADIUS	5.29	INCH	.1343	M
THROAT RADIUS	1.02	INCH	.0259	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	13.5	INCH	.34	M
L* LIQ/LIQ	9.76	INCH	.2479	M
L* LIQ/GAS	5.8	INCH	.1463	M
CONTRACTION RATIO	3.01		3.01	
FLOW RATES :				
OXIDIZER	2.019	LBM/SEC	.917	KG/SEC
FUEL	.828	LBM/SEC	.376	KG/SEC
TOTAL	2.847	LBM/SEC	1.291	KG/SEC
MIXTURE RATIO (TCA)	2.44		2.44	
(CORE)	2.75		2.75	
(BARRIER)	1.09		1.09	
(INJECTOR)	3.00		3.00	
% FILM COOLING	18.75		18.75	

A E R O J E T   L I Q U I D   R O C K E T   C O M P A N Y

JSC RCE APS STUDY   FUEL FILM COOLING

CASE NUMBER 9

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	87[illegible].	LBF	3870.	N
CHAMBER PRESSURE	10[illegible].	PSIA	689475.	PA
C-STAR DELIVERED	539[illegible].	FT/SEC	1643.	M/SEC
GAMMA	1.234		1.234	
ISP ODE VACUUM	339.4	LBF-SEC/LBM	3328.4	N-SEC/KG
ISP DELIVERED VAC	297.9	LBF-SEC/LBM	2921.3	N-SEC/KG
EFFICIENCY :				
KINETIC	.9775		.9775	
ENERGY RELEASE	.9300		.9300	
BOUNDARY LAYER	.9953		.9953	
DIVERGENCE	.9910		.9910	
COOLING	.97[illegible]		.9797	
OVERALL VACUUM	.8[illegible]		.8777	
GEOMETRY				
AREA RATIO	18.		18.0	
EXIT RADIUS	5.35	INCH	.1359	M
THROAT RADIUS	1.26	INCH	.0323	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	13.[illegible]	INCH	.33	M
L* LIQ/LIQ	10.71	INCH	.2721	M
L* LIQ/GAS	6.3	INCH	.1606	M
CONTRACTION RATIO	3.01		3.01	
FLOW RATES :				
OXIDIZER	2.072	LBM/SEC	.940	KG/SEC
FUEL	.849	LBM/SEC	.385	KG/SEC
TOTAL	2.921	LBM/SEC	1.325	KG/SEC
MIXTURE RATIO (TCA)	2.44		2.44	
(CORE)	2.75		2.75	
(BARRIER)	1.09		1.09	
(INJECTOR)	3.00		3.00	
% FILM COOLING	18.60		18.60	

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AEROJET LIQUID ROCKET COMPANY

JSC RCE APS STUDY FUEL FILM COOLING

CASE NUMBER 10

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	870.	LBF	3870.	N
CHAMBER PRESSURE	300.	PSIA	2068425.	[illegible]A
C-STAR DELIVERED	5485.	FT/SEC	1672.	M/SEC
GAMMA	1.231		1.231	
ISP ODE VACUUM	358.3	LBF-SEC/LBM	3513.3	N-SEC/KG
ISP DELIVERED VAC	318.8	LBF-SEC/LBM	3126.5	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9824		.9824	
ENERGY RELEASE	.9300		.9300	
BOUNDARY LAYER	.9942		.9942	
DIVERGENCE	.9935		.9935	
COOLING	.9872		.9872	
OVERALL VACUUM	.8899		.8899	
GEOMETRY				
AREA RATIO	56.0		56.0	
EXIT RADIUS	5.29	INCH	.1344	M
THROAT RADIUS	.71	INCH	.0180	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	14.5	INCH	.37	M
L* LIQ/LIQ	8.32	INCH	.2113	M
L* LIQ/GAS	4.9	INCH	.1247	M
CONTRACTION RATIO	3.01		3.01	
FLOW RATES :				
OXIDIZER	1.926	LBM/SEC	.874	KG/SEC
FUEL	.803	LBM/SEC	.364	KG/SEC
TOTAL	2.729	LBM/SEC	1.238	KG/SEC
MIXTURE RATIO (TCA)	2.40		2.40	
(CORE)	2.75		2.75	
(BARRIER)	1.04		1.04	
(INJECTOR)	3.02		3.02	
% FILM COOLING	20.50		20.50	

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Aerojet Liquid Rocket Company

ROCKET ENGINEERING

THERMODYNAMIC ANALYSIS REPORT	NUMBER: 9751:0737 DATE: 9 October 1981
SUBJECT: PERFORMANCE PREDICTIONS FOR ORBIT MANEUVERING ENGINE (OME) USING LOX/HYDROCARBON PROPELLANTS	PAGE 1 OF 2 NO. OF TABLES 2 NO. OF FIGURES
ADDITIONAL INFORMATION AND WORK NOTES INCLUDED IN MICROFILM FILE CDN 7272	

PREPARED FOR: S. Hart

Performance predictions for six additional cases in the LOX/Hydrocarbon APS Study were analyzed to expand the data base previously generated for several baseline and parametric operating points documented in References (a) and (b). Table I identifies these cases and the corresponding performance. The cases documented in this report differ from the previous work because nickel or stainless steel was assumed for the thermal analysis of the chamber coolant channelsin place of copper.

Performance was unaffected by changing the channel material when the thermal analysis indicated regenerative cooling was possible. Use of methane results in higher performance than propane for an engine at the same operating point (thrust/Pc) and cooling scheme; this is consistent with the previous results (Ref. (a) & (b)). For the 6K/100 regenerative cooled engine, methane

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R. Ewen, J. Ito, D. Kors, R. Michel, J. Salmon, L. Schoenman, C. Teague, W. Thompson, 9751 File	PREPARED BY: G. M. MEAGHER *G. M. Meagher*
	REVIEWED BY: R. E. WALKER *R E Walker*
	APPROVED BY: *J. L. Pieper* J. L. PIEPER, MANAGER

resulted in 1.8% higher predicted performance than the propane case.

At the 6K/400 operating point four cases were evaluated, three LOX/C_3H_8 cases and one LOX/CH_4 case. Two of the LOX/C_3H_8 cases assumed nickel channels, Case 2 had a flat heat transfer coefficient (C_g) profile and Case 3 assumed a C_g gradient which varied with chamber location. The final LOX/C_3H_8 case, Case 4, assumed stainless steel channels and a varying C_g. The 6K/400 LOX/CH_4 case, Case 6, assumed nickel channels with a varying C_g. Comparing the two cases with nickel channels and a variable C_g profile, the LOX/Methane case yielded 1.3% higher performance than the LOX/Propane case. The two LOX/Propane cases with nickel channels, but different C_g profiles, results in nearly identical performance; a 0.1% performance penalty resulted from the 3% fuel film cooling required for the higher assumed C_g. Use of stainless steel channels (Case 4) required 25% fuel film cooling and resulted in a 3.5% performance decrease compared to the nickel chamber case.

Computer printouts for the cases analyzed are in the appendix.

TABLE I

OME PERFORMANCE

	LOX/C_3H_8				LOX/CH_4	
Case	1	2	3	4	5	6
Thrust (lbf)	6K	6K	6K	6K	6K	6K
Pc (psia)	100	400	400	400	100	400
TCA MR	2.75	2.80	2.72	2.27	3.00	3.50
CORE MR	2.75	2.80	2.80	2.80	3.00	3.50
Barrier MR	-	-	.17	.67	-	-
$\dot{W}_{ox}$ (lbm/sec)	13.06	12.28	12.20	11.97	13.11	12.80
$\dot{W}_f$ (lbm/sec)	4.75	4.39	4.48	5.28	4.37	3.66
% Fuel Film Cool	-	-	3.0	25.0	-	-
A_e/A_t	46	194	193	186	46	196
TCA Isp (lbf-sec/lbm)	337.0	360.1	359.7	347.9	343.2	364.5
Core Isp (lbf-sec/lbm)	350.7	377.1	377.0	375.5	356.4	382.2
R_t (in)	3.17	1.54	1.55	1.58	3.18	1.54
R_{Exit} (in)	21.5	21.5	21.5	21.5	21.6	21.5
Channel Design	N_i	N_i* Flat C_q Profile	N_i	S.S.	N_i	N_i

N_i - Nickel

S.S.- Stainless Steel

*All other designs assumed a variable C_q profile

REFERENCES

(a) ALRC Memo 9751:0674, Rev. A, G. M. Meagher to S. Hart, "Subject: Performance Predictions for Orbit Maneuvering and Reaction Control Engines Using LOX/Ethyl Alcohol Propellants", dated 20 July 1981

(b) ALRC Memo 9751:0694, G. M. Meagher to S. Hart, "Subject: Performance Predictions for Orbit Maneuvering (OME) and Reaction Control Engines (RCE) Using LOX/Hydrocarbon and Ammonia Propellants, dated 14 July 1981

APPENDIX

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AEROJET LIQUID ROCKET COMPANY

JSC OME APS STUDY PARAMETRIC POINTS

CASE NUMBER 1

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	100.	PSIA	689475.	PA
C-STAR DELIVERED	5705.	FT/SEC	1739.	M/SEC
GAMMA	1.230		1.230	
ISP ODE VACUUM	360.4	LBF-SEC/LBM	3534.6	N-SEC/KG
ISP DELIVERED VAC	337.0	LBF-SEC/LBM	3304.7	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9730		.9730	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9938		.9938	
DIVERGENCE	.9920		.9920	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9350		.9350	
GEOMETRY				
AREA RATIO	46.0		46.0	
EXIT RADIUS	21.50	INCH	.5461	M
THROAT RADIUS	3.17	INCH	.0805	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	58.1	INCH	1.48	M
L* LIQ/LIQ	16.70	INCH	.4242	M
L* LIQ/GAS	9.9	INCH	.2504	M
CONTRACTION RATIO	2.00		2.00	
FLOW RATES :				
OXIDIZER	13.057	LBM/SEC	5.927	KG/SEC
FUEL	4.748	LBM/SEC	2.155	KG/SEC
TOTAL	17.805	LBM/SEC	8.076	KG/SEC
MIXTURE RATIO (TCA)	2.75		2.75	

AEROJET LIQUID ROCKET COMPANY

JSC AFE OME STUDY ADD-ON

CASE NUMBER 2

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5765.	FT/SEC	1763.	M/SEC
GAMMA	1.225		1.225	
ISP ODE VACUUM	384.6	LBF-SEC/LBM	3772.1	N-SEC/KG
ISP DELIVERED VAC	360.1	LBF-SEC/LBM	3530.9	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9806		.9806	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9863		.9863	
DIVERGENCE	.9933		.9933	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9361		.9361	
GEOMETRY				
AREA RATIO	194.0		194.0	
EXIT RADIUS	21.51	INCH	.5463	M
THROAT RADIUS	1.54	INCH	.0392	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	63.3	INCH	1.61	M
L* LIQ/LIQ	12.14	INCH	.3084	M
L* LIQ/GAS	7.2	INCH	.1820	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	12.279	LBM/SEC	5.574	KG/SEC
FUEL	4.385	LBM/SEC	1.991	KG/SEC
TOTAL	16.664	LBM/SEC	7.559	KG/SEC
MIXTURE RATIO (TCA)	2.80		2.80	

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AEROJET LIQUID ROCKET COMPANY

JSC AFS OME STUDY ADD-ON

CASE NUMBER 3

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	400.	PSIA	27579.[illegible]	PA
C-STAR DELIVERED	5807.	FT/SEC	17[illegible]	M/SEC
GAMMA	1.225		1.225	
ISP ODE VACUUM	383.1	LBF-SEC/LBM	3756.9	N-SEC/KG
ISP DELIVERED VAC	359.7	LBF-SEC/LBM	3527.7	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9831		.9831	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9862		.9862	
DIVERGENCE	.9933		.9933	
COOLING	1.0006		1.0006	
OVERALL VACUUM	.9390		.9390	
GEOMETRY				
AREA RATIO	193.0		193.0	
EXIT RADIUS	21.50	INCH	.5460	M
THROAT RADIUS	1.55	INCH	.0393	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	63.3	INCH	1.61	M
L* LIQ/LIQ	12.14	INCH	.3084	M
L* LIQ/GAS	7.2	INCH	.1820	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	12.197	LBM/SEC	5.536	KG/SEC
FUEL	4.483	LBM/SEC	2.035	KG/SEC
TOTAL	16.680	LBM/SEC	7.566	KG/SEC
MIXTURE RATIO (TCA)	2.72		2.72	
(CORE)	2.60		2.80	
(BARRIER)	.17		.17	
(INJECTOR)	2.81		2.81	
% FILM COOLING	3.00		3.00	

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JSC APS OME STUDY ADD-ON

CASE NUMBER 4

PROPELLANTS LOX - PROPANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5850.	FT/SEC	1783.	M/SEC
GAMMA	1.230		1.230	
ISP ODE VACUUM	369.0	LBF-SEC/LBM	3618.4	N-SEC/KG
ISP DELIVERED VAC	347.9	LBF-SEC/LBM	3411.5	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9857		.9857	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9860		.9860	
DIVERGENCE	.9934		.9934	
COOLING	1.0021		1.0021	
OVERALL VACUUM	.9428		.9428	
GEOMETRY				
AREA RATIO	186.0		186.0	
EXIT RADIUS	21.52	INCH	.5467	M
THROAT RADIUS	1.58	INCH	.0401	M
PER CENT BELL	65.0		65.0	
NOZZLE LENGTH	63.3	INCH	1.61	M
L* LIQ/LIQ	12.14	INCH	.3084	M
L* LIQ/GAS	7.2	INCH	.1820	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	11.972	LBM/SEC	5.434	KG/SEC
FUEL	5.276	LBM/SEC	2.395	KG/SEC
TOTAL	17.248	LBM/SEC	7.823	KG/SEC
MIXTURE RATIO (TCA)	2.27		2.27	
(CORE)	2.80		2.80	
(BARRIER)	.67		.67	
(INJECTOR)	3.03		3.03	
% FILM COOLING	25.00		25.00	

AEROJET LIQUID ROCKET COMPANY

JSC APS OME STUDY ADD-ON

CASE NUMBER 5

PROPELLANTS LOX - METHANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	100.	PSIA	689475.	PA
C-STAR DELIVERED	5835.	FT/SEC	1779.	M/SEC
GAMMA	1.206		1.206	
ISP ODE VACUUM	364.6	LBF-SEC/LBM	3575.8	N-SEC/KG
ISP DELIVERED VAC	343.2	LBF-SEC/LBM	3365.3	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9787		.9787	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9932		.9932	
DIVERGENCE	.9933		.9933	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9411		.9411	
GEOMETRY				
AREA RATIO	46.0		46.0	
EXIT RADIUS	21.55	INCH	.5473	M
THROAT RADIUS	3.18	INCH	.0807	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	58.3	INCH	1.48	M
L* LIQ/LIQ	16.70	INCH	.4242	M
L* LIQ/GAS	9.9	INCH	.2504	M
CONTRACTION RATIO	2.00		2.00	
FLOW RATES :				
OXIDIZER	13.113	LBM/SEC	5.952	KG/SEC
FUEL	4.371	LBM/SEC	1.984	KG/SEC
TOTAL	17.484	LBM/SEC	7.931	KG/SEC
MIXTURE RATIO (TCA)	3.00		3.00	

OF POOR QUALITY

AEROJET LIQUID ROCKET COMPANY

JSC APS OME STUDY ADD-ON

CASE NUMBER 6

PROPELLANTS LOX - METHANE

	ENGLISH	UNITS	S.I.	UNITS
PERFORMANCE :				
THRUST VACUUM	6000.	LBF	26689.	N
CHAMBER PRESSURE	400.	PSIA	2757901.	PA
C-STAR DELIVERED	5799.	FT/SEC	1768.	M/SEC
GAMMA	1.200		1.200	
ISP ODE VACUUM	392.8	LBF-SEC/LBM	3852.5	N-SEC/KG
ISP DELIVERED VAC	364.5	LBF-SEC/LBM	3574.1	N-SEC/KG
EFFICIENCIES :				
KINETIC	.9729		.9729	
ENERGY RELEASE	.9750		.9750	
BOUNDARY LAYER	.9848		.9848	
DIVERGENCE	.9940		.9940	
COOLING	1.0000		1.0000	
OVERALL VACUUM	.9277		.9277	
GEOMETRY				
AREA RATIO	196.0		196.0	
EXIT RADIUS	21.51	INCH	.5464	M
THROAT RADIUS	1.54	INCH	.0390	M
PER CENT BELL	85.0		85.0	
NOZZLE LENGTH	63.4	INCH	1.61	M
L* LIQ/LIQ	12.14	INCH	.3084	M
L* LIQ/GAS	7.2	INCH	.1820	M
CONTRACTION RATIO	3.30		3.30	
FLOW RATES :				
OXIDIZER	12.804	LBM/SEC	5.812	KG/SEC
FUEL	3.658	LBM/SEC	1.661	KG/SEC
TOTAL	16.463	LBM/SEC	7.467	KG/SEC
MIXTURE RATIO (TCA)	3.50		3.50	