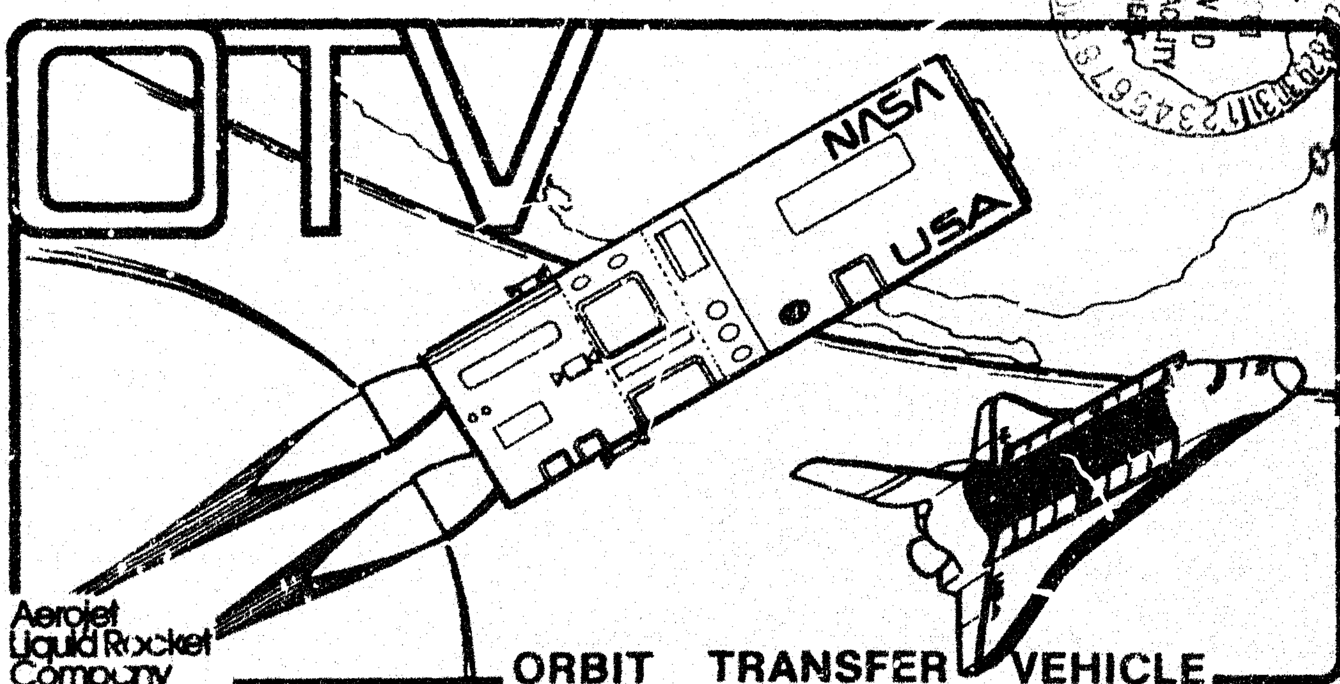
Orbit Transfer Vehicle Engine Study
Phase A, Extension 2

Contract NAS 8-32999
Final Report 32999-F-E2
January 1981

Prepared For:
George C. Marshall Space Flight Center
National Aeronautics And Space Administration



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ORBIT TRANSFER VEHICLE (OTV) ENGINE
PHASE "A" STUDY
EXTENSION 2

FINAL REPORT

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Prepared for

National Aeronautics and Space Administration
George C. Marshall Space Flight Center
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FOREWORD

This final report is submitted for the Orbit Transfer Vehicle (OTV) Engine Phase "A" Study Extension 2 per the requirements of Contract NAS 8-32999, Data Procurement Document No. 559, Data Requirement No. MA-05. This work was performed by the Aerojet Liquid Rocket Company for the NASA-Marshall Space Flight Center with Mr. Fred Braam, NASA/MSFC, as the Contracting Officer Representative (COR). The ALRC Program Manager was Mr. Larry B. Bassham and the Study Manager was Mr. Joseph A. Mellish

The study program consisted of three major technical tasks:

- Generation of additional OTV parametric engine data.
- Analysis of intermediate thrust level operation of the OTV engine.
- Analysis of engine operation during an aerobraking OTV (ABOTV) maneuver.

The technical period of performance for this study was from 6 October 1980 to 5 January 1981.

The following Aerojet personnel contributed significantly to the study effort and the final report.

R. A. Hewitt	-	Stability Analysis
J. I. Ito	-	Performance Analysis
B. R. Lawver	-	Performance and Stability Analysis
G. M. Meagher	-	Performance Analysis
W. R. Thompson	-	Thermal Analysis

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I INTRODUCTION

A. BACKGROUND

The Space Transportation System (STS) includes an Orbit Transfer Vehicle (OTV) that is carried into low Earth orbit by the Space Shuttle. The primary function of this OTV is to extend the STS operating regime beyond the Shuttle to include orbit plane changes, higher orbits, geosynchronous orbits and beyond. The NASA and DOD have been studying various types of OTV's in recent years. Data have been accumulated from the analyses of the various concepts, operating modes and projected missions. The foundation formulated by these studies established the desirability and the benefits of a low operating cost, reusable, high performance, versatile OTV.

The Orbit Transfer Vehicle (OTV) planned is a manned, reusable cryogenic upper stage.

The required round trip payload to geosynchronous orbit is 13,000 lbm, and the weight of the OTV, with propellants and payload, cannot exceed 97,300 lbm. The design mission is a four-man, 30-day sortie to geosynchronous orbit. An Orbiter of 100,000 lbm payload capability is planned, however, the OTV must be capable of interim operation with the present 65,000 lbm Orbiter. The cargo bay dimensions of the 100,000 lbm-Orbiter are assumed to be the same as the 65,000 lbm Orbiter, i.e., a cylinder 15-feet in diameter and 60-feet in length. The OTV cannot exceed 34 feet in length. The OTV is be Earth-based and will return from geosynchronous orbit for rendezevous with the Orbiter.

The OTV has as a goal the same basic characteristics as the Space Shuttle, i.e., reusability, operational flexibility, and payload retrieval along with a high reliability and low operating cost.

I Introduction (cont.)

B. STUDY OBJECTIVES

The primary objectives of this study were to evaluate operation of the Advanced Expander Cycle OTV engine at high mixture ratios (up to 8:1), and intermediate and low thrust levels to establish the impact upon the engine design and costs, and/or operating characteristics. Specific objectives were:

- Expand engine performance, weight and envelope parametrics data to include mixture ratios of 7.5 and 8.0.

- Assess engine operation at intermediate thrust levels and provide DDT&E, production and operations cost and schedule changes.

- Analyze engine operation at tank-head and pumped-idle conditions with the nozzle extension retracted.

The results of the three technical tasks corresponding to these objectives are presented in detail in the following sections. The task numbers refer to the number system in the contract Statement of Work (SOW).

II TASK 6.2.14 - EXPANDED PARAMETRIC DATA

A. STUDY GUIDELINES

The results of this task (per SOW, Exhibit "D", Paragraph 6.2.14) are parametric engine performance, weight and envelope data for the Advanced Expander Cycle Engine. The data presented herein is consistent with the parametric engine data generated earlier in the contract for SOW Exhibit A, Paragraph 6.2.3 and presented in Reference 1.

To maintain overall data consistency, the same computer model was used to generate the original and current sets of parametric data. The parameters and ranges used in the present analysis are listed in Table I below.

Table I
Parametric Data Ranges

Parameter	Range
Engine Thrust, KLBf	10, 15, 20 and 30
Engine Stowed Length, in.	40, 50, 60 and 70

Although not requested by the SOW, the data at the engine stowed length of 40 inches was generated at no increase in cost to the contract. The engine deployed length was assumed to be equal to twice the stowed length in all cases to minimize the engine stowed envelope. However, this assumption may not be valid for all the 40 inch stowed engine length cases. In these cases, the nozzles are necessarily small because of the length constraint. The extendible nozzles may not be retractable over the engine powerhead assembly. Preliminary calculations show this required clearance to be very small at best in most of the 40 inch stowed engine length cases. A more detailed design evaluation is required to determine the actual powerhead diameter and compare it to the extendible nozzle forward diameter if these cases are of further interest. The retraction of the extendible nozzle over the power head does not appear to be a problem for the other stowed lengths.

B. DISCUSSION OF PARAMETRIC DATA TRENDS

Figure 1 shows the nozzle exit diameter and engine weight as a function of engine thrust and stowed length. As these plots show, engine weight increases with both thrust level and stowed (or deployed) engine length. On the other hand, nozzle exit diameter either increases slightly or remains nearly constant with increasing thrust for a given engine stowed length. This trend is expected for a length constrained application such as the OTV and is explained herein.

First, by assuming a nearly constant engine thrust coefficient, it can be shown that

$$De \propto (\varepsilon F/Pc)^{1/2} \qquad (1)$$

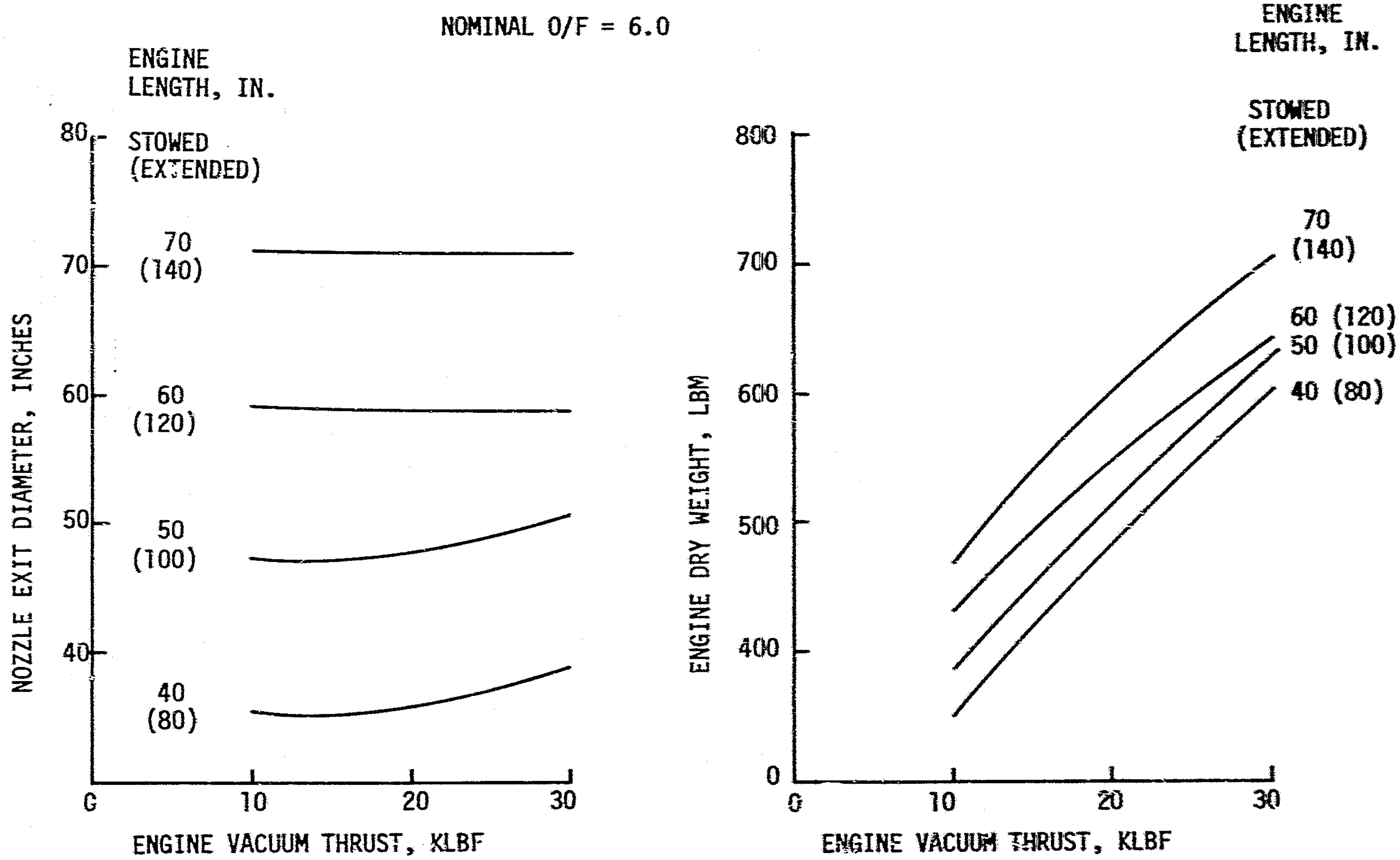


Figure 1. Advanced Expander Cycle Engine Nozzle Exit Diameter and Engine Dry Weight vs. Engine Vacuum Thrust and Engine Length

where

	De	=	Nozzle exit diameter
	ε	=	Nozzle area ratio
	F	=	Engine thrust
and	Pc	=	Chamber pressure

Secondly, for a fixed engine (and nozzle) length and constant chamber pressure, nozzle area ratio decreases as thrust level increases as shown by Figure 2. Similarly for a given thrust level and chamber pressure, as the engine length increases, so does the nozzle length (and size).

Finally, the expander cycle power balance requirements and thrust chamber cooling considerations dictate that the lower thrust engines will operate at higher chamber pressures than the higher thrust engines. Therefore, chamber pressure is decreasing with increasing thrust (see Table II below). These operating pressure levels were established during the initial Phase A study efforts. (Ref. 2).

Table II
Parametric Chamber Pressure Values

Thrust, KLBf	Chamber Pressure, psia
10	1300
15	1200
20	1100
30	950

These three considerations together result in a nozzle exit diameter that always increases with engine length but is relatively unaffected by thrust level as seen in Figure 1.

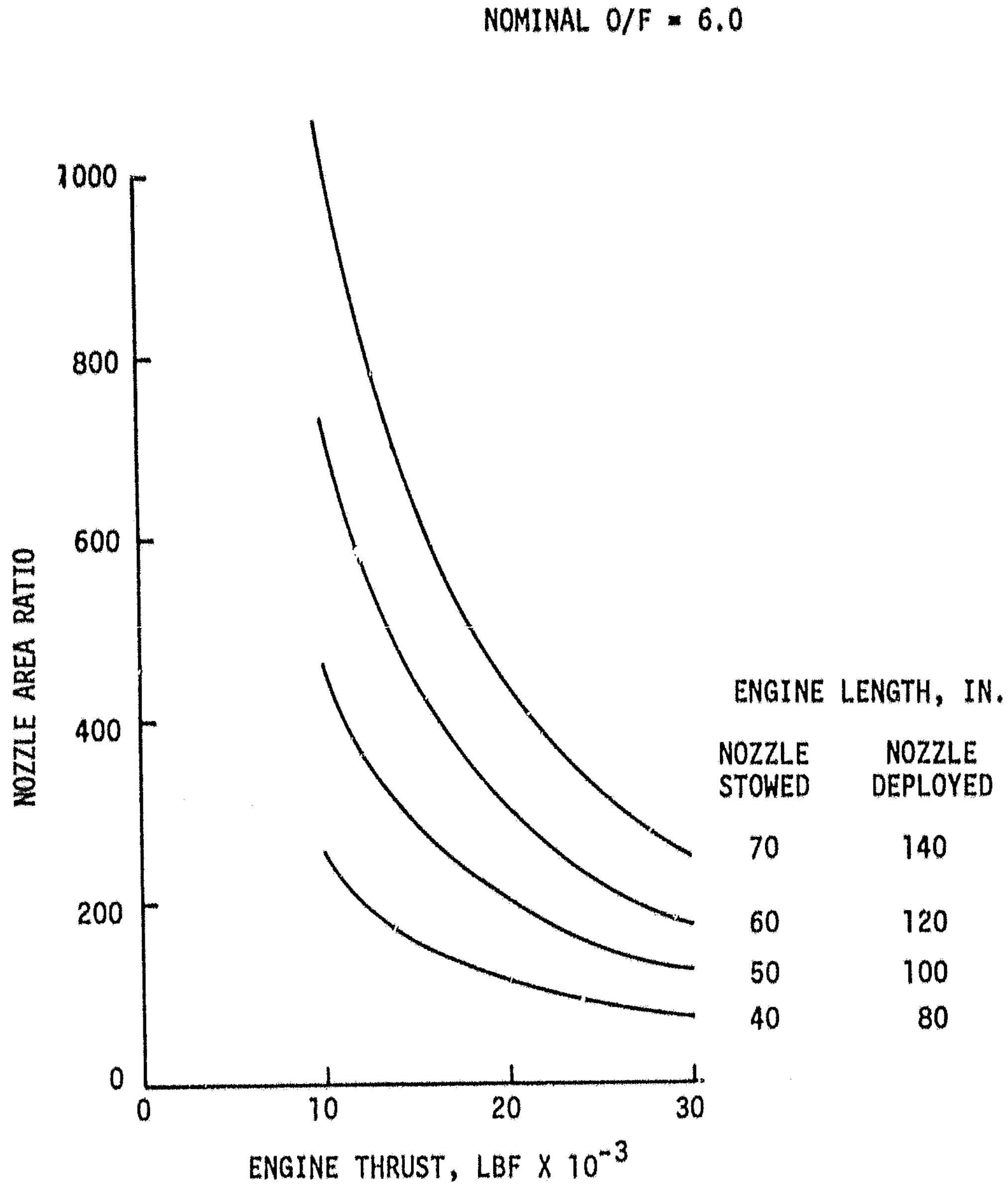


Figure 2. Advanced Expander Cycle Engine Nozzle Area Ratio vs. Engine Thrust and Engine Length

II Task 6.2.14 - Expanded Parametric Data (cont.)

Figure 3 shows engine delivered specific impulse as a function of engine thrust level and engine length (stowed and/or deployed). Of course, the specific impulse is higher for the longer engines. Also, specific impulse is lower at the higher thrust levels. Both chamber pressure and nozzle area ratio decrease as thrust level increases. These two factors both contribute to lower specific impulse as the thrust level increases.

Figure 4 through 7 are plots of specific impulse as a function of thrust level and engine mixture ratio. These figures correspond to engine stowed lengths of 40, 50, 60 and 70 inches, respectively. Performance gains with lengths greater than 60 inches are modest because the nozzle area ratios are so large that the theoretical gains diminish. For any given engine length and thrust level, the specific impulse is at, or near, a maximum at a mixture ratio between 6.0 and 6.5. Also, in every case, as expected, specific impulse is rapidly declining for mixture ratios greater than 6.5 because both the one-dimensional equilibrium (ODE) and kinetic (ODK) performance values decrease rapidly.

The delivered specific impulse values shown in Figures 3 through 7 were calculated according to the JANNAF Simplified Performance Prediction Methodology. The original JANNAF procedures were defined in 1968 (Reference 3), and were limited in scope to thrust chamber performance and an empirically based procedure for determining the energy release performance loss. Since this standard procedure is relatively costly in terms of both engineering-hours and computer time, there is a great incentive to use some simpler more economical procedure to perform parametric analyses.

Subsequent work by JANNAF led to less restrictive procedures and an expanded analytical approach. These updated procedures are defined in CPIA publications 245 (Reference 4) and 246 (Reference 5). CPIA 245 contains the specifications for performance test data acquisition and interpretation. CPIA 246 contains the specifications for liquid rocket engine performance prediction and evaluation, including the Simplified Performance Prediction Methodology.

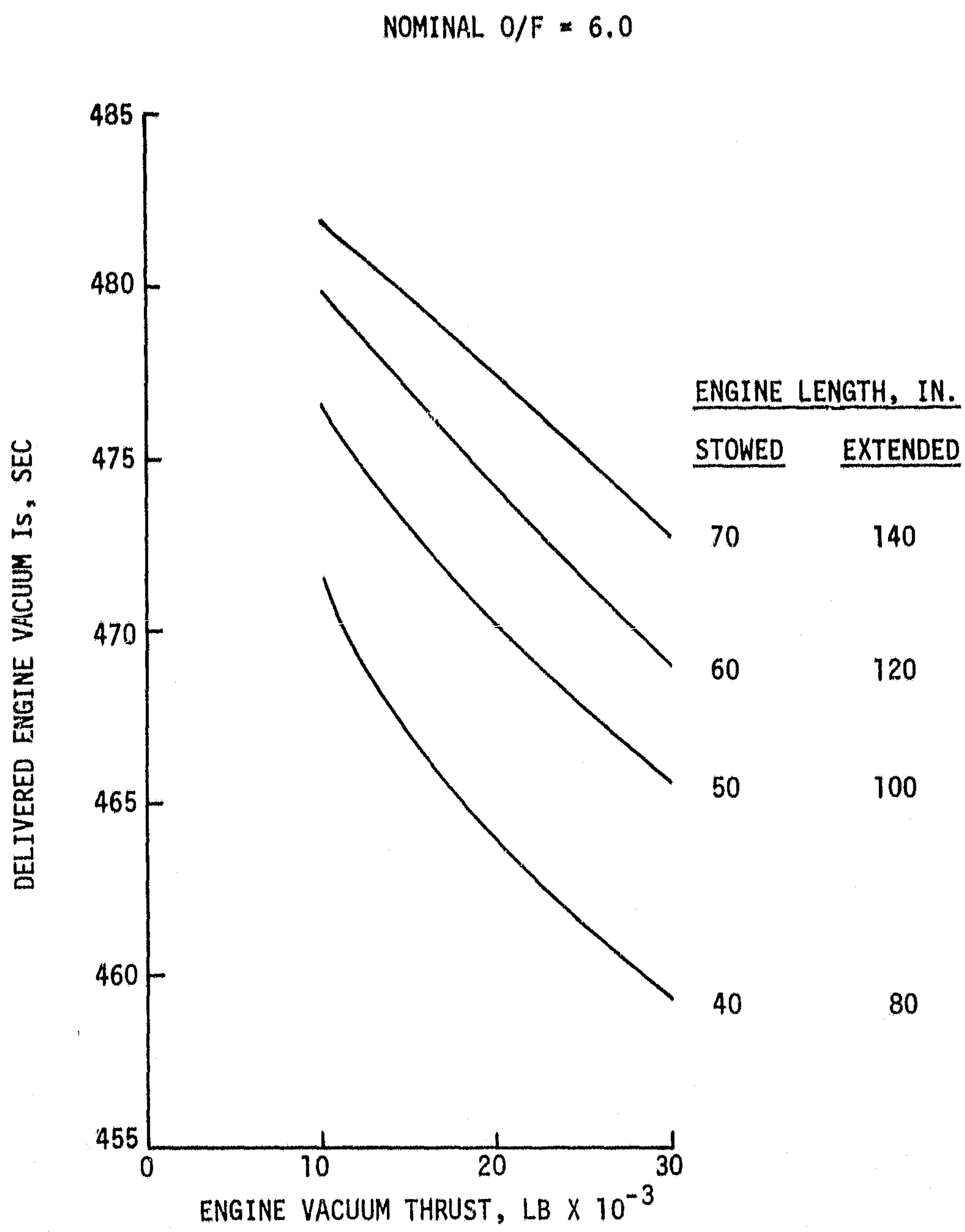


Figure 3. Advanced Expander Cycle Engine Performance Variations with Rated Thrust

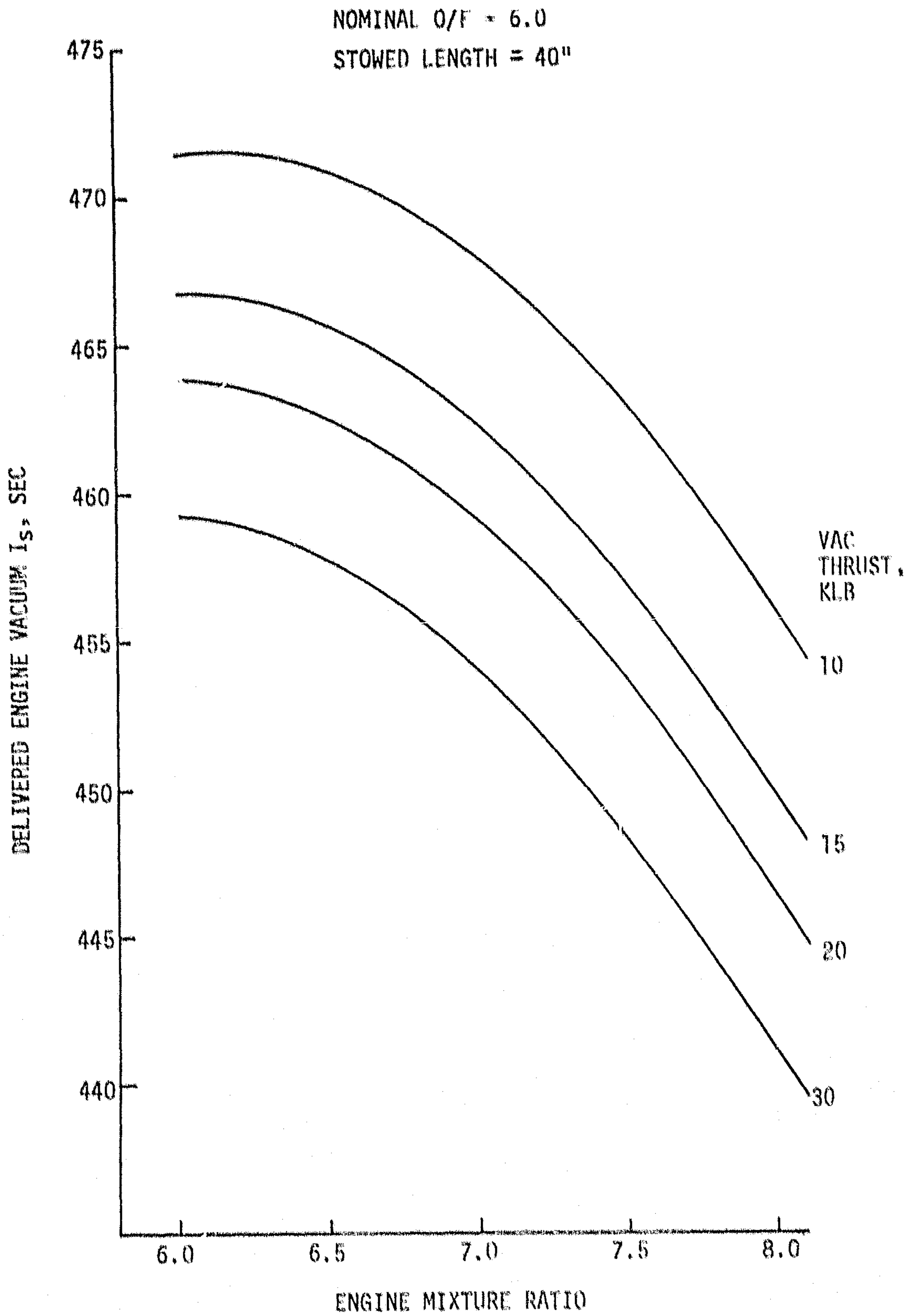


Figure 4. Advanced Expander Cycle Engine Performance at Design and Off-Design O/F (Stowed Length = 40 Inches)

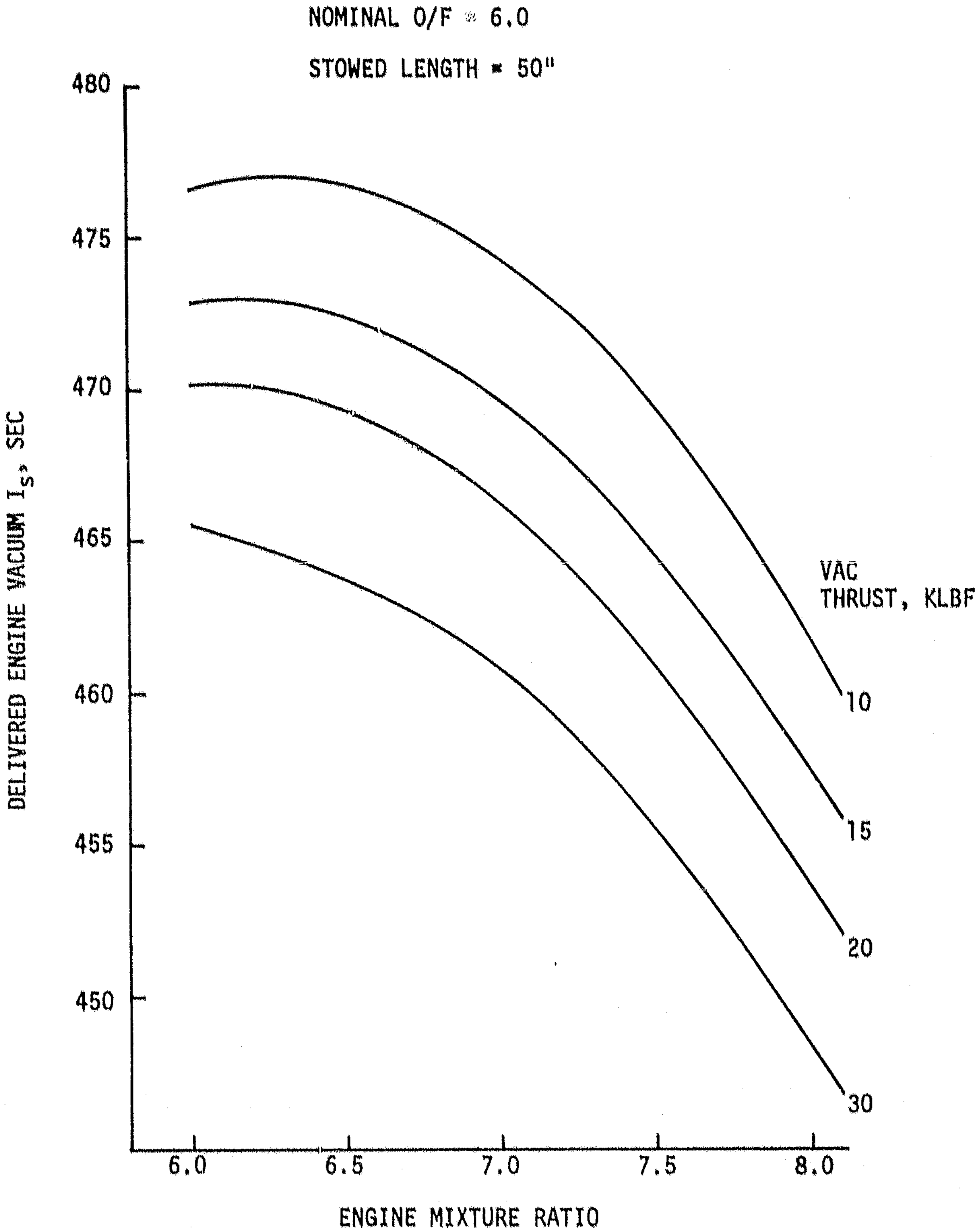


Figure 5. Advanced Expander Cycle Engine Performance at Design and Off-Design O/F (Stowed Length = 50 Inches)

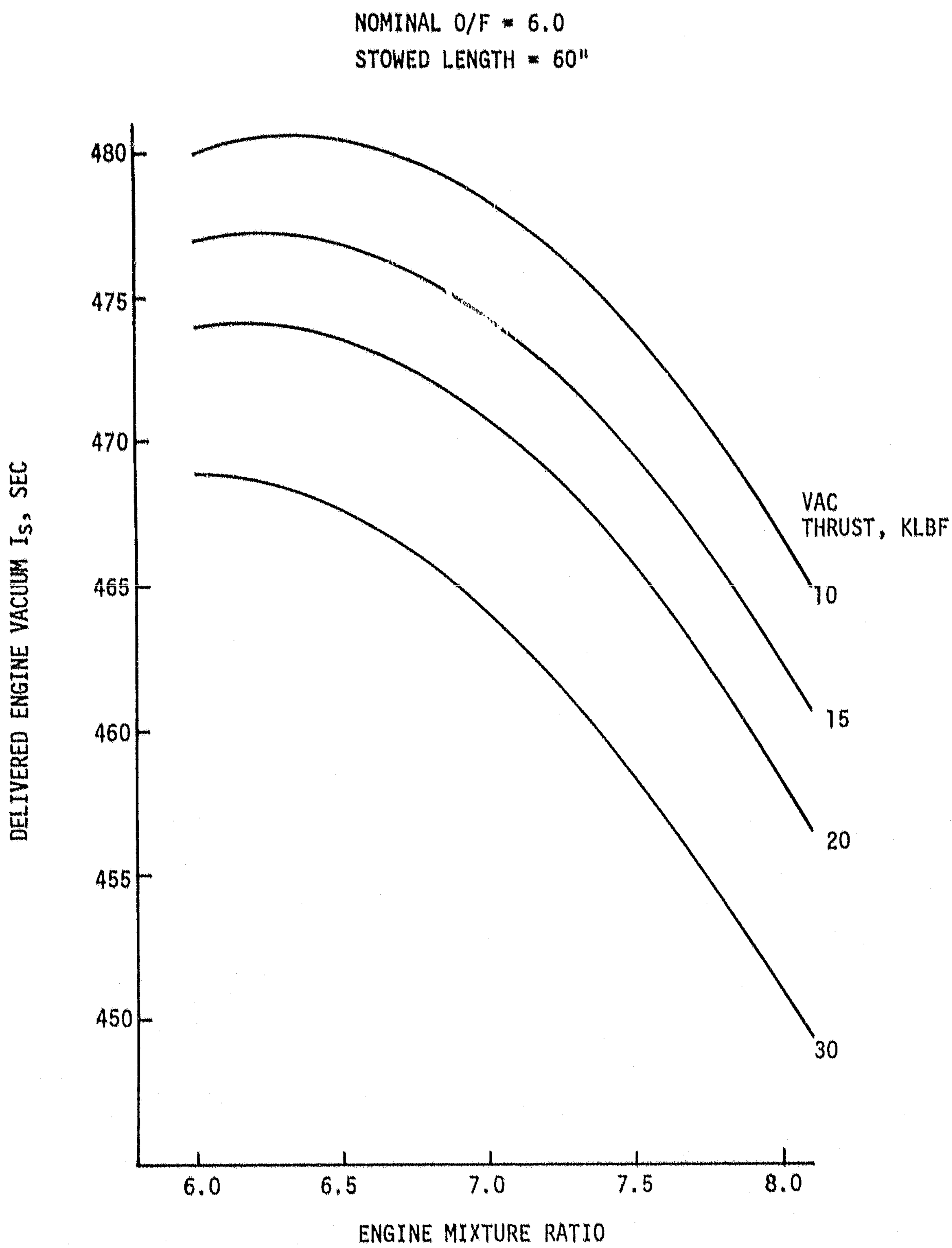


Figure 6. Advanced Expander Cycle Engine Performance at Design and Off-Design O/F (Stowed Length = 60 Inches)

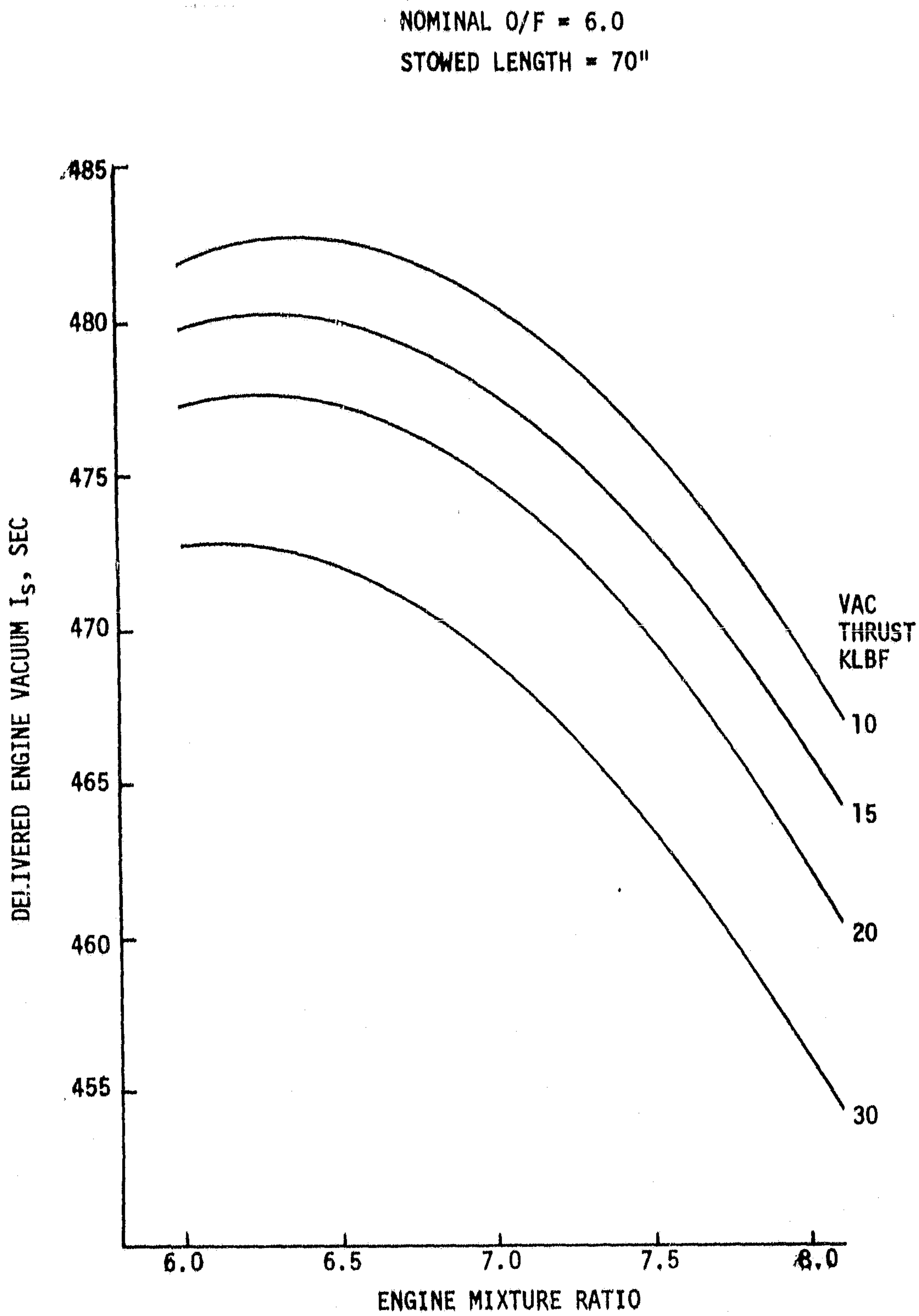


Figure 7. Advanced Expander Cycle Engine Performance at Design and Off-Design O/F (Stowed Length = 70 Inches)

As described in CPIA 246, the simplified procedure is "less accurate but quicker and less expensive than the rigorous" method and is thus appropriate for the preliminary design type analysis required for the generation of engine parametric performance data.

As used in the computer model, which generated the parametric performance data in Figures 3 through 7, the JANNAF Simplified Performance Prediction Methodology is represented by this expression:

$$Isp_d = Isp_{ode} \frac{\eta_{DIV} \cdot \eta_{ERE} \cdot \eta_{KIN}}{\left(1 + \frac{\Delta F_{BL}}{F}\right)} \quad (1)$$

where

Isp_d	=	Isp delivered (seconds)
Isp_{ode}	=	Isp ode (seconds)
η_{DIV}	=	Nozzle Efficiency
η_{ERE}	=	Energy Release Efficiency
η_{KIN}	=	Kinetic Efficiency
ΔF_{BL}	=	Boundary Layer Loss (lbf)
F	=	Delivered Thrust (lbf)

The accuracy of this simplified procedure however can be made nearly equivalent to that of the more rigorous procedures providing that the proper performance efficiencies are defined and shortcut calculational methods or correlations are calibrated or anchored over the parametric range under consideration. These procedures, once qualified, can then be used to develop reasonable predictions of attainable specific impulse for, in this case, the Advanced Expander Cycle OTV engine.

First, an analysis of the experimental performance results obtained with the ASE (Contracts NAS 3-17825 and NAS 3-19713) using the JANNAF Standard Analysis Techniques was performed to qualify or anchor the analytical performance procedures when applied to high pressure (>1000 psia), high area ratio (190-400), hydrogen/oxygen rocket engines. The performance efficiencies for both the ASE and RL-10 were then calculated using simplified techniques and compared to those resulting from the JANNAF rigorous performance prediction. The rigorous method used both the Two-Dimensional Kinetic with enthalpy addition and the BLIMP (Cebeci-Smith) boundary layer solution. The simplified model used ODK at propellant tank enthalpy with TBL-Chart/adiabatic wall conditions. While there were significant differences between the Isp_{TDK} and the boundary layer performance losses (ΔIsp_{BL}) between the two approaches, there was only 0.5 sec difference in the predicted specific impulse between the simplified and the TDK/BLIMP (Cebeci-Smith) results. Differences were used to calibrate the simplified technique, primarily in the calculation of the boundary layer loss.

Finally, this calibrated ALRC simplified performance model was used to calculate the delivered performance of both the ASE and RL-10 Derivative II Baseline Engine as a final check of its prediction capabilities. The simplified model provided calculated specific impulse values within 0.3% of the reported experimental values for both H_2/O_2 engine systems.

This same methodology described above was used in contracts NAS 3-21940 (Low Thrust Chemical Rocket Engine Study) and NAS 8-33574 (OTV Engine Point Design Study).

III TASK 6.2.15 - INTERMEDIATE THRUST OPERATION

The purpose of this task was first to evaluate the throttling capability of the currently baselined Advanced Expander Cycle engine at nominal thrusts of 10K, 15K and 20K lbf. Specifically, engine operation at thrust levels of 7000, 5000, 3000 and 2000 lbf was evaluated. Secondly, two different throttling methods were evaluated and compared. These two methods were:

- A bipropellant heat exchanger which gasifies the LO_2 so as to obtain GO_2/GH_2 injection at all thrust levels instead of LO_2/GH_2 injection.

- A dual manifolded oxidizer injector which provides higher element pressure drops, and hence improved chug stability, at low flow rates.

Both methods utilize swirl co-ax injector elements. Chug stability is a primary factor that limits the throttling range of a particular thrust chamber. Therefore, the emphasis was placed on evaluating the chug stability.

In all cases, a study guideline was that engine operation at rated thrust was not to be compromised with excessively high pressure drop injectors because of the throttling requirement. However, some other minor compromises are unavoidable. These include:

- Added weight of heat exchanger (for GO_2 generation) if used.

- Added weight of solenoid valves and plumbing (for utilization of dual manifold ox injector).

- Additional system pressure drops and higher pump discharge pressures attributable to use of heat exchanger.

The more important compromise to be avoided is excessive ox circuit injector pressure drop at rated thrust. Although this would enable deep throttling, it would also require a much higher ox pump discharge pressure and higher resulting turbine horsepower. This adversely affects the cycle power balance and would require an almost complete engine redesign (assuming the engine would even power balance).

On the basis of the comparative analysis, one of the two techniques evaluated was to be selected for comparison to the current method for obtaining low-thrust operation which is "kitting" of the engine. This low-thrust operation analysis is reported in Ref. 6. The low-thrust engine "kit" consists of replacable oxidizer injector elements, an orifice downstream of the coolant jacket to maintain the hydrogen coolant pressure above critical, and a recirculation line and valve around the oxidizer pump to avoid pump instability.

The comparisons also include impacts to the DDT&E, production and operations costs and schedules.

A. LO_2/GH_2 OTV ENGINE

The throttling capability of three OTV engine designs was evaluated to determine their inherent low thrust operating capability. The three engines have thrusts of 10K, 15K and 20K lbf. The assumed design conditions of each are listed in Table III. The baseline OTV engine concept uses LO_2/GH_2 injection with swirl co-ax elements.

The estimated throttling limits of the three engine design points are shown in Figure 8. These limits were determined on the basis of the 15K engine chug analysis reported in Refs. 6 and 7. The chug stability of this engine is governed by the oxidizer pressure drop (i.e., stiffness). The limiting oxidizer stiffness ($\Delta P_{ox}/P_c$) is a function of the combustion time lags and chamber L*. The 15K engine stiffness limit was found to be about 0.08 (Ref. 6). Since the chamber L* and injector design are the same for the 10K and 20K the limiting injector stiffness will be the same.

TABLE III
OTV ENGINE DESIGN POINTS

THRUST	10K	15K	20K
PC	1300	1200	1100
MR	6.0	6.0	6.0
ϵ_{NOZ}	792	473	322
ϵ_c	3.66	3.66	3.66
L'	18"	18"	18"
R_t	1.092"	1.395"	1.687"
A_t	3.75	6.11	8.941
R_c	2.09	2.67	3.23
$\dot{W}_T$	21.05	31.44	42.54
$\dot{W}_{ox}$	18.04	26.95	36.46
$\dot{W}_{H_2}$	3.01	4.49	6.08

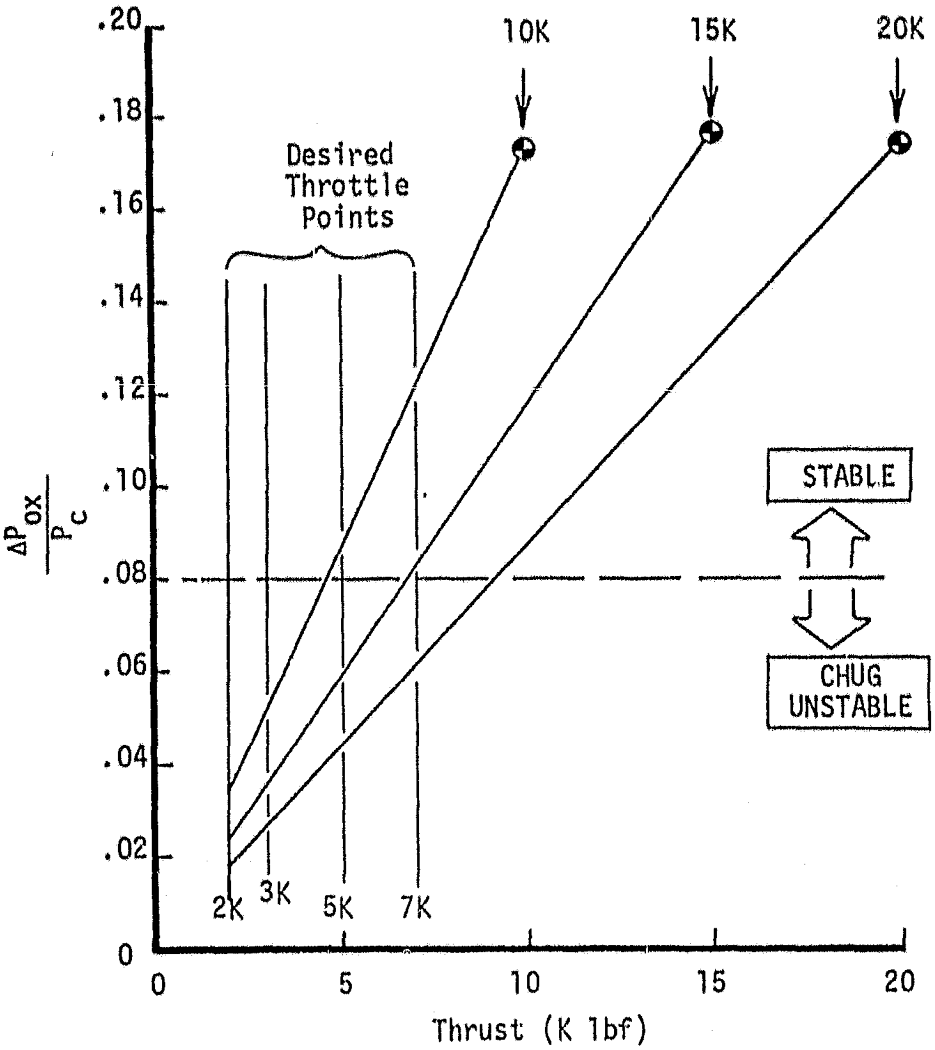


Figure 8. Predicted Baseline OTV Engine Throttle Limits

III Task 6.2.15 - Intermediate Thrust Operation (cont.)

The oxidizer injector stiffness is linearly related to the thrust as shown in Figure 8 if the performance is assumed to be constant. The results of this analysis show that the 10K engine can best achieve low thrust operation.

B. DUAL MANIFOLD LO_2/GH_2 ENGINE

A schematic of the dual manifold LO_2/GH_2 injector is shown in Figures 9 & 10. This is a modification of the baseline swirl coax injector in which the oxidizer is introduced through two flow circuits. This design is based on work performed on an ALRC Throttling Injector IR&D Program. The intent of the design is to permit throttling of the flow over a wide flow range while maintaining good atomization and injector stiffness ($\Delta P/P_c$). This design accomplishes this by varying the flow split between the axial flow circuit and the tangential flow circuit. The net result is a variation in orifice discharge coefficient. The predicted stiffness factor for this injector is shown in Figure 11. The stiffness of the baseline swirl injector is shown for comparison.

The predicted chug stability limit is shown in Figure 12. The chug Pc limit is much lower than the baseline swirl coax due to better stiffness characteristics and shorter combustion time lags due to better atomization at the low flow conditions.

C. GO_2/GH_2 ENGINE

The throttling limits of the GO_2/GH_2 engine are not limited by chug instability because the injector stiffness remain constant over the entire throttle range. This effort was limited to sizing the heat exchanger for the full thrust condition and determining the performance at several throttled down points.

The selected heat exchanger design is shown in Figure 13. It is a simple tube shell design. The hydrogen flows through the tubes and the oxidizer flows across the tube bundle. The propellant inlet and outlet conditions are shown in Table IV for several thrust levels.

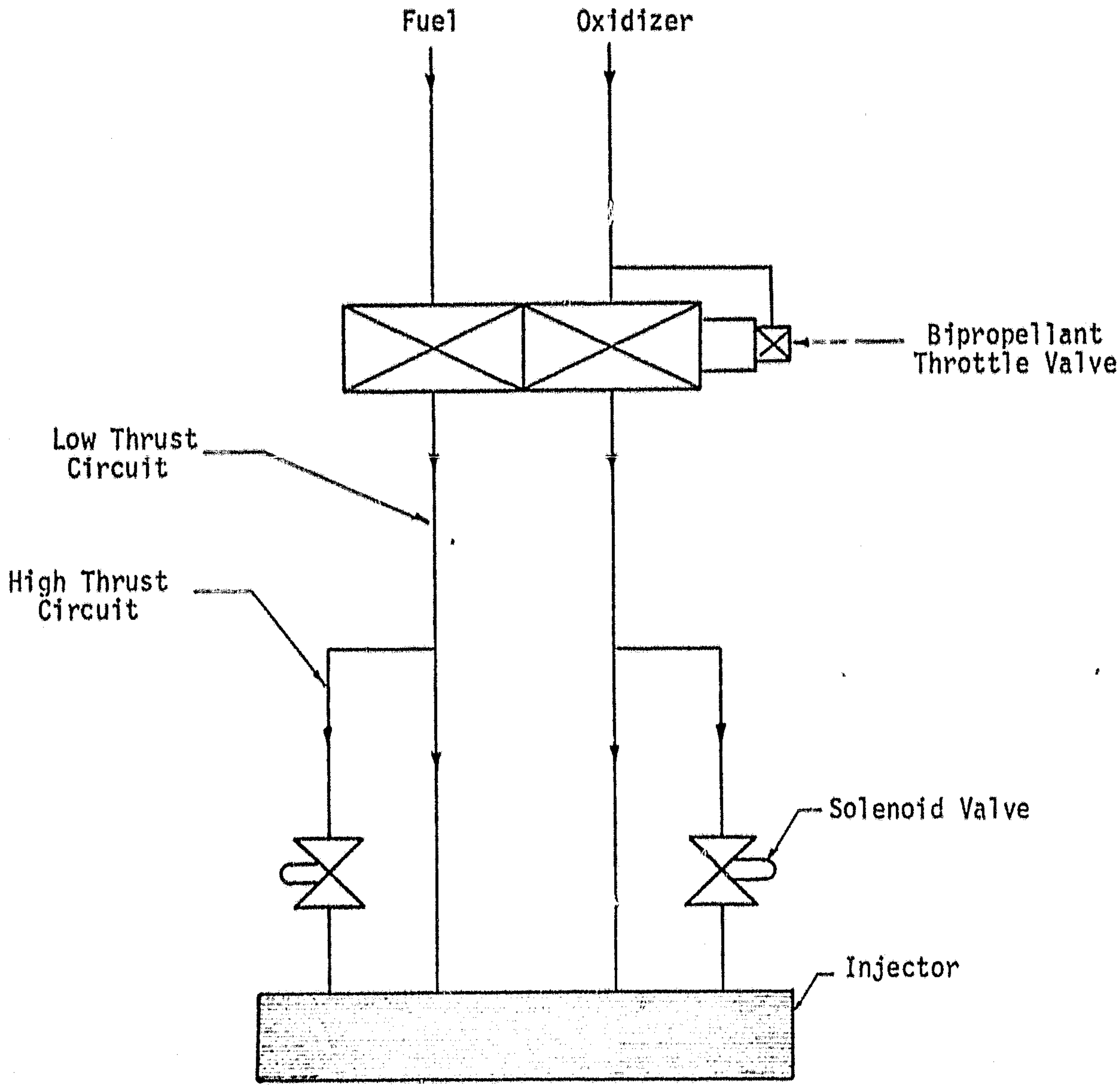


Figure 9. Dual Injector Valving Concept

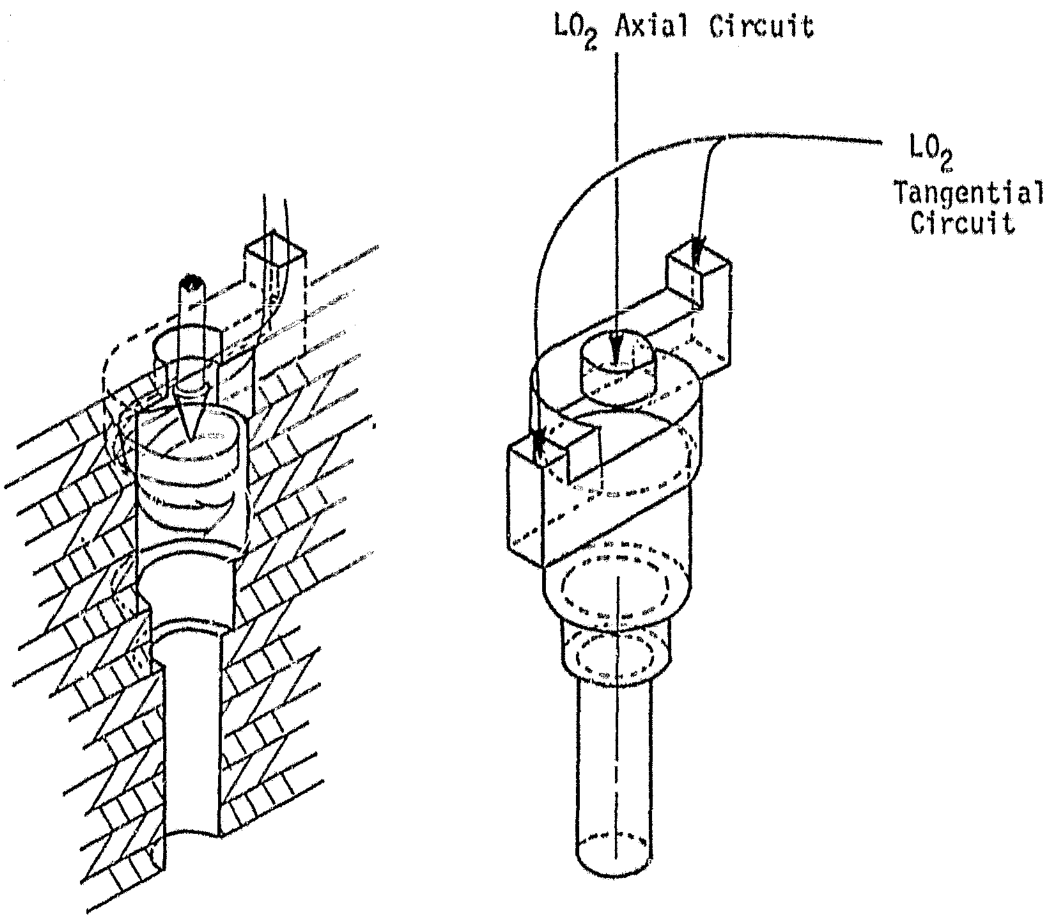


Figure 10. Dual Manifold Vortex Swirler Element

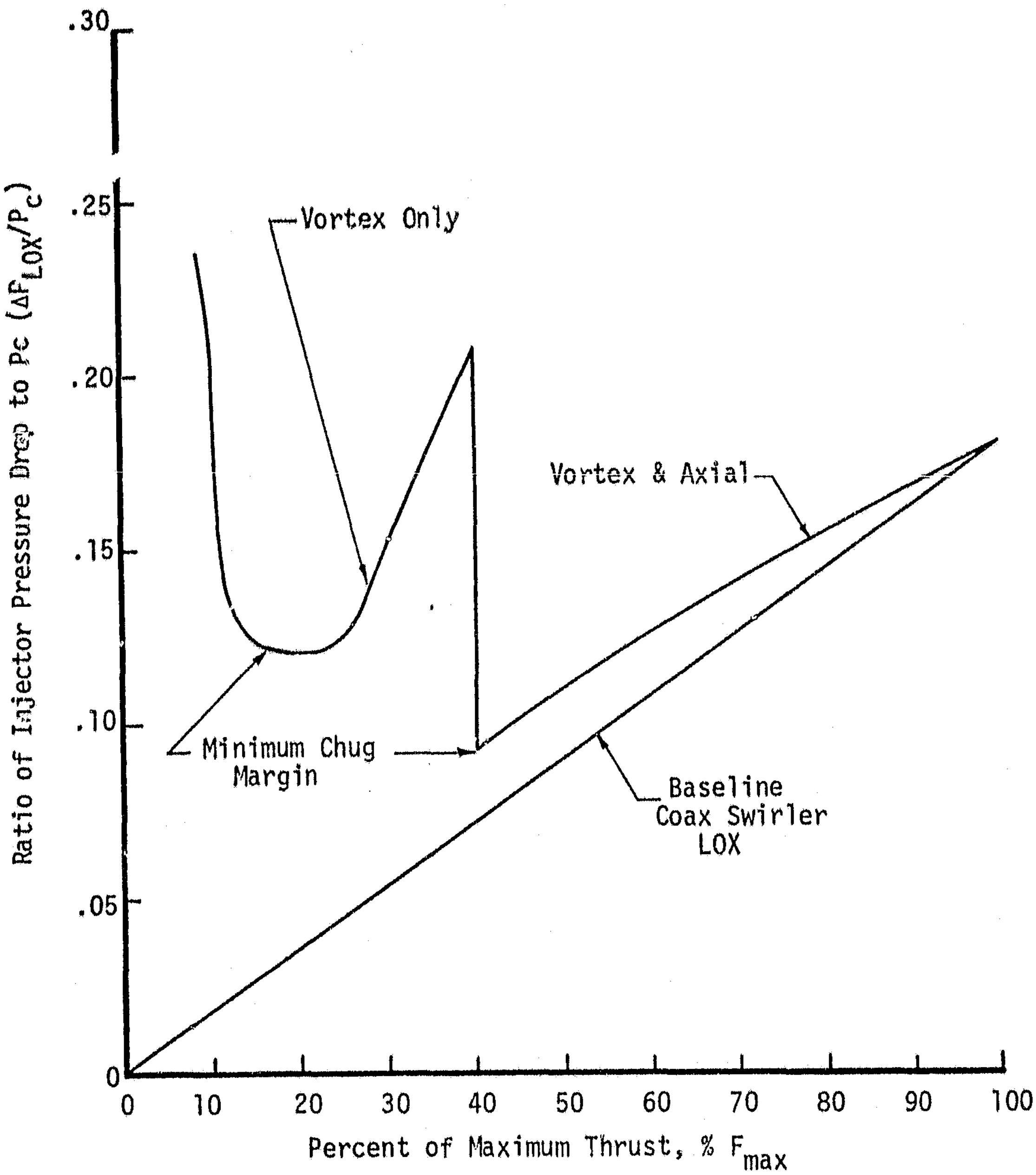


Figure 11. Dual Manifold Injector Stiffness

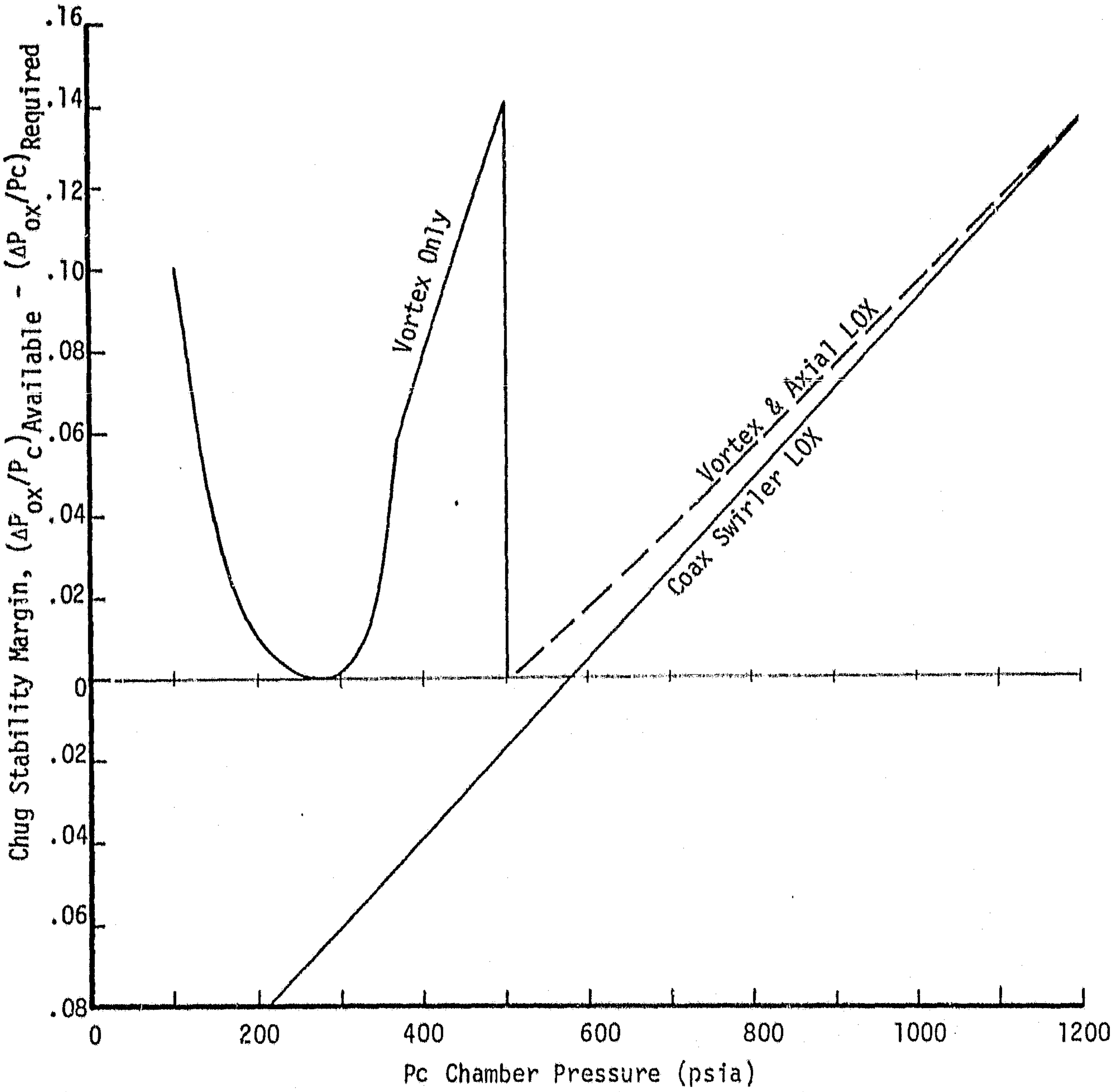
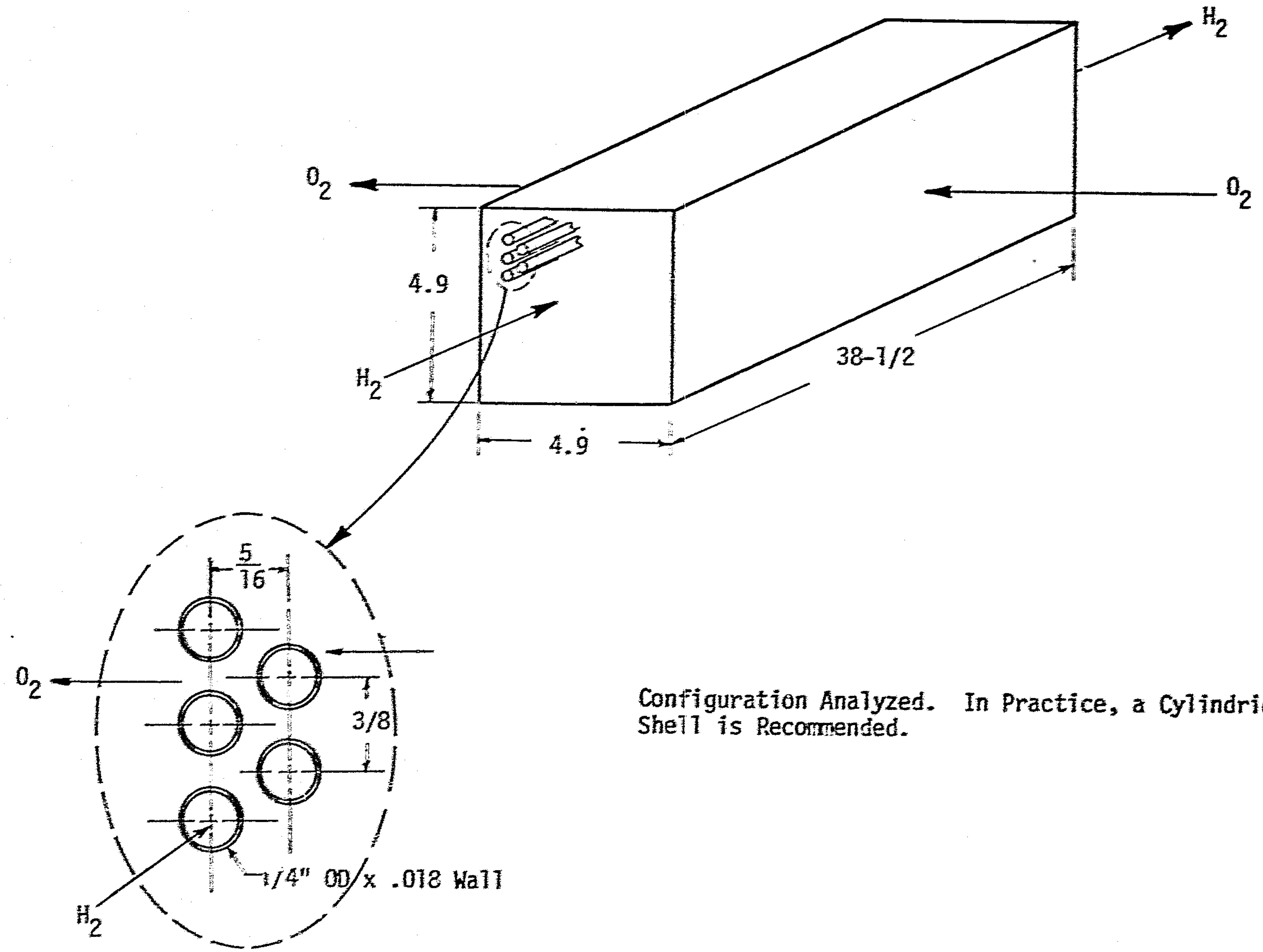


Figure 12. Chug Stability Limit for Dual Manifold Injector



Configuration Analyzed. In Practice, a Cylindrical Shell is Recommended.

Figure 13. Tube Bundle Heat Exchanger

TABLE IV

OTV HEAT EXCHANGER INLET/OUTLET CONDITIONS

HYDROGEN

% F	F lbf	T_{in} °R	T_{out} °R	ΔT_b °R	P_{in} psia	P_{out} psia	ΔP psi
100%	15K	442	358	84	1300	1295	5.4
26.7%	4K	586	394	192	386	384	2.24
13.3%	2K	670	437	233	198	197	1.12

OXYGEN

	15K	180	288	108	1420	1416	3.9
	4K	180	422	242	421	421	.009
	2K	180	508	328	192	192	.002

This heat exchanger would be placed downstream of the turbine so as not to lower the temperature of GH_2 entering the turbine. This last requirement is an enabling factor in achieving a reasonable engine system power balance at full or throttled thrust operation.

The gas-gas injector elementis shown in Figure 14. The energy release efficiency (ERE) was determined using the ALRC Generalized Gas/Gas Cold Flow and Combustion Mixing Computer Program. The predicted ERE is shown in Figure 15 for several thrust levels. The performance loss compared to the GH_2/LO_2 engine is about 3% at full thrust.

This performance loss is due primarily to increased propellant blowapart in the chamber as compared to GH_2/LO_2 injection. This same effect was observed experimentally with the Extended Temperature Range (ETR) cryogenic thruster (See Ref. 8). In that test program, with constant chamber length and diameter, a significant drop in ERE (4% to 8%) resulted when GO_2 injection replaced LO_2 injection.

Also, as Figure 15 shows, this performance loss falls off as engine thrust is throttled down. This happens primarily because the chamber pressure and resulting Reynolds Number in the chamber also decrease. As the Reynolds Number decreases, viscous effects begin to dominate with resulting improved propellant mixing. The improved mixing in turn results in higher ERE values.

D. COMPARISON OF THROTTLING METHODS

On the basis of the comparison described below the dual manifolded oxidizer injector concept has been selected over the heat exchanger, as the superior means of enabling engine throttling to 2000 lbf thrust for all three nominal thrust level engines (10K, 15K and 20K).

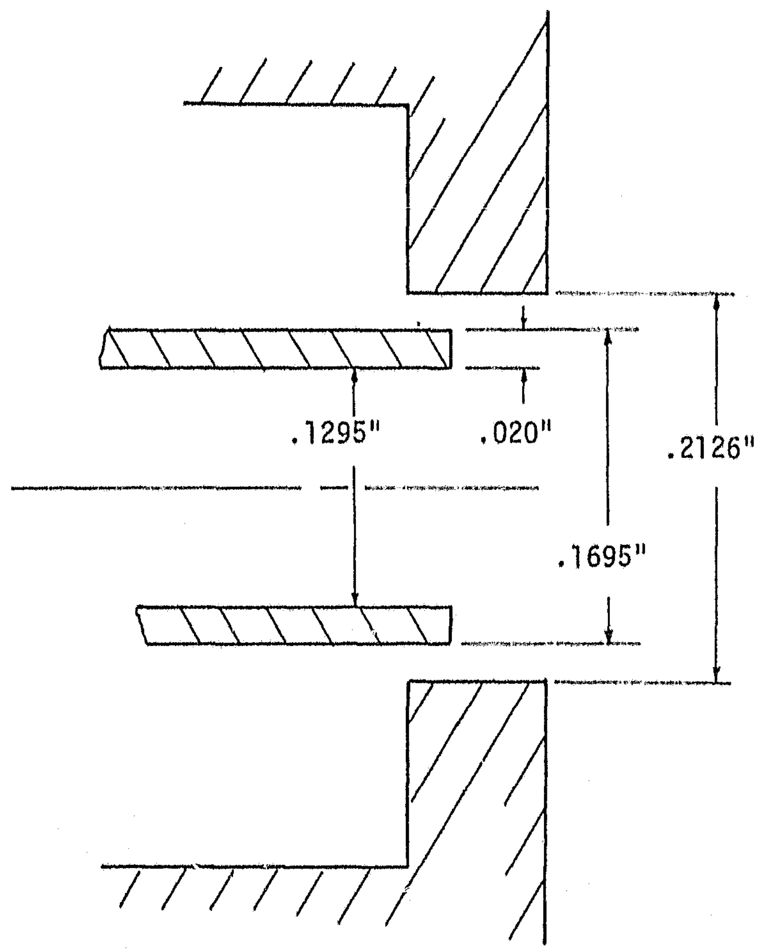


Figure 14.. Swirler Coax Injector Element

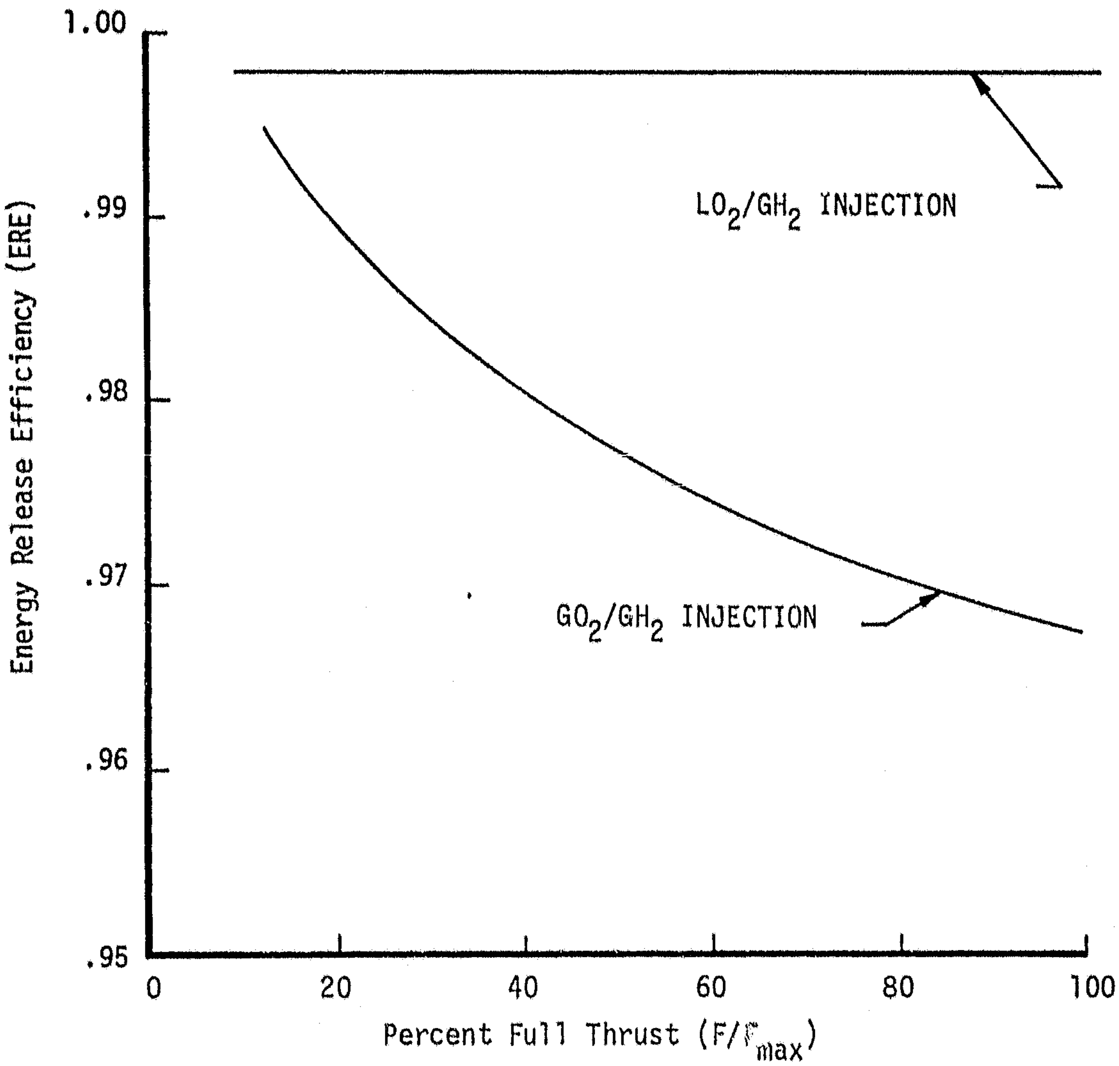


Figure 15. ERE vs Throttleable Thrust

A summary of this comparison is shown in Table V. Table V compares the two alternate throttling methods to the baseline "kitted" engine and to each other. As the table shows, the recommended method is essentially more expensive (greater DDT&E and production costs) but does not incur the significant payload penalties associated with the heat exchanger option. These considerations, and others, are described more in detail below.

1. Technical Comparison

The results of intermediate thrust operation analysis show that the LO_2/GH_2 propellant engine is limited to a throttle range of 100-47% of full thrust due to chug instability. Engine "kitting" enables stable operation at thrust levels of 7000, 5000, 3000 and 2000 lbf.

In comparing the dual manifolded ox injector concept to the GO_2/GH_2 heat exchanger concept, the most important difference is the ERE loss at rated thrust associated with the latter method. The 3% loss shown in Figure 15 is equivalent to approximately 14 seconds of Isp as compared to the dual manifolded concept. This Isp loss can be converted to an approximate OTV payload penalty by use of OTV "payload partials" presented in NASA Technical Memorandum TMX-73394. These partials are:

	All Propulsive OTV (APOTV)	Aeromaneuvering OTV (AMOTV)
$\frac{\Delta \text{Weight Payload}}{\Delta \text{Isp}}$, $\frac{\text{lb}}{\text{sec}}$	60	73
$\frac{\Delta \text{Weight Payload}}{\Delta \text{Weight Engine}}$, $\frac{\text{lb}}{\text{lb}}$	-1.1	-1.1

That is, for an All Propulsive OTV (APOTV), 14 seconds of Isp loss is equivalent to 840 lbs of delivered payload loss. This value is comparable to the weight of three astronauts with pressurized suits. Clearly, this is a significant payload penalty.

TABLE V

OVERALL THROTTLING METHOD COMPARISON

	BASELINE THROTTLING METHOD: BASELINE "KITTED" ENGINE ENGINE STOWED LENGTH = 60"			CONCEPT	ALTERNATE THROTTLING METHOD (DIFFERENCES COMPARED TO KITTED ENGINE): DUAL MANIFOLDED OX INJECTOR			HEAT EXCHANGER (GO_2/GH_2 INJECTION)		
Thrust, KLBf	10	15	20	Thrust, KLBf	10	15	20	10	15	20
Isp*, Seconds	480	477	474	ΔIsp, seconds	0	0	0	-14	-14	-14
Engine Weight, lbs	430	505	550	ΔEngine Wt, lb	29	35	40	35	50	65
DDT&E Costs, $ Millions of 1979 Dollars	208.7	218.2	228.4	ΔDDT&E Costs, $ Millions of 1979 Dollars	16.7	17.5	18.2	9.2	10.3	11.1
Total Production Costs, $ Millions of 1979 Dollars	110.6	120.2	129.7	ΔProduction Costs, $ Millions of 1979 Dollars	3.4	3.7	3.9	1.5	1.7	1.8
Payload**, lb (round trip from LEO to GEO)	13,000	13,000	13,000	ΔPayload, lb (round trip from LEO to GEO)	-32	-39	-44	-802	-785	-769

*Isp at rated thrust

**100,000 lb payload shuttle

The use of a heat exchanger also necessitates some additional system pressure drops in both the fuel and oxidizer circuits. As Table IV shows, these values for the fuel and oxidizer circuits at full thrust operation (15K lbf) are small. They are 5.4 psia and 3.9 psia for the fuel and oxidizer circuits, respectively. The corresponding discharge pressure rises for the fuel and oxidizer pumps are approximately 8 psia and 4 psia.

The heat exchanger represents an additional component in the system and hence additional weight. The heat exchanger for the 15K engine weighs approximately 50 lbs. Similarly, for the 10K and 20K engines the heat exchanger weight is approximately 35 lbs and 65 lbs, respectively. As the payload partials show, these weights would result in nearly equivalent payload losses. The dual manifolded oxidizer injector also incurs a weight penalty because of additional solenoid valves, propellant lines (refer to Figure 9) and injector manifolding. For the 15K engine, the total weight of these components is estimated to be 35 lbs. For the 10K and 20K engines this weight is estimated to be 29 lbs and 40 lbs respectively. Again, the payload losses would be nearly equal to these values. Therefore, it appears that the weight penalties for either option are similar based upon this preliminary conceptual analysis.

Another important consideration is cost. Basically, as will be explained in the next section on cost comparisons, the dual manifold injector concept requires only slightly more funds during the DDT&E phase. This is because the dual manifolded injector concept requires a more complex engine control system (more valves, lines, etc) in addition to the increased testing required by both concepts to insure the desired system reliability.

Both options are nearly equivalent in being capable of stepped and/or continuous throttling to 10% of rated thrust.

Finally, the two options may be compared on the basis of considerations more difficult to quantify. For instance, the dual manifolded injector requires, as noted above, more complex controls. This can degrade engine system reliability. However, the heat exchanger also represents increased complexity. Perhaps more importantly, the heat exchanger is a potential bipropellant leak path not present in the dual manifolded system.

The dual manifolded oxidizer injector concept was also compared to the baseline "kitted" engine concept.

The dual manifolded engine concept results in increased weight compared to the"kitted" engine as previously discussed. In addition, the necessity of a more complex control system results in increased DDT&E costs. In the case of the kitted engine, the control system does not change when the injector is fitted with different size oxidizer elements. However, the engine must still be tested to insure chamber/injector compatibility with both the standard (full thrust) oxidizer elements and low thrust (2000 lbf) oxidizer elements (kitted).

A second consideration is the fact that the kitted engine will operate stably only over discrete ranges. That is, the unkitted engine can only throttle down to 47% of full thrust as a minimum. If the engine is kitted for 2000 lbf thrust operation then the operating range can be enlarged around this point. This consideration makes the kitted engine less flexible for mission planning. The dual manifolded injector concept on the other hand will allow throttling to 10% of rated thrust during a single mission. This provides options not available with the kitted engine.

2. Cost Comparison

The increases in DDT&E, production and operations costs and scheduling changes attributable to the dual manifolded oxidizer injector and heat exchanger concepts are discussed here. These cost increases are also compared to the kitted engine concepts. All dollar values, with the exceptions noted, are cost increases to be added to figures already presented for the Advanced Expander Cycle engine in the initial Phase A Final Report (See Ref. 9) and Phase A Extension 1 Final Report (See Ref. 10). Also, all costs are in 1979 dollars to maintain consistency with the original data.

The increases to DDT&E costs, by Level 4 WBS identification numbers, required by both throttling concepts are listed and compared to the kitted engine in Table VI below. The variation with thrust level is assumed to be the same as the total engine system DDT&E cost variations with thrust (See Ref. 9 and 10) . Table VI indicates that the cost of developing the dual manifolded injector with the resulting increased testing is nearly equivalent to the cost of kitting when only the injector and the engine testing are considered. The DDT&E cost increases for engine kitting, shown in Table VI, are all inclusive. That is, they include all costs to design, develop and test the low-thrust injector, demonstrate injector/chamber compatibility and to flight certify the low-thrust engine. This program is assumed to be conducted in parallel with the rated thrust engine development. The major cost difference then between kitting and using the dual manifolded oxidizer injector is due to the increased controls cost required by the dual manifolded injector.

All of these cost increases should be compared to the total engine system DDT&E costs also listed in Table VI.

The increase in production costs resulting from use of either of the two throttling concepts are outlined in Table VII, similar in format to Table VI. Again, the cost increase variation with thrust level parallels that for total production cost variation with thrust presented in Ref. 10. As mentioned earlier, the production costs for the dual manifolded oxidizer injector are higher than the heat exchanger. These cost increases are still minor (approx. 3%) when compared to the total production costs of the baseline engine.

The increase in operations costs due to the implementation of any throttling technique (kitting, heat exchanger or dual manifolded oxidizer injector) are negligible.

TABLE VI

DDT&E COST INCREASES ($ MILLIONS OF 1979 DOLLARS)

THROTTLING CONCEPT		ENGINE THRUST LEVEL, LBF		
Level 4 WBS No. & Description		10K	15K	20K
DUAL MANIFOLDED OX INJECTOR				
1.1.2.1	Injector	4.4	4.6	4.8
1.1.2.6	Ass'y & Checkout	5.5	5.8	6.0
1.1.5.1	Engine Controller & Harness	1.0	1.0	1.1
1.1.5.2	Control Valves	0.7	0.8	0.8
1.1.5.3	Instrumentation & Harness	0.3	0.3	0.3
1.1.5.4	Ass'y & Checkout	0.8	0.8	0.9
1.1.10.1	Development Testing	1.4	1.5	1.5
1.1.10.2	PFC Testing	1.2	1.2	1.3
1.1.10.3	FFC Testing	1.4	1.5	1.5
	TOTAL:	16.7	17.5	18.2
HEAT EXCHANGER				
1.1.7	Propellant Systems	5.2	6.1	6.8
1.1.10.1	Development Testing	1.4	1.5	1.5
1.1.10.2	PFC Testing	1.2	1.2	1.3
1.1.10.3	FFC Testing	1.4	1.5	1.5
	TOTAL:	9.2	10.3	11.1
KITTING				
	TOTAL	14.3	15.0	16.4
TOTAL ENGINE SYSTEM (UNKITTED) DDT&E COSTS:		194.4	203.2	212.0

TABLE VII

PRODUCTION COST INCREASES ($ MILLIONS OF 1979 DOLLARS)

THROTTLING CONCEPT		ENGINE THRUST LEVEL, LBF		
Level 4 WBS No. & Description		10K	15K	20K
DUAL MANIFOLDED OX INJECTOR				
1.2.1.2 Injector		1.65	1.81	1.93
1.2.1.3 Controls		1.70	1.87	1.99
	TOTAL:	3.35	3.68	3.92
HEAT EXCHANGER				
1.2.1.5 Heat Exchanger		1.54	1.69	1.80
	TOTAL:	1.54	1.69	1.80
TOTAL ENGINE SYSTEM (UNKITTED) PRODUCTION COSTS:		110.6	120.2	129.7

The DDT&E schedule analysis for the heat exchanger and dual manifolded oxidizer injector concepts resulted in a additional time requirement of approximately one month to allow for the increased engine systems testing required. The component testing required is assumed to occur in parallel with other component testing.

IV TASK 6.2.16 - ENGINE OPERATION FOR AN AEROBRAKING OTV (ABOTV)

The objective of the Aerobraking Analysis was to evaluate the performance of a 15K lbf OTV engine during the ABOTV maneuver. Evaluations of operation with H_2 cold flow, at tank head idle (THI) mode and at pumped idle (PI) mode were made.

The baseline 15K engine nominal operating conditions are as listed in Table III. The Aerobraking maneuver duty cycle was provided in the contract SOW and is repeated in Table VIII. It was assumed that the dynamic pressures listed in the table exist at the nozzle exit plane. The nozzle extension is retracted to an expansion ratio of 172:1.

The H_2 cold flow performance was determined using isentropic flow relationships. The hydrogen inlet conditions vary with time during the chilldown. Therefore, the performance was determined as a function of inlet temperature as shown in Figure 16. The performance drops off both due to the reduction in temperature and due to a reduction in cold flow chamber pressure. The reduction in Pc causes nozzle flow separation to occur earlier in the nozzle which reduces the effective expansion ratio. The predicted separation area ratios are also indicated in Figure 16. The flow separation criteria specified in Ref. 11 (i.e., $P_e < .3\ P_a$) was used to determine the point of separation.

The nozzle exit pressures corresponding to the truncated nozzle (ε = 172:1) and the point of flow separation are listed in Table IX. As the table shows, the PI mode is actually slightly overexpanded for the assumed ambient pressure

TABLE VIII

ENGINE OPERATING DUTY CYCLE
ABOTV AEROBRAKING MANEUVER

OPERATING MODE	ALTITUDE FEET	AVE. DYN. PRESSURE PSIA	DURATION SECONDS
H2 Only	400K	.0035	60
Tank Head Idle	292K	.0347	30-50
Pumped Idle	265K	.0694	0-40
Tank Head Idle	262K	.0347	120

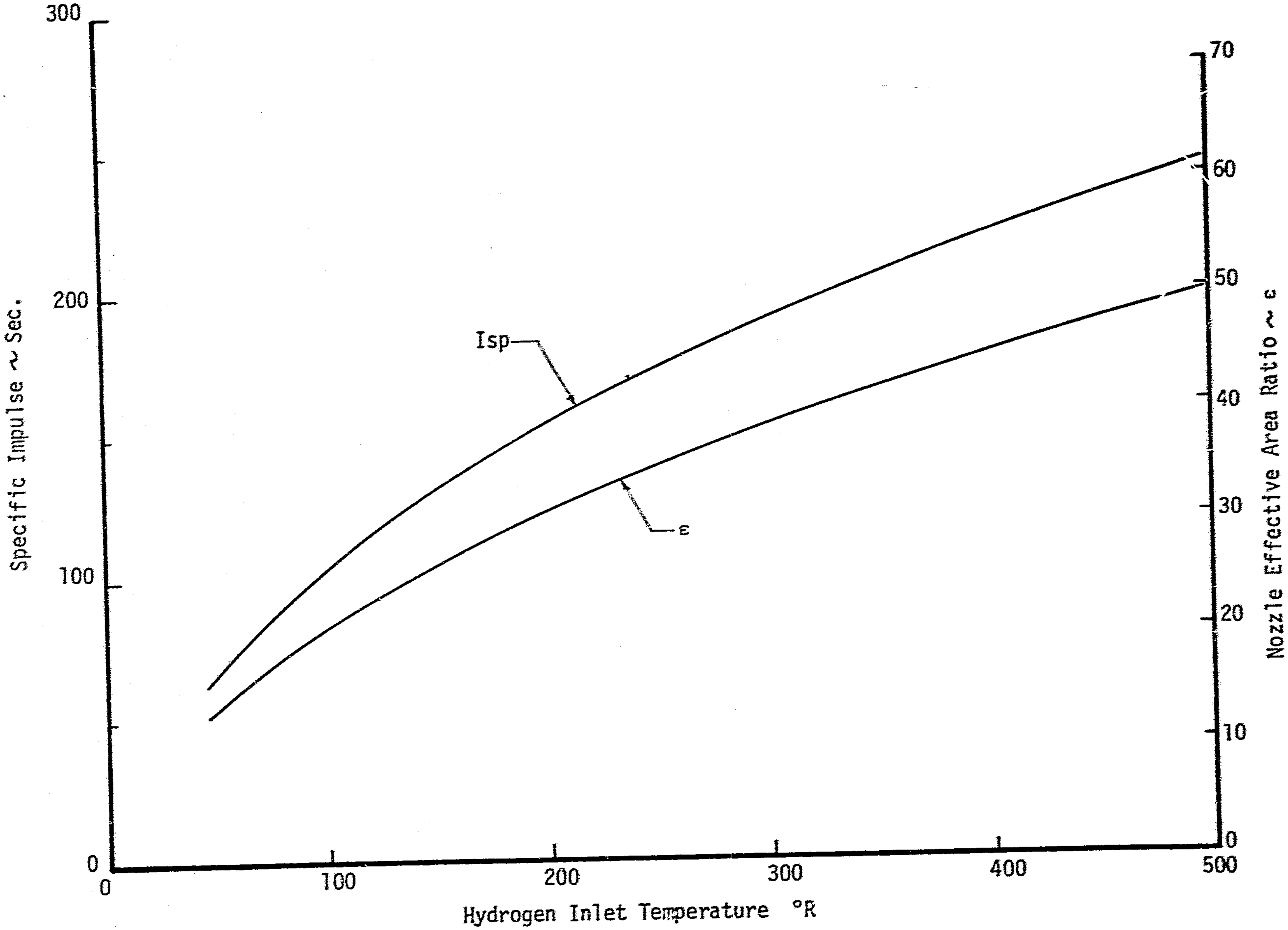


Figure 16. Effect of Hydrogen Inlet Temperature on Performance

TABLE IX

NOZZLE EXIT PRESSURES FOR ABOTV OPERATION

$\epsilon_{Nozzle} = 172:1$

Operation Mode	P_c (psia)	$P_{ambient}$* (psia)	$P_{separation} = .3\ P_{ambient}$ (psia)	P_{nozzle} exit $\epsilon = 172:1$ (psia)	ϵ_{eff} (ϵ at which separation occurs)	P (psia) at ϵ_{eff}	GH_2 Inlet Temp. °R)
Pump Idle (PI)	120	.0694	.0208	.0816	172	.0816	>500
Tank Head Idle (THI)	6	.0347	.0104	.003	60	.0104	>500
GH_2 Chilldown	.7	.0035	.0011	.0003	50	.0011	500
GH_2 Chilldown	.4	.0035	.0011	.0002	28	.0011	175
GH_2 Chilldown	.2	.0035	.0011	.0001	13	.0011	37

*See Table VIII (From SOW)

The GH_2 is actually under the vapor dome at a temperature of 37°F and pressure of 0.2 psia. However, it is predicted that the chilldown sequence, starting with a "hot" engine (i.e., chamber hardware temperature $\geq$ 500°R), will not be long enough for the GH_2 to drop to 37°R at the injector inlet. Rather the minimum GH_2 inlet temperature is estimated to be closer to 150°R at a pressure close to 0.4 psia. The GH_2 will be a gas at those conditions.

The tank head idle mode and pumped idle mode performance was determined using the ALRC calibrated simplified JANNAF procedures. The performances for the Tank Head Idle and Pumped Idle modes are shown in Table X. The operating Pc and MR for the Tank Head Idle mode are the same as those specified in the original Phase "A" OTV study contract (see Ref. 2).

TABLE X

PREDICTED PERFORMANCE FOR THE ABOTV
15K LBF BASELINE ENGINE

Operation Mode	Temp (°R)	P_c (psia)	Effective Nozzle Area Ratio	Engine Mixture Ratio	Engine Thrust (lbf)	Delivered Specific Impulse (Seconds)	Flowrates lb/sec		
							Total	O_2	H_2
Pump Idle (PI)	NBP in Tanks	120	172	6.0	1402	440.8	3.18	2.73	.45
Tank Head* Idle (THI)	NBP in Tanks	6	60	4.0	60	394.7	.15	.12	.03
GH_2 Childown*	500	0.7	50	-	6.6	246.4	-	-	.027
GH_2 Childown	175	0.4	28	-	3.7	141.6	-	-	.026
GH_2 Childown	37	0.2	13	-	1.8	62.8	-	-	.029

*Flow separation occurs in nozzle since $P_e < .3\ P_a$, see Table IX

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