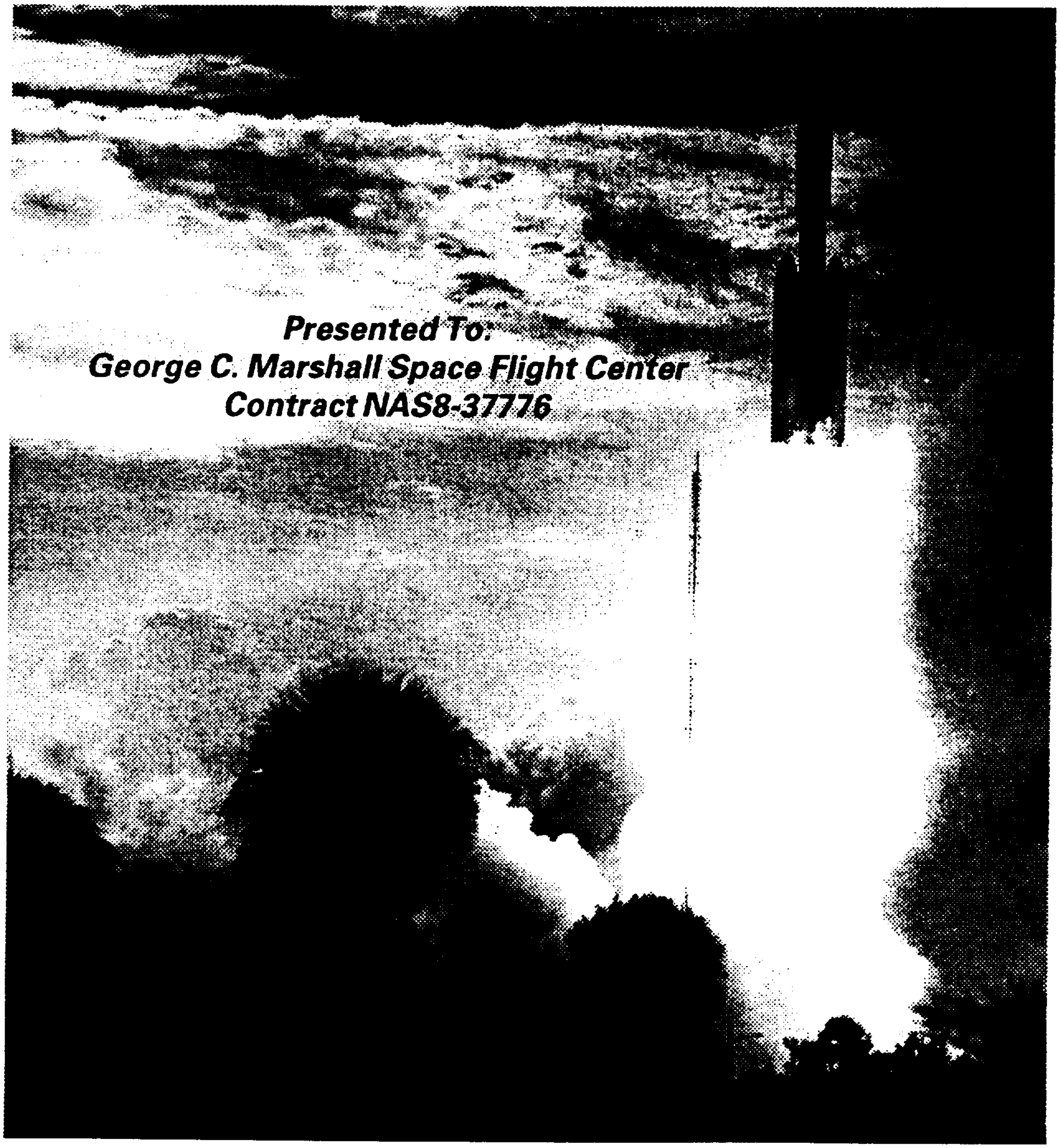
Hybrid Propulsion Technology Program
Final Report



Presented To:
George C. Marshall Space Flight Center
Contract NAS8-37776

Atlantic Research Corporation
Virginia Propulsion Division
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1.0 INTRODUCTION

This report describes work accomplished during the Hybrid Propulsion Technology Program, contract number NAS8-37776. The program objective was to identify the technology to enable application of hybrid propulsion to manned and unmanned space launch vehicles. The Hybrid Propulsion Technology Program is designed to identify the necessary technology in Phase 1, acquire that technology in Phase 2, and demonstrate it in a large subscale system in Phase 3. The scope of this report is to cover the tasks completed in Phase 1.

Atlantic Research Corporation (ARC) proposed two design concepts in response to the request for proposal (RFP) MSFC 8-1-8-EP. The first was a hybrid propulsion system utilizing the classical method of regression (classical hybrid) resulting from the flow of oxidizer across a fuel grain surface. The second system utilized a self-sustaining gas generator (gas generator hybrid) to produce a fuel-rich exhaust that was mixed with oxidizer in a separate combustor. Both systems offered cost and reliability improvements over the existing solid rocket boosters and proposed liquid boosters.

The contracted ARC program was designed to address the selection of one of the hybrid concepts by developing a booster point design for each propulsion system. The designs were evaluated using life cycle cost and reliability. Our program consisted of: (1) identification and evaluation of candidate oxidizers and fuels; (2) preliminary evaluation of booster design concepts; (3) preparation of a detailed point design including life cycle cost and reliability analyses; (4) identification of those hybrid specific technologies needing development; and (5) preparation of a technology acquisition plan and large-scale demonstration plan.

In addition to the expertise provided by ARC, the Preliminary Design Group of the Boeing Aerospace and Electronics Company was placed under subcontract to provide system integration and life cycle cost analysis; AiResearch Los Angeles Division and the Fluid System Division of Allied-Signal Aerospace Company were placed under subcontract to provide turbomachinery design and performance data, and liquid injection thrust vector control designs, respectively; ARC Liquid Propulsion provided the oxidizer delivery system design trades; and the Aerotherm Division of the Acurex Corporation (Huntsville

Operations), under subcontract to ARC, provided additional information on turbopumps and controls.

During the program, ARC evaluated eight classical hybrid and gas generator hybrid conceptual designs. ARC selected the gas generator hybrid with liquid oxygen oxidizer (LOX) because: (1) it provided a lower life cycle cost for 150 missions over 10 years of operation ($11.4 to $15.3 billion) than the classical hybrid ($12.9 to $19.2 billion); (2) had the same calculated reliability (R = 0.998); (3) offered an approach to solve the historical scaling uncertainty associated with the classical hybrid; and (4) offered all of the operational advantages historically associated with liquid propulsion.

2.0 TECHNICAL DISCUSSION

To encompass a range of possible vehicle system requirements, two hybrid rocket motors were conceptualized: a full-size motor which produces 13.3 10^6 N (3.0 x 10^6 lb_f) of thrust; and a single motor, four of which, in combination, produce the same thrust. Each motor should meet the following requirements:

- Concepts shall use thrust vector control (TVC).
- Concepts shall not use asbestos-containing materials.
- Concepts shall utilize active control system for performance, thrust imbalance, propellant utilization, and all transients.
- Concepts shall minimize environmentally degrading exhaust products.
- Concepts shall maximize shelf life.
- Solid propellant grain shall extinguish when the fluid propellant flow is stopped; no restart capability.
- Safety and reliability requirements shall be identical for manned and unmanned systems.
- Recoverable and reusable concepts versus expendable concepts shall be evaluated.

During the program, ARC evaluated eight classical hybrid and gas generator hybrid designs (Table 1). The concepts were configured from the components listed in Table 2, and booster weights were estimated to calculate cost and reliability. As a result of our conceptual studies, we selected the

Table 1. Concept Summary Overview.

ID Number	Cycle	Oxidizer	Motor Case	Oxidizer Tank	Oxidizer Feed System	TVC	Recovery
1	GG	H_2O_2	Carbon Epoxy	Carbon Epoxy	Pressure Fed	Flexseal	Expendable
1T	GG	H_2O_2	Carbon Epoxy	Al-Li	Pump Fed	Flexseal	Recoverable
1A	GG	LOX	Carbon Epoxy	Carbon Epoxy	Pressure Fed	LITVC	Expendable
1AT	GG	LOX	Carbon Epoxy	Al-Li	Pump Fed	LITVC	Recoverable
2	Classical	H_2O_2	Carbon Epoxy	Carbon Epoxy	Pressure Fed	Flexseal	Expendable
2T	Classical	H_2O_2	Carbon Epoxy	Carbon Epoxy	Pump Fed	Flexseal	Expendable
2A	Classical	LOX	Carbon Epoxy	Carbon Epoxy	Pressure Fed	LITVC	Expendable
2AT	Classical	LOX	Carbon Epoxy	Carbon Epoxy	Pump Fed	LITVC	Expendable

Table 2. Design Selection Summary.

Component	Selection	Also Considered	Selection Rationale
Concept	Gas Generator	Classical	Lower Life Cycle Cost (LCC) Lower Development Risk
Oxidizer	LOX	Hydrogen Peroxide	Currently Used Lower LCC
Oxidizer Feed System	Turbopump	Pressure Fed	Lower LCC Offers Pump Out
Gas Generator Fuel	ARCADENE 399C	ARCADENE 246B and Others (see pgs 6-8)	Higher Specific Impulse High Ejection Efficiency
Gas Generator Case	Carbon/Epoxy Composite	D6AC Steel	Lower LCC Improved Manufacturing
Thrust Chamber	Ablative	Regenerative Cooled	Improved Reliability Lower LCC
Thrust Vector Control	LITVC	Flexseal	Higher Reliability Lower LCC
Oxidizer Tank	Al-Li	Carbon/Epoxy	Recommended by ALS Contractors
Oxidizer Pressurization System	Tridyne	Cold Gas and Others, (see pgs 40-46)	Simplest System Lowest LCC
Recovery System	Expendable	Recover Nested High Cost Items	Simplest System Lowest LCC

pump-fed gas generator hybrid for our baseline point design. This hybrid offers the following advantages:

- Calculated reliability of 0.998.
- Reduced number of critical parts; only one cryogen (LOX) compared to liquid boosters.
- $11.4 billion life cycle cost for 150 missions over 10 years of operation.
- Engine shutdown and throttling capability.
- Mission accomplished even with the loss of one turbopump.
- On-pad abort.
- 13,608 kilograms (46 percent) shuttle payload improvement (over ASRM boosters).

In addition to the features shown above, the gas generator hybrid approach also offers an approach to solve the historical scaling uncertainty associated with the classical hybrid; i.e., the complex interaction between the oxidizer flow and the changing (regressing) fuel grain.

The oxidizer in the classical hybrid flows down the free stream portion of the fuel grain ports and reacts with the fuel at the edge of the boundary layer while fuel materials are ejected from the grain surface. The heat released in this oxidizer reaction controls the rate of fuel ablation and regression. Thus, the oxidizer/fuel ratio and total mass and energy generation rate are tied to the boundary layer because it controls the rate of mixing and the rate of heat feedback. Discussions with Professor Robert Beddini, a leading expert in the analysis of rocket motor port flows, confirmed ARC's analysis that there is a great deal of uncertainty associated with the scaling of such a boundary layer process since the combustion phenomena do not directly scale with port size.[1] Professor Beddini also pointed out that there is a strong dependency of fuel regression rate with distance down the grain port. As the flow moves down the port, the boundary layer thickens, and the ejected fuel and accumulated combustion products lead to reduced heat

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1. Personal Communication, Beddini, R. A., Dept. of Aeronautical and Astronautical Engineering, University of Illinois, Urbana, IL, April 1989.

feedback, yet higher mass flux, resulting in uncertain localized regression rates. In addition, the low regression rate historically associated with classical hybrids [0.003 - 0.01 cm/sec (0.001 - 0.004 in/sec)] require complex grain designs to produce sufficient surface area to generate the required mass and energy release rates. The complex grain designs increase the probability of sliver ejected from the nozzle during grain burnback and reduce the volumetric packing efficiency resulting in large booster designs.

2.1 Oxidizer Evaluation

ARC performed oxidizer evaluations, fuel evaluations, propulsion conceptual studies, developed point designs for two sizes of booster, and performed life cycle cost studies and reliability analyses. The results of these studies are discussed in the following sections. Two oxidizers were considered to be viable candidates for the hybrid booster, liquid oxygen (LOX), and 95-percent hydrogen peroxide (H_2O_2). Both oxidizers were evaluated on the basis of safety, cost, and performance impacts. Alternative oxidizers such as nitrogen tetraoxide were also considered, but quickly ruled out because of system safety. ARC selected LOX as the oxidizer for the classical and gas generator hybrid point designs as a result of our evaluation.

LOX as an oxidizer is well known for the performance it provides with any fuel. Its use and handling are well understood and are currently practiced. Most of the core vehicle designs to be incorporated with a hybrid booster use LOX; therefore, if LOX was used for the hybrid, there would be system commonality and reduction in facility requirements. LOX is relatively inexpensive; however, the complexity of handling and designing for a cryogenic fluid cannot be minimized.

Hydrogen peroxide has been proposed by ARC for use in other hybrid propulsion systems. It has not been extensively used in the last 20 years and would require training to enable its use. Since H_2O_2 is a monopropellant, it has applications to drive turbopumps, as an injectant for thrust vector control, and an energy source to pressurize the helium expulsion tank. The density of H_2O_2 is 24 percent higher than LOX, which results in a smaller booster at the same mixture ratio; the flame temperature at the optimum mixture ratio is 978K (1,760°R) lower than a LOX system, reducing the thermal protection requirements. The disadvantages of using H_2O_2 are as numerous as the advantages. Hydrogen peroxide of the purity required for use in the booster is not currently manufactured in the United States and has a higher ingredient cost. Peroxide can also decompose spontaneously due to contamination; the specific impulse (I_{sp}) of a hybrid system using H_2O_2 is 9 percent lower than LOX for the classical hybrid, and 6 percent lower for the gas generator hybrid; and the operations costs for H_2O_2 are greater than LOX due to the training requirements and lack of personnel experience.

2.2 Fuels Evaluation

A number of fuels were evaluated using thermochemical calculations and trajectory analysis for both the classical hybrid and the gas generator hybrid concepts. Hybrid fuels evaluation included definition of the theoretical vacuum I_{sp} and the theoretical characteristic exhaust velocity (C*) of the fuel and oxidizer combination as a function of mixture ratio, quantity of propellant to provide the required vacuum total impulse, and estimation of the relative payload capability.

2.2.1 Gas Generator Fuels

The fuels evaluated for the gas generator hybrid (Table 3) are derived from propellant formulations. Requirements for these fuels, established by the program statement of work (SOW) and by ARC, include: (1) total extinguishment below 2.06 MPa (300 psia); (2) burning rates of 0.76 to 1.27 centimeters-per-second (0.3 to 0.5 in/sec) at 6.88 MPa (1,000 psia); and (3) production of less than 1 percent hydrogen chloride (HCl) emissions in the exhaust.

ARC selected ARCADENE 399® [34 percent polystyrene, 25 percent carboxyl-terminated polybutadiene (CTPB), 37 percent ammonium perchlorate (AP), 4 percent iron oxide (Fe_2O_3)] as the initial formulation to be evaluated because it has: (1) high theoretical specific impulse; (2) demonstrated burning rate tailorability of 0.51 to 2.03 cm/sec (0.2 to 0.8 in/sec); and (3) a high ejection efficiency. This fuel-rich formulation demonstrated good performance in the Fixed Flow Ducted Rocket Development program (DRPTV), Contract No. F33615-77-C-2057. The formulation was tested in 7.62 cm (3 inch) and 17.8 cm (7 inch) heavywall hardware and 17.8 cm (7-inch) flightweight hardware in wind tunnel tests at Arnold Engineering Development Center (AEDC). For the hybrid program, the original formulation was subsequently modified by replacing some of the AP with sodium nitrate on an equal molar basis to scavenge the HCl formed in the exhaust products.

Table 3. Gas Generator Hybrid Fuels Evaluated.

No.	Fuel	Oxidizer	Maximum I_{sp} N-S/Kg	Mixture Ratio
1	ARCADENE 399	LOX H_2O_2 95%	3112.1 2947.3	1.5 3.5
2	ARCADENE 399C (w $NaNO_3$) (wo Fe_2O_3)	LOX H_2O_2 95%	3128.7 2945.3	1.5 4.0
3	AGN	LOX H_2O_2 95%	2873.7 2785.5	0.5 1.0
4	ARCADENE 246B	LOX	2084.2	0.5
5	ARCADENE 246*	LOX	3040.5	1.0
6	ARCADENE 246*	LOX	3148.4	1.5
7	ARCADENE 246*	LOX	3040.5	2.0
8	12% HTPB 48% AP 40% Al	LOX H_2O_2 95%	2812.9 2880.6	0.33 0.67

NOTES:

1. ARCADENE 399C: scavenged version of ARCADENE 399.
2. HTPB; hydroxyl terminated polybutadiene.
3. AP: ammonium perchlorate.
4. Al: aluminum.
5. AGN: aminoguanidine nitrate.
6. ARCADENE 246*: scavenged version of ARCADENE 246B with 35% solid oxidizer.

A conventional gas generator propellant (ARCADENE 246B) was also evaluated. The formulation [25.6 percent polybutadiene acrylonitrile (PBAN), 69.5 percent ammonium perchlorate (AP), 4.5 percent curative (DER-331) 0.4 percent iron oxide (Fe_2O_3)] was selected because it: (1) was characterized over a wide range of burning rates; (2) had excellent propellant reproducibility; and (3) had excellent processing and physical property performance. The formulation was used to pressurize the HARDROCK Silo Lid Door Opening Actuator (Contract F04704-A3-C-0048), the UPSTAGE Jet Gas Generator program (Contract F04704-87-C-0054), and the MX Buried Trench Weapon System (Contract F04704-85-C-0039). The original formulation was modified by: (1) replacing some of the

AP with sodium nitrate on an equal molar basis to meet the HCl emissions requirement; and (2) reducing the weight-percent of the solid oxidizer from 69.5 to 35 percent, and subsequently increasing the binder content to make the exhaust products more fuel-rich.

Metallized fuels were also evaluated. The best-performing metallized formulation had 40 percent aluminum and 48 percent AP. The I_{sp} for this formulation was 9.6 percent lower than the scavenged ARCADENE 399, and the system optimized at a lower mixture ratio. One of the design issues which resulted from this evaluation was higher flame temperatures; these higher temperatures for metallized systems were incompatible with many of the advanced material concepts considered for this design.

A limited evaluation of an ARCADENE 399 variant formulation was completed under corporate IR&D. This formulation variant consisted of 25 percent hydroxyl-terminated polybutadiene (HTPB) binder including 3 percent plasticizer, 34 percent polystyrene, 21.5 percent ammonium perchlorate, 15.5 percent sodium nitrate, 2 percent iron oxide, and 2 percent fluorinated graphite (CF_x). Pint mixes were made and cast into cartons. Samples of the fuel were cut from the cartons and tested in a strand burner at six pressures [from 1.38 to 13.8 MPa (200 to 2,000 psi)] and atmospheric pressure. The strands had a burning rate of 0.38 cm/sec (0.15 in/sec) at a chamber pressure of 6.88 MPa (1,000 psi). Further, they exhibited good ejection characteristics and would not burn below 3.44 MPa (500 psi). A limited evaluation of an ARCADENE 246 variant formulation was also completed under corporate IR&D funding. This formulation consisted of 65 percent polybutadiene-acrylic acid-acrylonitrile (PBAN), 20.3 percent ammonium perchlorate, and 14.7 percent sodium nitrate. The strands had a burning rate of less than 0.25 cm/sec (0.1 in/sec) at a chamber pressure of 6.88 MPa (1,000 psi).

The scavenged version of ARCADENE 399 was eventually selected for the point design. The formulation was selected because it exhibited better ejection characteristics than ARCADENE 246, and the theoretical I_{sp} as a function of mixture ratio was flat above 1.5, which provided a wider operability range.

2.2.2 Classical Hybrid Fuels

The fuels utilized in the classical hybrid (Table 4) were selected from our solid fuel ramjet (SFRJ) database. Based on our airbreathing experience, we assumed that the oxidizers would be gasified prior to injection to minimize concerns of flameholding, injection, mixing efficiency, and hypergolic combustion. The H_2O_2 was decomposed using a catalyst bed prior to injection, and the LOX was preburned using propane to obtain a gasified oxidizer (GOX) temperature of 667K (1,200°R).

Table 4. Classical Hybrid Fuels Evaluated.

No.	Fuel	Oxidizer	Maximum I_{sp} N-S/Kg	Mixture Ratio	Propane Kgs
1	HTPB	GOX*667K	3291.6	2.5	5615
		H_2O_2 95%	2996.3	6.5	0
2	75% HTPB	GOX*1000K	3276.8	2.75	8601
	25% PS	GOX*667K	3277.8	2.5	5638
		H_2O_2 95%	2990.4	6.5	0
		H_2O_2 88%	2898.3	7.5	0
3	HC + 10% AP	GOX*667K	3245.5	2.0	5315
		H_2O_2 95%	2988.5	6.0	0
4	HC + 20% AP	GOX*667K	3241.5	2.0	5321
		H_2O_2 95%	2981.6	5.0	0
5	HC + 18% Al	GOX*667K	3271.0	2.0	5273
		H_2O_2 95%	3020.0	5.5	0
6	HC + 18% Mg/Al	H_2O_2 95%	3015.0	5.5	0
7	HC + 18% Al +	GOX*667K	3252.3	1.75	5063
	10% AP	H_2O_2 95%	3018.9	4.75	0
8	50% HTPB				
	50% Mg/Al	H_2O_2 95%	3061.1	2.5	0
9	50% HTPB	GOX*667K	3174.8	1.5	4890
	50% Al	H_2O_2 95%	3074.8	3.0	0

NOTES:

1. HC: hydrocarbon fuel 75% hydroxyl terminated polybutadiene (HTPB); 25% polystyrene (PS)
2. Propane is used to gasify LOX.
3. AP: ammonium perchlorate.
4. Al: aluminum.
5. Mg/Al: magnesium aluminum alloy.

The baseline fuel for the classical hybrid approach is a hydrocarbon (HC) SFRJ fuel which contains no solid oxidizer and is 75 percent HTPB and 25 percent polystyrene (PS). The addition of solid oxidizer and metals to the baseline fuel was evaluated at the reference conditions of 6.88 MPa (1,000 psia) and an expansion ratio of 15. Alternate binders and nonmetallic fillers were also evaluated and found to provide minimal differences in performance and density.

We concluded from the evaluation of fuels and oxidizers that fuel additives provide different results with gasified oxygen (GOX) than with H_2O_2 as the oxidizers. The addition of aluminum to the baseline solid fuel decreases the I_{sp} and lowers the optimum mixture ratio with GOX; using peroxide, only the I_{sp} is reduced. The addition of solid oxidizer decreases I_{sp}, reduces the optimum mixture ratio, and improves the burning rate tailorability. The performance penalty and the shift in optimum mixture ratio associated with increased solid oxidizer levels is shown in Table 4. Further, AP concentrations above 10 percent in the solid fuel will require scavenging to meet the HCl emissions goal of less than 1 percent.

Preliminary analysis of the payload performance of these fuels did not indicate a formulation with a superior capability. The higher theoretical I_{sp} for the classical hybrid was offset by the increase in system weight due to the propane system required to gasify the oxidizer. The increased density of H_2O_2 was offset by the lower I_{sp} and the requirement to carry a catalyst bed.

To summarize the fuel evaluation, ARC selected the scavenged ARCADENE 399 as the fuel of choice for the gas generator point design and the hydrocarbon fuel containing HTPB and PS for the classical hybrid point design. These two fuels were used for all of the engineering trade studies (Section 2.3) and cost parametrics developed and presented in Section 2.6.

2.3 Propulsion Conceptual Studies

Concurrent with the oxidizer and fuel studies, booster system trade studies were initiated. The hybrids were evaluated using LOX and H_2O_2 oxidizers with either a pressure-fed or turbopump delivery system. Eight configurations were evaluated (two hybrid concepts, two oxidizers, two oxidizer feed systems). In order to compare the eight configurations, certain vehicle parameters were held constant.

The overall vehicle diameter was set at 3.7 m (12 feet), close to the shuttle solid rocket booster (SRB) diameter [3.7 m (12.2 feet)]. The thrust profile established for the Advanced Solid Rocket Motor (ASRM) was provided by MSFC for these calculations. The maximum operating pressure occurs about 10 seconds into operation and was calculated to be 7.57 MPa (1,100 psia). The nozzle expansion ratio was set at 15. Using these values, a mixture ratio was selected to produce the highest vacuum I_{sp}, and this ratio was held constant for the entire burn.

Fuel and oxidizer requirements were calculated to meet the vacuum thrust profile specified (ASRM) in the statement of work. These values were considered to be independent of oxidizer feed system; therefore, only four unique sets of values were calculated. With the propellant weights (fuel and oxidizer) identified, conceptual booster designs were laid out for each of the eight options. Packing efficiencies and fuel utilizations were assumed to be 95 and 98 percent for the classical hybrid and gas generator grains, respectively. Grain geometry was not optimized at this time, but consideration was given to avoid high port velocities which could lead to erosive burning phenomena. Structural materials were selected for the major components (motor cases, oxidizer tanks, gas pressurization tanks). Composite materials were used extensively, especially for the pressure-fed designs; results from previous trade studies clearly indicated that based on weight, large pressure-fed boosters with metal tanks could not compete with turbopump booster designs.

Since the major goal of this effort was to evaluate relative merits of the eight configurations, components which were common to all eight were not evaluated in great detail. These items include thrust vector control, electronics, instrumentation, nose cone, and recovery system. Weight allocations for these items were derived from similar systems, notably the shuttle SRB.

A single turbopump derived from the F-1 pump design for the Saturn V booster was used to generate weight breakdown. The gas generator designs carried separate solid gas generators to power the turbines. The classic hybrid designs used propane, burned with some oxidizer, to power the turbines and to gasify the oxidizer.

Pressure-fed design options considered a number of methods to pressurize the oxidizer tank. These options are discussed in detail in Appendix A. For the purpose of the engineering trade studies, the pressure-fed LOX options used Tridyne to pressurize the oxidizer tank. Tridyne was developed by Aerojet and consists of a small fraction of reactive gases (0.06 moles hydrogen and 0.03 moles oxygen) combined with an inert diluent (0.91 moles helium) to produce a nondetonable mixture that can be stored at high pressure. The hot-gas temperature is controlled by varying the mixture concentration. Pressure-fed H_2O_2 options used subcooled helium, which was heated in a heat exchanger by the decomposition of H_2O_2 to pressurize the oxidizer tank. Tank pressurization in the turbopump options was accomplished using helium stored at ambient temperature to provide positive suction head.

Booster layout and component weight breakdown for seven of the eight designs are provided in Figures 1 through 7; the classical hybrid-H_2O_2-turbopump design was never completed because by combining the results from the other design efforts, it was determined that this option would not be cost competitive with LOX (Figure 8).

Table 5 summarizes the results of the study: (1) the classical hybrids were 0.5 to 2.5 percent lighter than equivalent gas generator hybrids; (2) systems using LOX were 7 to 10 percent lighter than systems using H_2O_2, but they were also 5 to 17 percent longer due to the lower density of LOX; and (3) turbopump systems were approximately 2 percent lighter than the pressure-fed options, and 33 to 68 percent lower cost. Additional conclusions drawn from this study were: (1) use of composites in large structural components provides substantial performance improvement; (2) pressure-fed systems benefit the most from the use of composites; and (3) the benefits of using composites for expendable systems warrant continued consideration and development.

Incorporated into each design was a reliability goal of 0.9995 for the booster. This goal was apportioned to each major component using historical data supplied by Boeing Aerospace. For the initial trade studies, reliability was evaluated as a weight impact on the system. The liquid oxidizer system incorporated redundancy (additional turbopump to provide pump out capability) to meet the reliability goals; the remaining systems were designed at a higher margin of safety (1.6). Each design met the MSFC thrust trace.

The life cycle cost (LCC) for each configuration was estimated using a constant flight rate of one flight per month for 10 years. The lowest LCC was provided by Concept 1AT, the pump-fed gas generator hybrid with LOX oxidizer ($11.4 billion), and the highest was provided by Concept 2, the classical

FOLDOUT FRAME

Concept No. 1
ARCADENE-399C/Hydrogen Peroxide
Pressure Fed Version

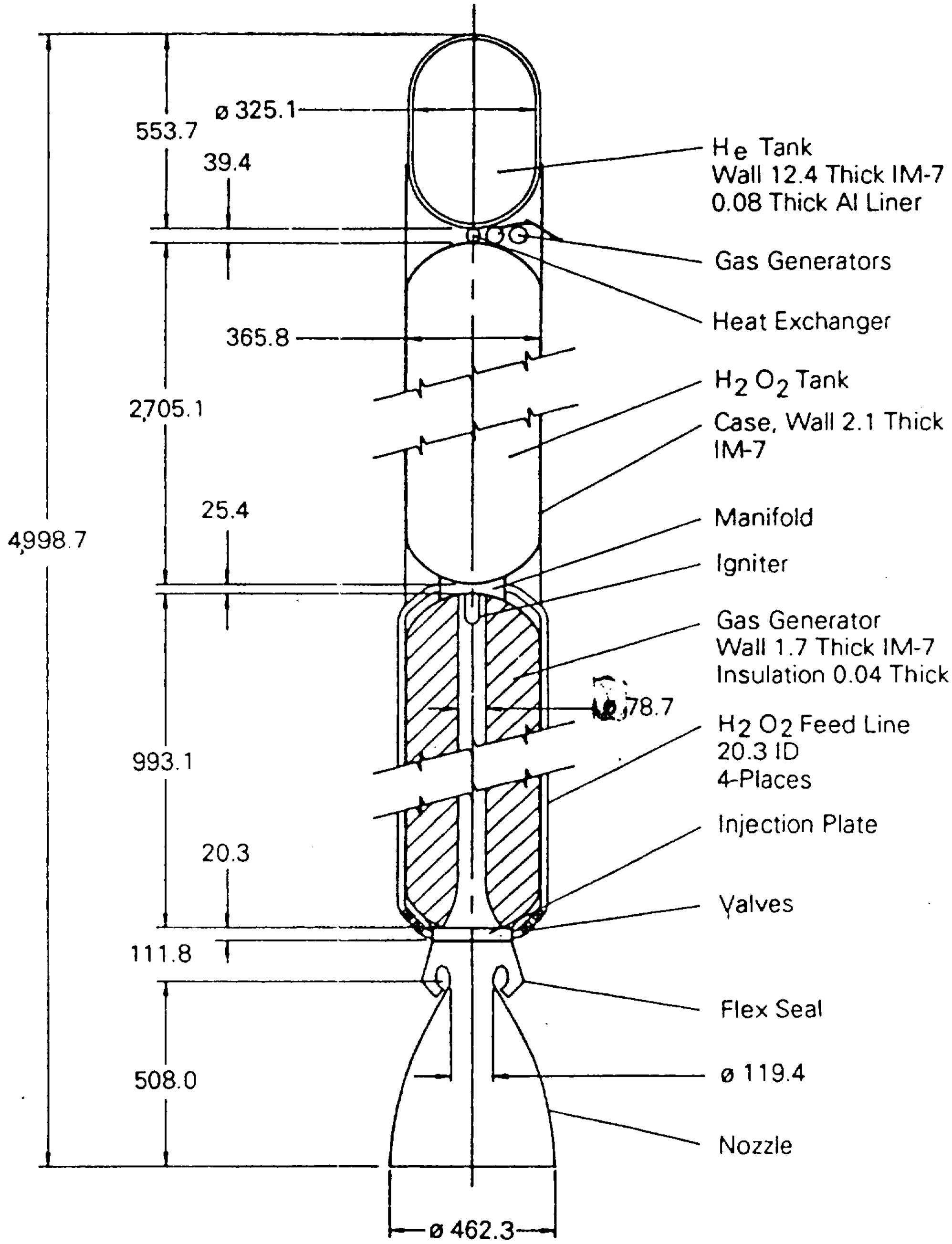
Component Weight Breakdown

Subsystem	Component	Material	Weight
Gas Generator	Propellant	ARCADENE-399	107,9
	Case	IM-7/826	2,7
	Liner/Insul.	Kevlar/EPDM	4
	Igniter		2
Oxidizer Delivery System	Oxidizer	95% H_2O_2	435,9
	Tank	IM-7/826	9,7
	Liner	Aluminum	7
	Piping	S.S.	6
	Manifolds		
	Valves		
Pressurizing System	Gas	Helium	6,7
	Tank	IM-7/826	14,7
	Liner	Aluminum	
	Ext. Insul.	Blown Foam	
	Chiller		
	Feed Lines	S.S.	
	Valving		
	Heat Exchangers		2
	Gas Generator	Catalyst Bed	
Thrust Chamber	Injector	S.S.	3,2
	Case	HP9-4 Steel	6
	Insulation	Silica/EPDM	1,2
	Nozzle	Ablative	10,8
Ancillary Components	Ext. Insulation		2
	Interstage	Al-Li	4
Propellant Weight			543,9
Inert Weight			53,[illegible]
Total Propulsive System			597,0

* For a metal (aluminum-lithium) gas generator case add 12,796 kg

FOLDOUT FRAME 2

(kg)
'54
'80
:77
:17
)75
'59
'30
,83
-
-
50
'58
69
-
-
-
-
'66
38
:21
;35
:25
)86
:22
08
;29
47
)76



Note: All dimensions are in centimeters

Figure 1. *Gas generator hybrid with hydrogen peroxide (pressure fed version).*

0190-HYBRID-RPT

FOLDOUT FRAME 1.

Concept No. 1T
ARCADENE-399C/Hydrogen Peroxide
Turbopump Version

Component Weight Breakdown

Subsystem	Component	Material	Weight
Gas Generator	Propellant	ARCADENE-399	107,9
	Case	IM-7/826	2,7
	Liner/Insul.	Kevlar/EPDM	4
	Igniter		2
Oxidizer Delivery System	Oxidizer	95% H_2O_2	444,2
	Tank	Al-Li	5,1
	Liner	Teflon	2
	Piping	Aluminum	
	Manifolds		
	Valves		
Pressurizing System	Gas	Helium	4
	Tank	IM-7/826	6
	Liner	Aluminum	
Turbopump System	Hardware		1,5
	Catalyst Bed	Silver/Nickel	
Thrust Chamber	Injector	S.S.	3,2
	Case	HP9-4 Steel	6
	Insulation	Silica/EPDM	1,2
	Nozzle	Ablative	10,8
Ancillary Components	Ext. Insulation		2
	Interstage	Al-Li	3
Propellant Weight			552,1
Inert Weight			28,1
Total Propulsive System			580,2

* For a metal (aluminum-lithium) gas generator case add 12,796 kg

FOLDOUT FRAME 2

(kg)

54
80
77
17

34
70
03
-
-
-

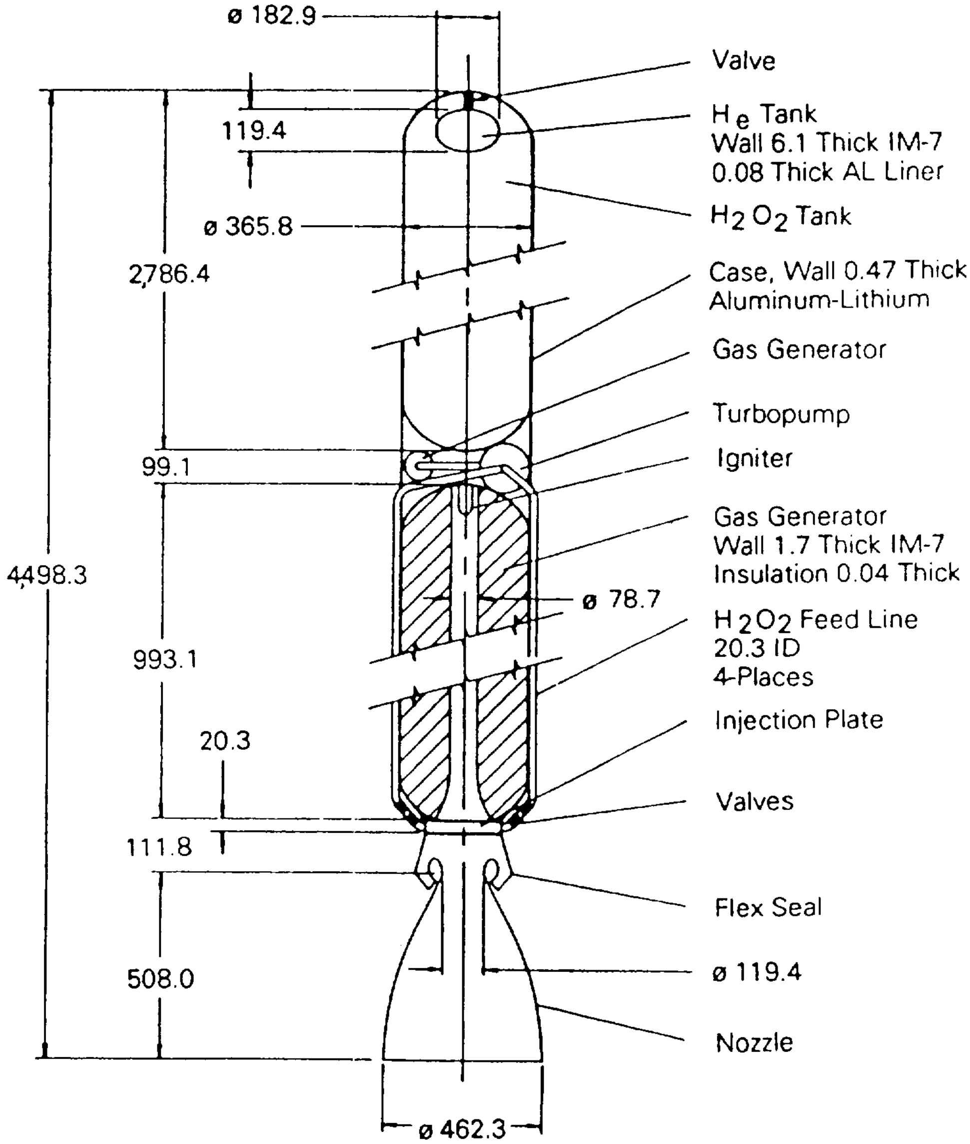
25
26
19

88
79

21
35
25
86

22
40

88
11
99



Note: All dimensions are in centimeters

Figure 2. Gas generator hybrid with hydrogen peroxide (turbopump version).

0190-HYBRID-RPT

FOLDOUT FRAME

Concept No. 1A
ARCADENE-399C/LOX
Pressure Fed Version

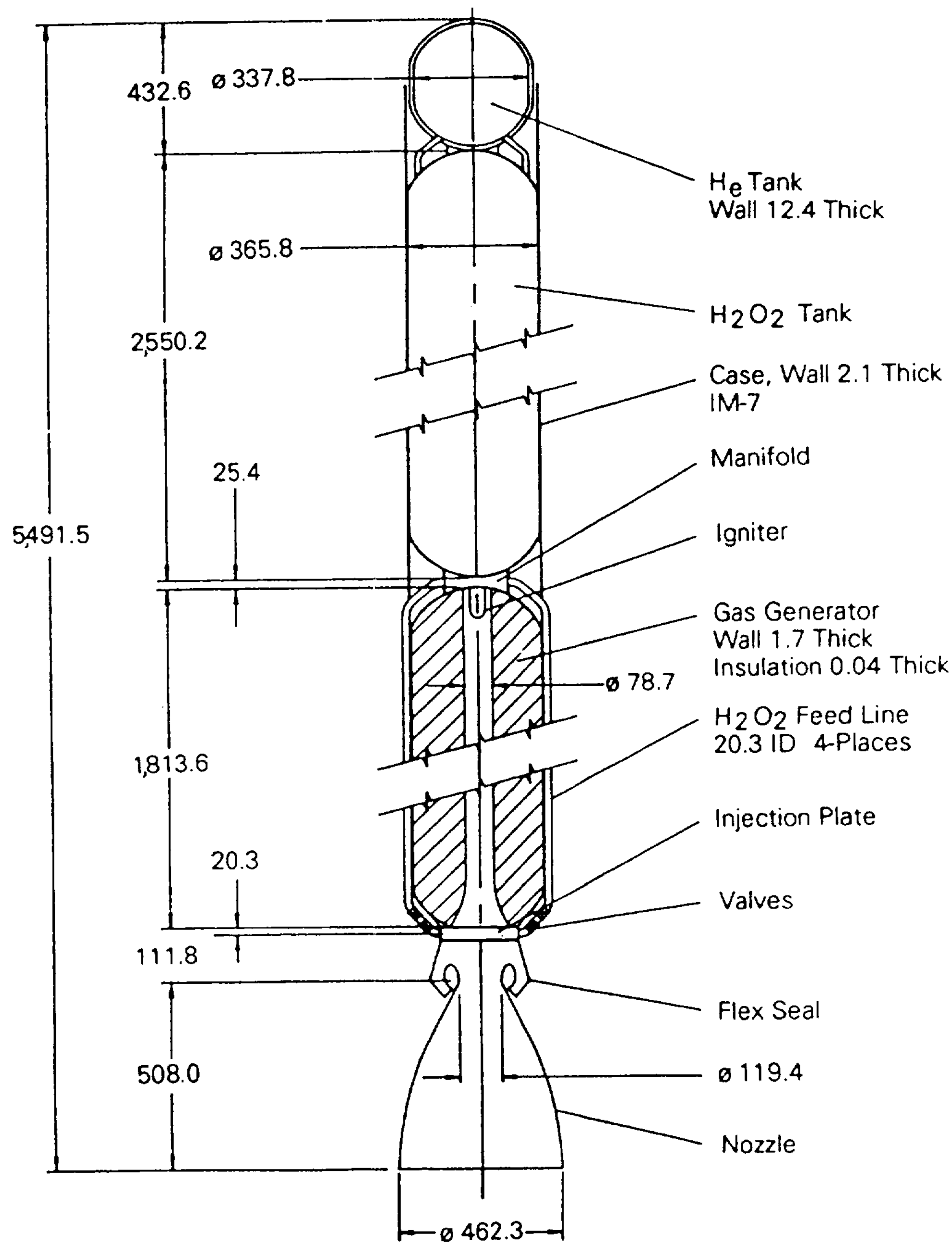
Component Weight Breakdown

Subsystem	Component	Material	Weight
Gas Generator	Propellant	ARCADENE-399	203,25
	Case	IM-7/826	4,9
	Liner/Insul.	Kevlar/EPDM	8
	Igniter		2
Oxidizer Delivery System	Oxidizer	LOX	304,8
	Tank	IM-7/826	7,9
	Liner	Aluminum	6
	Piping	S.S.	3
	Manifolds	S.S.	
	Valves	S.S.	
	Insulation	Blown Foam	
Pressurizing System	Gas	He/H2/O2	3,9
	Tank	IM-7/826	9,2
	Liner	Aluminum	
	Ext. Insul.	Blown Foam	
	Catalyst Bed		
	Feed Lines	S.S.	
	Valving		
Thrust Chamber	Injector	S.S.	3,2
	Case	HP9-4 Steel	6
	Insulation	Silica/EPDM	1,2
	Nozzle	Ablative	10,8
Ancillary Components	Ext. Insulation		2
	Interstage	Al-Li	4
Propellant Weight			508,
Inert Weight			45,
Total Propulsive System			553,

* For a metal aluminum-lithium gas generator case add 12,796 kg

FOLDOUT FRAME 2

(kg)
51
17
12
7
76
17
06
50
-
-
0
16
70
78
1
-
-
-
21
35
25
36
22
08
27
09
36



Note: All dimensions are in centimeters

Figure 3. *Gas generator hybrid with LOX (pressure fed version).*

0190-HYBRID-RPT

FOLDOUT FRAME /

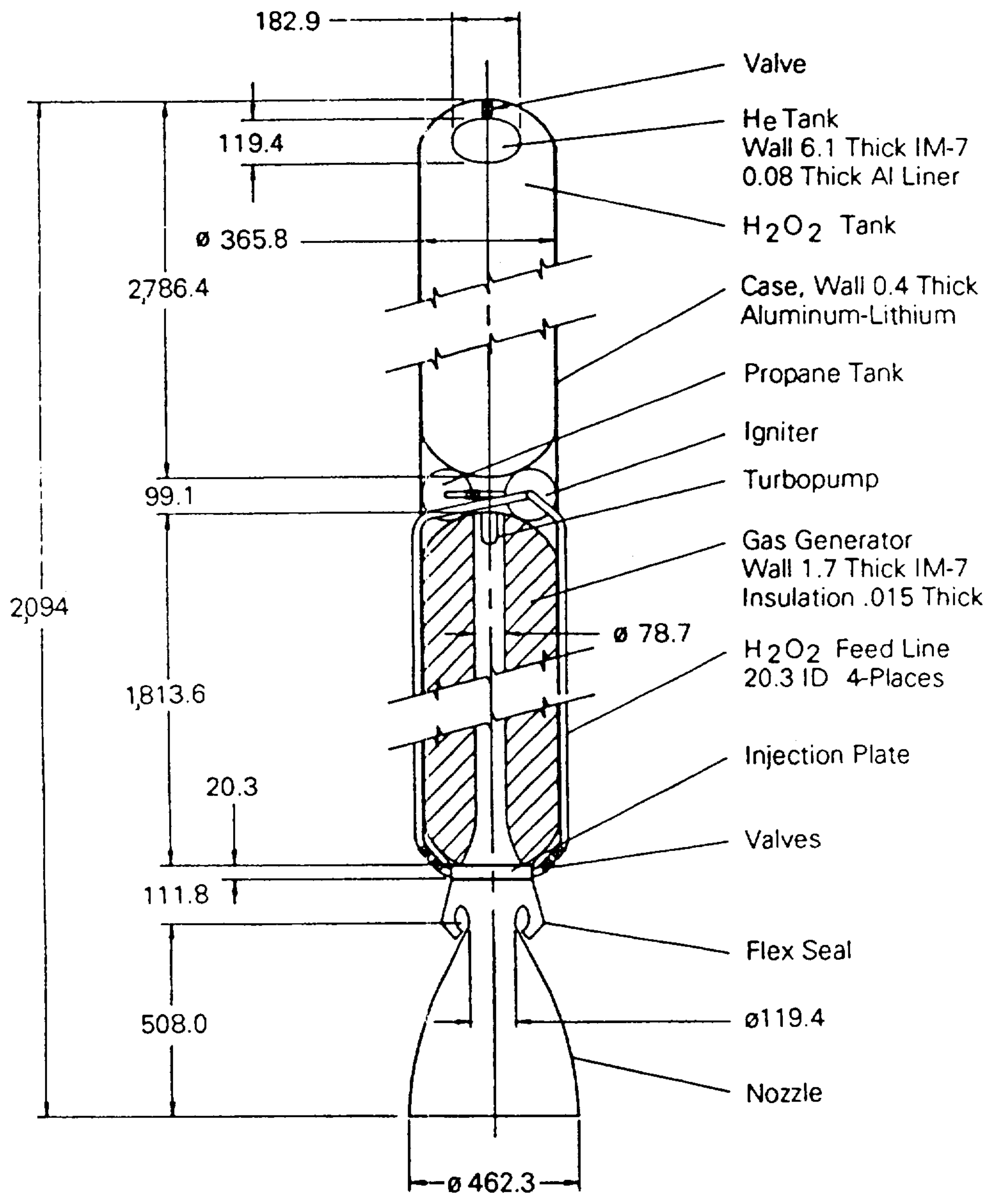
Concept No. 1AT
ARCADENE-399C/LOX
Turbopump Version

Component Weight Breakdown

Subsystem	Component	Material	Weight (kg)
Gas Generator	Propellant	ARCADENE-399	203,251
	Case	IM-7/826	4,918
	Liner/Insul.	Kevlar/EPDM	842
	Igniter		217
Oxidizer Delivery System	Oxidizer	LOX	310,166
	Tank	Al-Li	3,701
	Piping	Aluminum	350
	Manifolds	S.S.	-
	Valves	S.S.	-
	Insulation		
Pressurizing System	Gas	Helium	361
	Tank	IM-7/826	556
	Liner	Aluminum	17
	Valving	S.S.	-
	Piping	S.S.	-
Turbopump System	Hardware		1,463
	Propane		174
	Tank		-
	Propane Delivery System		-
Thrust Chamber	Injector	S.S.	3,221
	Case	HP9-4 Steel	635
	Insulation	Silica/EPDM	1,225
	Nozzle	Ablative	10,886
Ancillary Components	Ext. Insulation		222
	Interstage		340
Propellant Weight			513,416
Inert Weight			30,156
Total Propulsive System			543,568

• For a metal (aluminum-lithium) gas generator case add 12,796 kg

FOLDOUT FRAME 2



Note: All dimensions are in centimeters

Figure 4. *Gas generator hybrid with LOX (turbopump version).*

0190-HYBRID-RPT

FOLDOUT FRAME

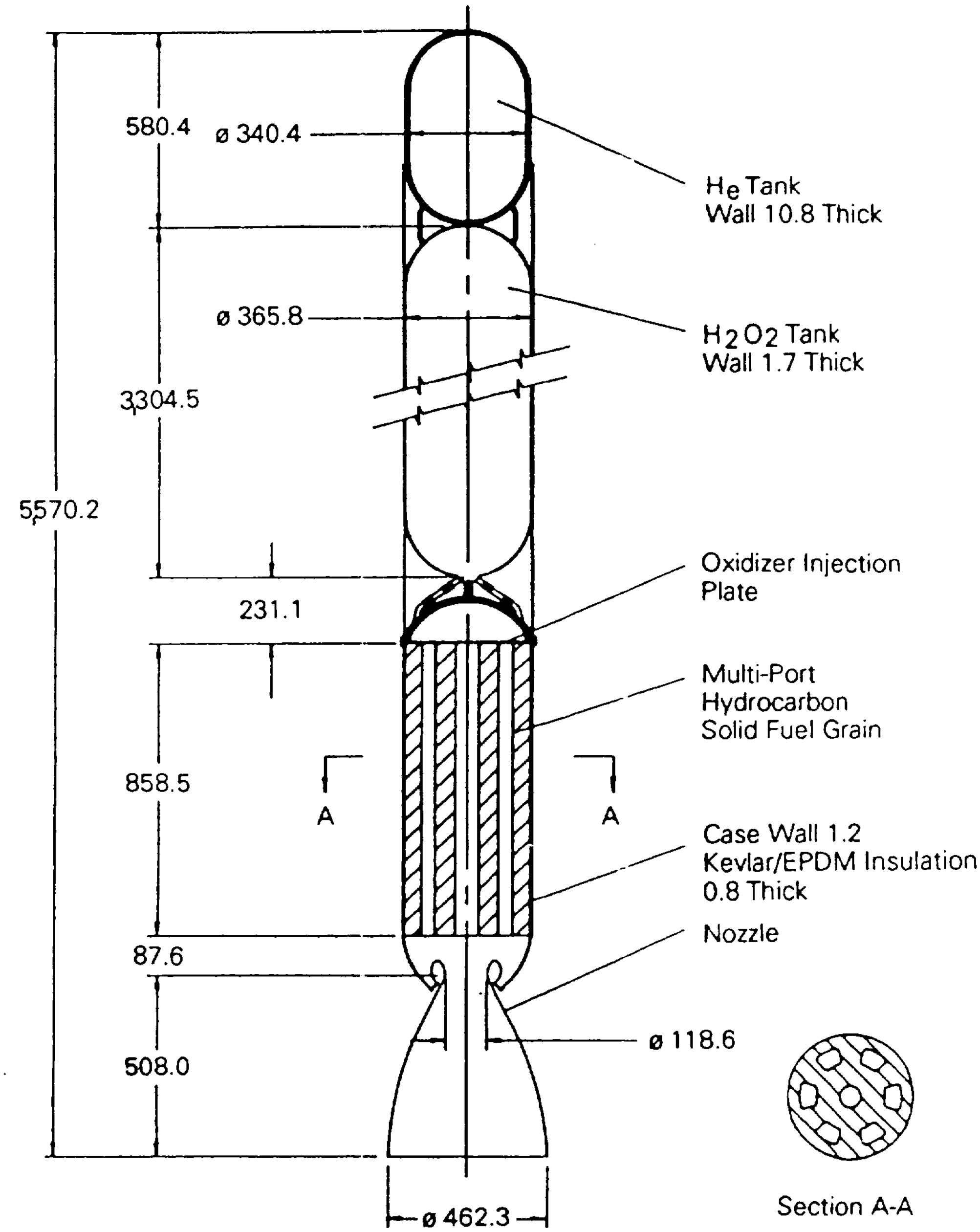
Concept No. 2
Hydrocarbon/H_2O_2
Pressure Fed Version

Component Weight Breakdown

Subsystem	Component	Material	Weight (k
Solid Motor	Fuel	Hydrocarbon	68,497
	Case	IM-7/826	2,474
	Insulation	Kevlar/EPDM	1,964
	Cat. Bed	Ag Plated Ni	4,540
	Injector Plate	Carbon-Carbon	798
Oxidizer Tank	Oxidizer	H_2O_2	473,187
	Tank	IM-7/826	10,351
	Liner	Aluminum	792
	Piping		
	Valves		
	Manifold		
Pressurizing System	Gas	Helium	7,414
	Tank	IM-7/826	12,629
	Liner	Aluminum	258
	Ht. Exchanger	Steel	310
	Cat. Bed	Ag Plated Ni	50
	Plumbing		
Nozzle	-	Ablative	10,886
Ancillary Components	Ext. Insulation	-	222
	Interstage	Al-Li	408
Propellant Weight			541,684
Inert Weight			52,084
Total Propulsive System			593,768

- For a metal (aluminum-lithium) gas generator case add 12,796 kg

FOLDOUT FRAME 2



Note: All dimensions are in centimeters

Figure 5. *Classical hybrid hydrocarbon/hydrogen peroxide (pressure fed version).*

0190-HYBRID-RPT

FOLDOUT FRAME

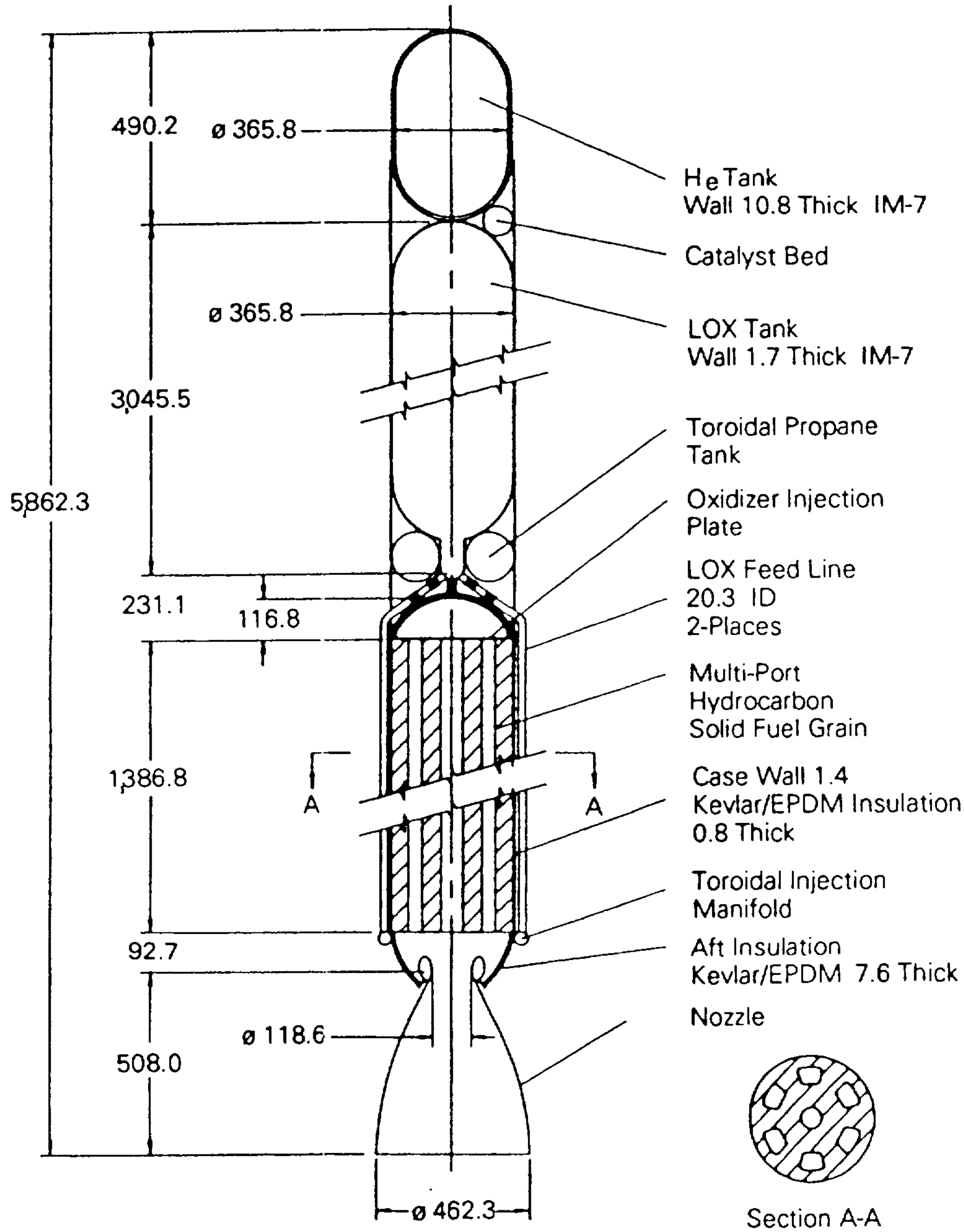
Concept No. 2A
Hydrocarbon/LOX
Pressure Fed Version

Component Weight Breakdown

Subsystem	Component	Material	Weight (kg)
Motor	Fuel	HTPB/Poly	126,371
	Case	IM-7/826	3,839
	Liner/Insul.	Kevlar/EPDM	2,580
	Injector Plate	Carbon-Carbon	-
Oxidizer Delivery System	Oxidizer	LOX	362,167
	Tank	IM-7/826	8,679
	Liner	Aluminum	722
	Piping	S.S.	-
	Manifolds		-
	Valves		-
LOX Pre-heater	Fuel	Propane	5,994
	Tank	IM-7/826	389
	Precombustor		-
	Piping		-
	Valves		-
	Manifolds		-
	Press. Gas	Helium	238
	Press. Tank	IM-7/826	582
Pressurizing System	Gas	$He/H_2/O_2$	4,219
	Tank	IM-7/826	11,399
	Liner	Aluminum	142
	Feed Lines	S.S.	-
	Valving		-
Thrust Chamber	Nozzle	Ablative	10,886
Ancillary Components	Ext. Insulation		222
	Interstate	Al-Li	408
Propellant Weight			494,532
Inert Weight			45,100
Total Propulsive System			539,632

* For a metal aluminum-lithium gas generator case add 12,796 kg

FOLDOUT FRAME 2



Note: All dimensions are in centimeters

Figure 6. Classical hybrid hydrocarbon/LOX (pressure fed version).

0190-HYBRID-RPT

FOLDOUT FRAME /.

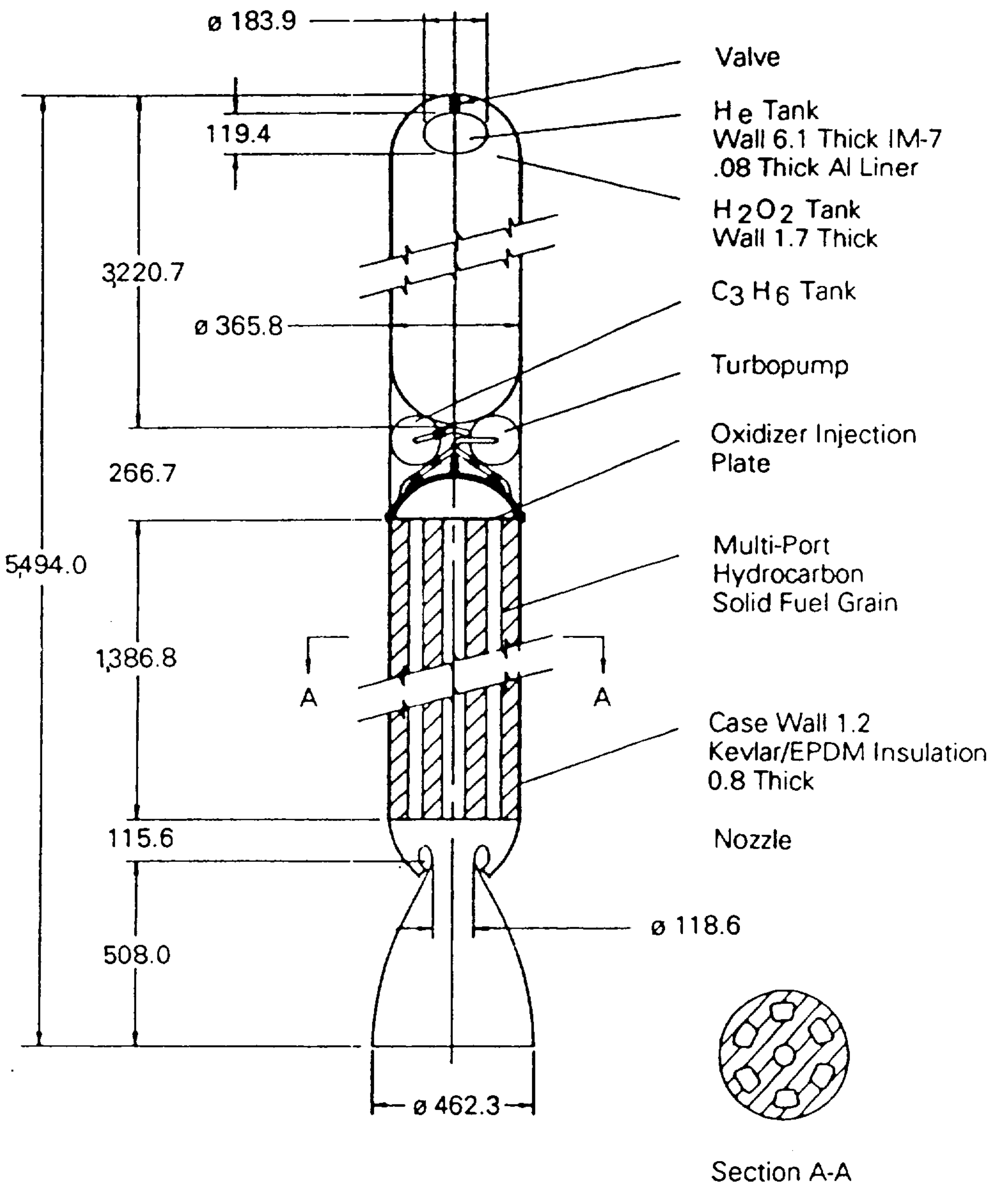
Concept No. 2AT
Hydrocarbon/LOX
Turbopump Version

Component Weight Breakdown

Subsystem	Component *	Material	Weight
Motor	Fuel	HTPB/Poly	126,3
	Case	IM-7/826	3,8
	Liner/Insul.	Kevlar/EPDM	2,5
	Injector Plate	Carbon-Carbon	
Oxidizer Delivery System	Oxidizer	LOX	370,9
	Tank	IM-7/826	3,8
	Piping	S.S.	5
	Manifolds		
	Valves		
Propane System	Propane		6,2
	Tank	IM-7/826	4
	Liner	Aluminum	
	Piping		
	Valves		
	Manifolds		
Pressurizing System	Gas	Helium	8
	Tank	IM-7/826	9
	Liner	Aluminum	
	Feed Lines	S.S.	
	Valving		
Thrust Chamber	Nozzle	Ablative	10,8
Turbopumps	F-1 Combustor		1,4
Ancillary Components	Ext. Insulation		2
	Interstate	Al-Li	3
Propellant Weight			503,5
Inert Weight			26,8
Total Propulsive System			530,3

* For a metal aluminum-lithium gas generator case add 12,796 kg

(kg)
71
39
80
-
70
61
04
-
-
45
05
34
-
-
-
52
91
31
-
-
86
3
2
40
86
05
91



Note: All measurements are in centimeters

Figure 7. *Classical hybrid hydrocarbon/ LOX (turbopump fed).*

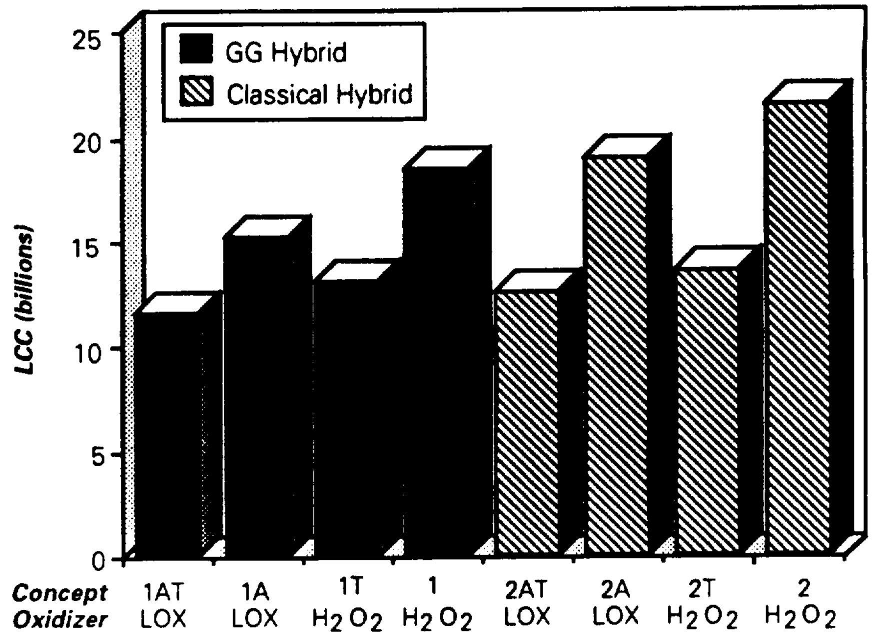


Figure 8. Hybrid configurations life cycle costs.

0190-HYBRID-RPT

Table 5. Concept Summary.

ID No.	Hybrid	Oxidizer	Feed System	Weight*	Length+	LCC (%)***
1	GG**	H_2O_2	Pressure	597	4978	166.5
1T	GG	H_2O_2	Turbopump	580	4539	117.3
1A	GG	LOX	Pressure	553	5532	133.6
1AT	GG	LOX	Turbopump	543	5819	100.0
2	Classical++	H_2O_2	Pressure	594	5560	189.0
2T	Classical	H_2O_2	Turbopump	---	---	120.9
2A	Classical	LOX	Pressure	540	5837	168.0
2AT	Classical	LOX	Turbopump	530	5494	111.4

+ In centimeters.

* In thousands of kilograms.

** Gas generator fuel was ARCADENE 399C.

++ Classical hybrid fuel was 75 percent HTPB, 25 percent PS.

*** Compared to the gas generator hybrid with pump-fed LOX.

hybrid with pressure-fed H_2O_2 ($22 billion). A summary of the results is shown in Table 5 and Figure 8. To calculate the costs of the eight conceptual designs, certain assumptions had to be made. These assumptions are as follows:

- Classical hybrid utilized gaseous oxidizer injection.
- H_2O_2 was decomposed by a catalyst bed.
- Fuel utilization for the gas generator was 98 percent, and the classical hybrid was 95 percent.
- Turbopump system had pump-out capability to meet the mission.

As a result of our initial trade studies, ARC selected the gas generator hybrid to develop a more-detailed point design and dropped all consideration of the classical hybrid. The gas generator hybrid had lower calculated life cycle cost, and the classical hybrid presented higher development risk due to the scaling uncertainties associated with the complex interactions between the oxidizer and the solid fuel grain.

2.4 Point Design

To encompass a range of possible vehicle system requirements, MSFC requested designs for two hybrid rocket motors: a large (full-size) motor, two of which in combination meet the specified ASRM thrust profile; and a small (quarter-size) motor, eight of which in combination meet the same thrust profile. The full-size motor point design will be described first, followed by the quarter-size motor design. The full-size design features a fuel-rich gas generator which contains sufficient solid oxidizer to be self-sustaining above a predetermined operating pressure (2.06 MPa, 300 psia), yet completely extinguishes at pressures below 2.06 MPa (300 psia) without a liquid or gaseous oxidizer. The fuel-rich products from the gas generator are injected into a separate thrust chamber, mixed with an oxidizer, and burned to completion. This approach eliminates many of the complex processes involved in classical hybrid rocket motor design. Flow between the gas generator and the thrust chamber is subsonic; thus, changes in chamber pressure are communicated to the gas generator. By this means, the fuel burning rate in the gas generator can be modulated by changing the oxidizer flow rate into the thrust chamber which affects chamber pressure. Thrust can be terminated by shutting off oxidizer flow which causes the gas generator pressure to fall below the combustion limit.

Design of the oxidizer delivery system considered both turbopump and pressure-fed options, and point designs were generated for both. The turbopump design features four oversized pumps, capable of supplying 100 percent of the required oxidizer flow, even with one pump out of operation. The pressure-fed design features a Tridyne system (helium, hydrogen, oxygen) at a pressure of 68.9 MPa (10,000 psia). Both designs utilize LOX as the oxidizer. The exhaust emissions have less than 1 percent HCl by weight.

The design effort focused upon maximizing the safety and reliability characteristics of the vehicle. A structural safety factor of 1.6 was chosen to provide a conservative margin. Design simplicity was emphasized where possible to improve safety, reliability, and cost. Although safety, reliability, and cost factors were given priority over performance, the resulting design provides performance gains over the current shuttle SRB or other advanced booster designs. Layout drawings for both turbopump and pressure-fed, full-size booster designs are given in Figure 9.

Point designs for both pressure-fed and turbopump options were generated assuming a peak chamber pressure of 8.62 MPa (1253 psia) and a nozzle expansion ratio of 15. It was recognized that these conditions might not be optimal for either of the systems, but this assumption permitted commonality in the subsequent design effort, as well as a straight-forward basis for comparing the two system designs. Weight breakdowns for the pressure-fed and turbopump options are presented in Tables 6 and 7. The turbopump version is lighter than the pressure-fed version by 3.4 percent. Both designs incorporate liquid injection thrust vector control designed for 3 to 5° of thrust deflection.

2.4.1 Gas Generator

The fuel-rich gas generator propellant was derived from a well-characterized formulation previously developed by ARC. The formulation is given in Table 8 and is identified as ARCADENE 399C. The original ARCADENE 399 formulation was modified by removing a portion of the AP and replacing it with sodium nitrate on an equal-molar basis. The sodium acts as a scavenger of the chlorine molecule, thereby preventing it from combining with hydrogen to form HCl. ARC has successfully demonstrated a different sodium-nitrate-scavenged propellant in 907 kg (2,000 lb) heavywall hardware (726 kg, 1,600 lbs of propellant); 363 kg (800 lb) Super BATES; and 32 kg (70 lb) BATES motors under

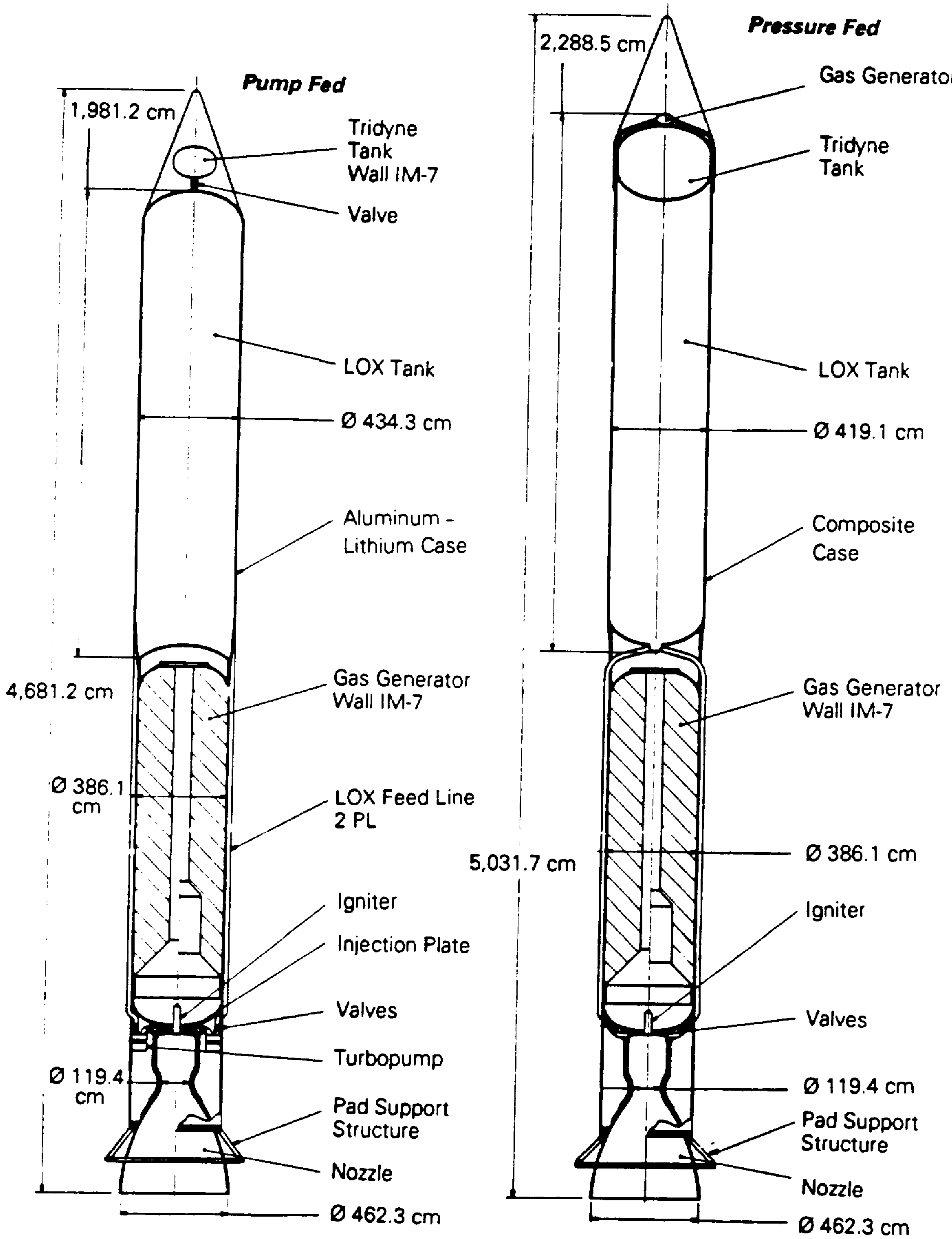


Figure 9. Full size booster designs.

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Table 6. Full-Size Vehicle Weight Breakdown (Pressure Fed).

Subsystem	Element	Weight (kg)
Gas Generator	Fuel	209,911
	Case	7,418
	Liner/Insulation	635
	Igniter	45
Oxidizer Delivery System	LOX	299,700
	Tank (Composite)	13,164
	Feed Lines	522
Pressurizing System	Tridyne	3,601
	Tank	9,740
	Liner	103
	Catalyst Bed	298
	Plumbing and Valving	105
Thrust Chamber	Injector Manifold	1,134
	Chamber/Nozzle	8,174
Ancillary Components	TVC	892
	External Insulation	2,428
	Interstage	1,592
	Nose Cone	497
	Skirt	2,631
	Thrust Transfer Ring	680
Total Weight		563,269 (1,241,796 lbm)

Table 7. Full-Size Vehicle Weight Breakdown (Turbopump).

Subsystem	Element	Weight (kg)
Gas Generator	Fuel	214,900
	Case	7,541
	Liner/Insulation	644
	Igniter	45
Oxidizer Delivery System	LOX	299,700
	Tank (Al-Li)	4,213
	Feed Lines	170
Pressurizing System	Tridyne/Inert	1,124
Turbopumps		816
Thrust Chamber	Injector Manifold	1,134
	Chamber	6,350
Ancillary Components	TVC	892
	External Insulation	2,428
	Interstage	594
	Nose Cone	497
	Skirt	2,631
	Thrust Transfer Ring	680
Total Weight		544,360 (1,200,109 lbm)

Table 8. Gas Generator Fuel.

ARCADENE 399C Formulation

Polystyrene	34.0%
HTPB	29.0%
Ammonium Perchlorate	21.5%
Sodium Nitrate	15.5%
Total	100.0%

Oxidizer

Liquid Oxygen (LOX) at 77.6K

Combustion Properties

Flame Temperature Without LOX (K)	392
Flame Temperature With LOX (K)	1,134
Density of Gas Generator Fuel (g/cm^3)	1.2
C* of Gas Generator Fuel (m/sec)	982
C* of Gas Generator Fuel and LOX (m/sec)	1,686

Major Exhaust Products from Gas Generator Fuel: (moles/100 grams)

H_2O	0.376
CO	0.718
CH_4	0.600
C (Solid)	3.262
NaCl (Liquid)	0.182

Major Exhaust Products from Gas Generator Fuel and LOX: (moles/100 grams)

H_2O	11.372
CO	0.691
N_2	0.076
CO_2	1.185
NaCl	0.044

Vacuum Specific Impulse Gas Generator Fuel (N-sec/kg)	1,208
Vacuum Specific Impulse Gas Generator Fuel and LOX (N-s/kg)	3,128

contract to the Astronautics Laboratory (F04611-89-C-0028). The formulation used in the point design will be demonstrated using subscale motor hardware in Phase 2.

It is the nature of fuel-rich propellants of this type to have extinguishment limits. ARC's point design takes advantage of this characteristic to provide thrust-termination capabilities for the booster.

The design of the gas generator grain was driven by fuel flow rate and total fuel requirements for the specified booster duty cycle. A desired mixture ratio (MR) of 1.4 was selected for optimum performance; this is demonstrated by the plot of vacuum I_{sp} (theoretical)-versus-mixture ratio in Figure 10. This curve shows that I_{sp} as a function of mixture ratio is fairly flat between mixture ratios of 1.25 and 1.5. To determine grain geometry and total propellant requirements, we assumed an impulse efficiency of 92.5 percent and fuel sliver (excess propellant left at burnout) of 2 percent, based on our airbreathing database.

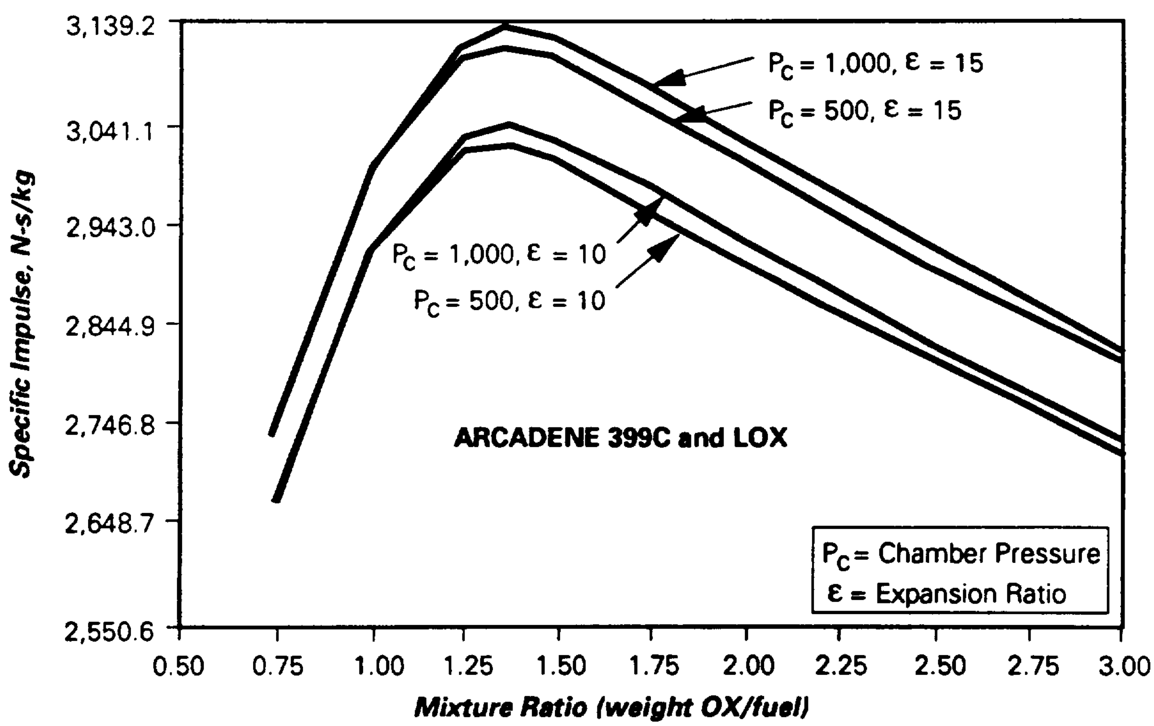

Figure 10. Vacuum I_{sp} versus mixture ratio, expansion ratio, and chamber pressure.

0190-HYBRID-RPT

An outer diameter (OD) of 386 centimeters (152 inches) was selected for the fuel grain; this is considered to be within the current industry manufacturing and transportation experience base. The gas generator grain design, resulting from the ballistic analysis, requires 209,911 kilograms (462,774 pounds) of propellant. An additional 3,402 kilograms (7,500 pounds) is required to drive the turbopumps. The grain design, shown in Figure 11a, is a center-perforated configuration with eight aft slots. The length of the grain for the pressure-fed option is 1,600 centimeters (630 inches), with a port diameter of 79 centimeters (31 inches); the grain length for the turbopump option is 1,625 centimeters (640 inches) to provide the additional fuel for the turbopumps. The slot design for both options is the same. Four of the eight slots extend 343 centimeters (135 inches) axially into the grain, while the remaining four extend only 292 centimeters (115 inches). The slots are 10 centimeters (4 inches) wide and equally spaced.

A structural analysis of the gas generator grain was completed using the Texas Grain Analysis Computer (TEXGAP) program.[2] This three-dimensional, finite-element analysis assumed the grain was cured at 328K (590°R) and then cooled to a bulk temperature of 278K (500°R) (worst case). The results are given in Figure 11b. The maximum strain of 18.2 percent occurs in the bore at the aft end of the grain. This value is within the maximum allowable for propellants when factors due to grain aging are considered. Design changes to provide stress relief would be required if lower bulk grain operating temperatures are specified.

To aid in ignition, the long slots of the fuel grain are overcast for a length of 368 centimeters (145 inches) with a 2.54 centimeter (1 inch) thick web of HTPB-based igniter propellant with a burning rate of 2.54 centimeters/second (1 inch/second). This overcast propellant provides the initial gas generator pressurization. Its burn time is sufficient to allow the LOX flow rates to reach the required levels for either the pressure-fed or turbopump systems. The gas generator pressure will be above the extinguishment limits of the propellant when the starter grain is exhausted due to the secondary

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2. TEXGAP 84, Anatech International Corporation, Report No. ANA-85-0029, Air Force Contract No. F04611-84-C-0017.

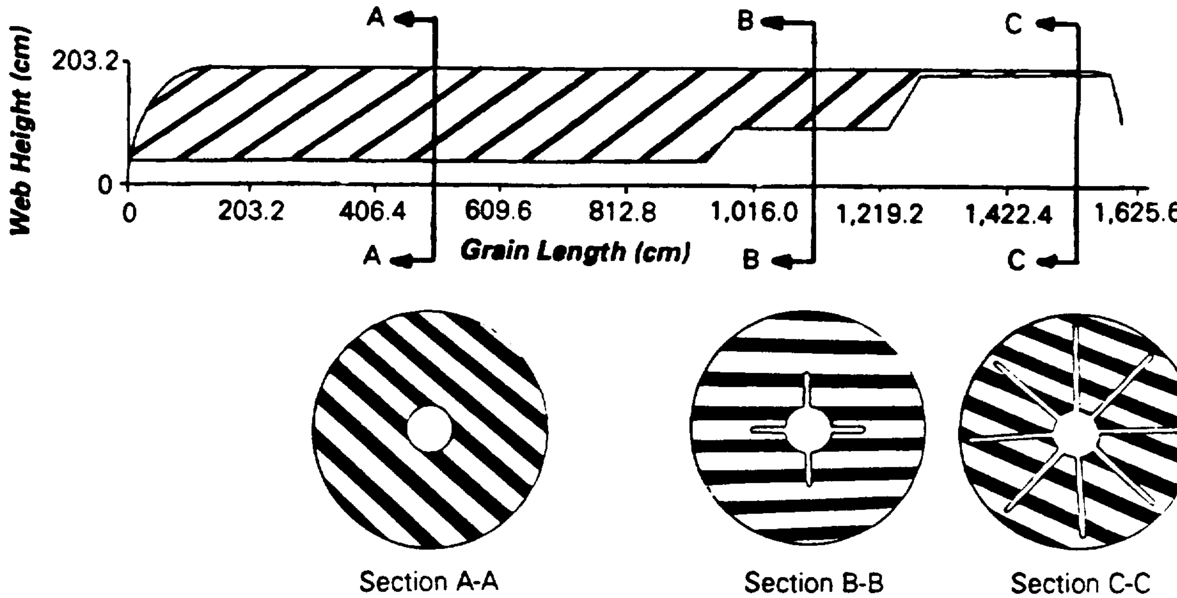


Figure 11a. Full size grain design.

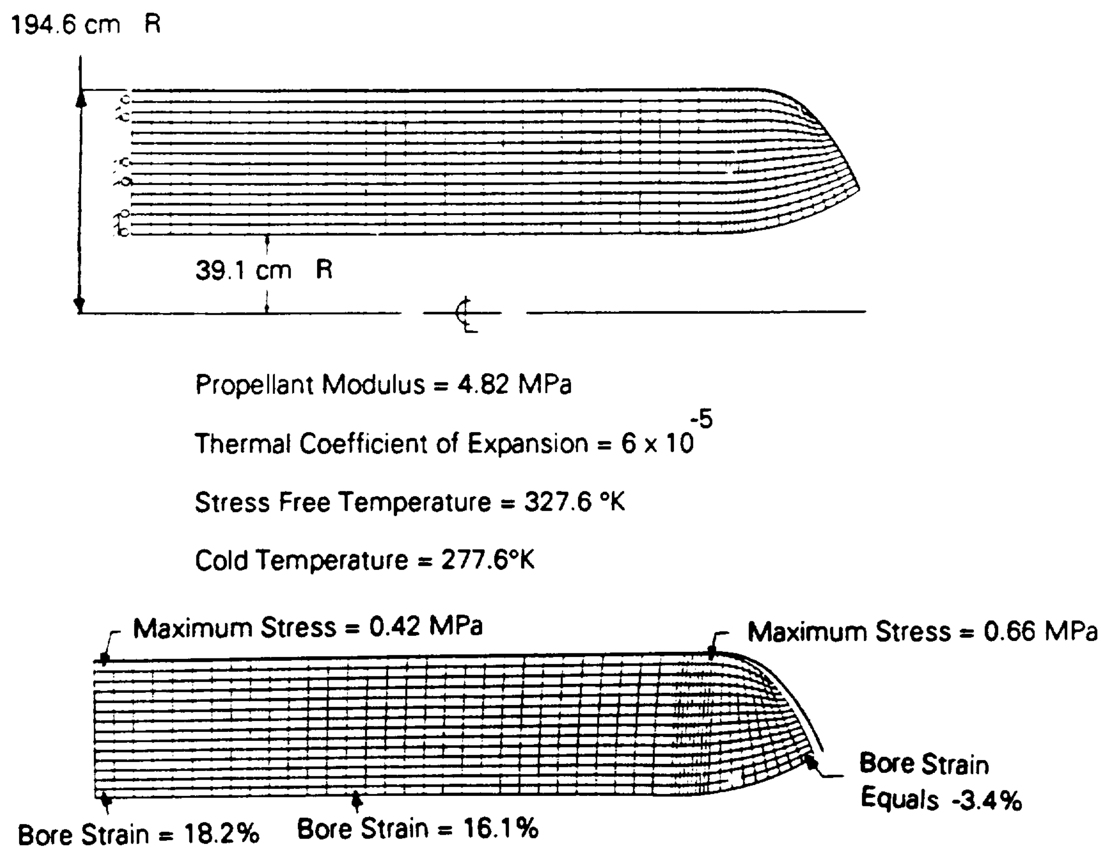


Figure 11b. Results of grain structural analysis.

combustion occurring in the thrust chamber between the fuel rich exhaust products and LOX. Stable gas generator propellant combustion will be maintained until the grain is exhausted or until LOX flow is terminated. The predicted chamber pressure trace due to the start-up propellant is given in Figure 12.

The use of an overcast grain is one of several possible schemes for spooling up the turbopumps and establishing required pressures and subsequent propellant combustion in the gas generator. This approach allows the use of a small aft-mounted igniter which can be easily installed and activated on the pad. An alternative design would be to use a cartridge-type, grain-mounted igniter located in the head end of the gas generator. Further design and trade studies should be performed before the final approach can be selected.

The baseline aft-mounted igniter, which is bolted to the fuel injector manifold, is shown in Figure 13. The igniter provides only limited pressurization of the gas generator, Figure 14, relying on the start-up propellant to build pressure and ignite the balance of the fuel. This minimizes the thrust loads that must be reacted through the injector plate to which the igniter is mounted.

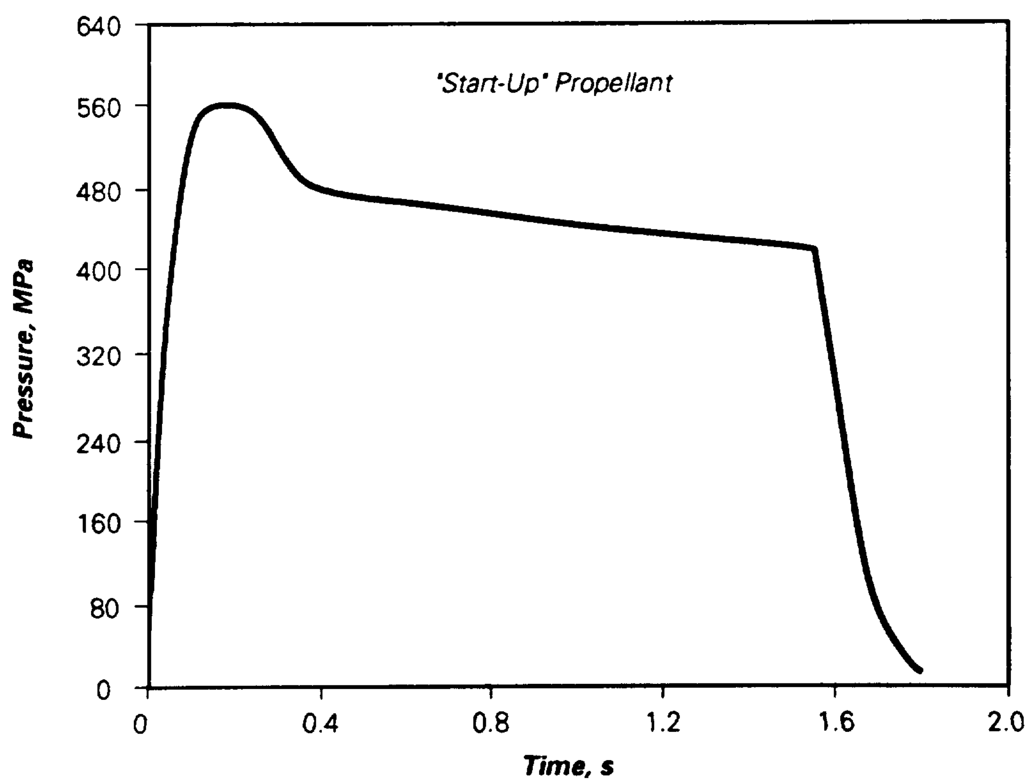
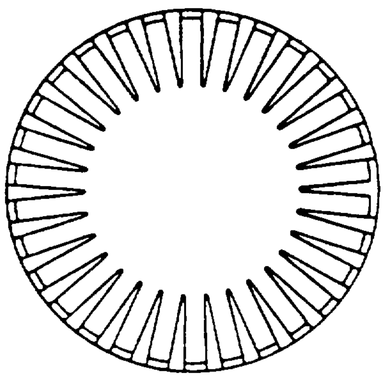


Figure 12. Igniter Pressure traces for full-size booster.

Cross-section	Specifications



Diameter	29.1 cm
Length	83.7 cm
No. of Spokes	30
Web	14.0 mm
Spoke Length	5.64 cm
Propellant Weight	27.2 kg

Figure 13. Igniter grain design (full-size booster).

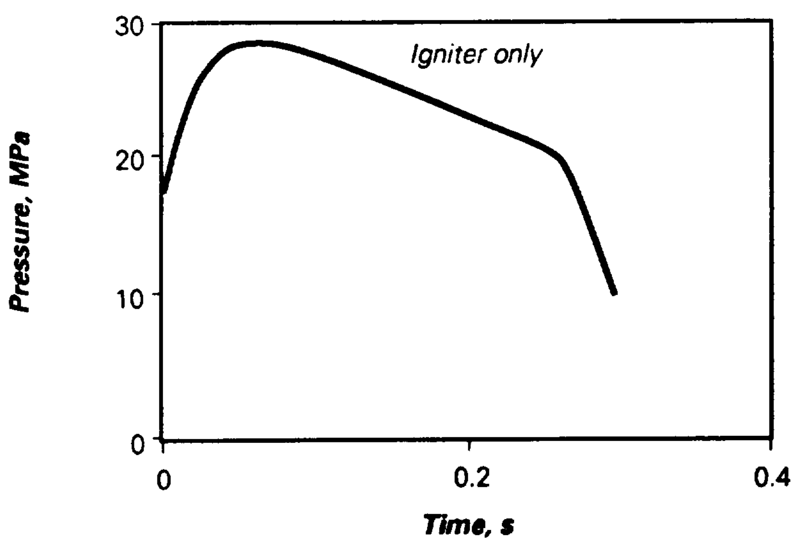


Figure 14. Igniter pressure traces (full-size booster).

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The baseline gas generator case design incorporates carbon/epoxy composite materials to provide weight savings over steel case construction. The gas generator case is monolithic, with steel polar bosses at both the forward and aft ends. The case thickness [2.3 centimeters (0.90 inches)], was calculated for a maximum expected operating pressure (MEOP) of 8.62 MPa (1,253 psia) and to meet bending stiffness requirements commensurate with those for the ASRM. ASRM axial stiffness requirements were not addressed because our booster design transmits loads to the core vehicle at the aft end rather than the forward end, as is the case for the shuttle. The fuel injector manifold interfaces with the aft polar boss and is discussed in the injector design section of this report. The case structural weight was calculated to be 7,416 kilograms (16,350 pounds) for the pressure-fed option, and 7,530 kilograms (16,600 pounds) for the turbopump option.

A steel gas generator case was also sized for comparison. The calculation assumed a tensile strength of 1,514 MPa (220,000 psi) and a 7 percent biaxial stress improvement factor. The resulting case thickness for the same loads and safety factors is 1.7 centimeters (0.66 inches). This results in a case weight of 30,617 kilograms (67,500 pounds) for the pressure-fed option, or 23,133 kilograms (51,000 pounds) heavier than the composite case design.

The baseline gas generator insulation is an ablative material made of HTPB with glass microballoons and has a density of 1.05 gm/cm^3 (0.038 lbs/in^3); it is designated the "ARC thioxotropic insulation process" (ARCTIP). The required insulation thickness is 1.3 centimeters (0.5 inches) in exposed regions such as the forward and aft domes and the tip regions of the long fins, and 0.13 centimeters (0.05 inches) in the areas which will have minimal flame exposure. These regions include wall areas covered by the maximum propellant web. Insulation thicknesses are minimal due to the low flame temperature of the gas generator propellant [1,278K (2,300°R)].

A high thermal margin of safety was imposed on the gas generator and components in the hybrid booster. The thermal margin of safety is defined as:

$$TMS = \frac{\text{(original insulation thickness)}}{\text{(erosion + pyrolysis + char thickness)}} - 1 \qquad (1)$$

The minimum acceptable TMS in the hybrid booster is 1.0.

Thermal analyses were performed at two locations in the gas generator using the charring and material ablation (CMA) computer code.[3] CMA models surface thermochemical erosion, in-depth decomposition, and temperature response for a one-dimensional axisymmetric model. Boundary conditions in the solid-fuel gas generator were calculated using pipe-flow theory (Sieder-Tate), corrected for predicted exposure times derived from the grain burnback profile.

Results of the thermal analysis predict that the insulation has a minimum thermal margin of safety of 2.75, with no temperature rise predicted in the composite case.

2.4.2 Thrust Chamber

The design requirements for the combustor and nozzle were established by modeling the combustion process. The throat diameter for the 8.62 MPa (1,253 psia) chamber pressure was calculated to be 119.4 centimeters (47 inches) with an exit diameter of 462 centimeters (182 inches). The bell-shaped nozzle has a throat-to-exit length of 470 centimeters (185 inches), and the thrust chamber diameter is 169 centimeters (66.5 inches), giving a two-to-one chamber-to-throat area ratio. The ratio of the combustor chamber free volume to the throat area, L*, was assumed to be 305 centimeters (120 inches) to minimize combustion instability. This L* value yielded a chamber length (cylinder only) of 147 centimeters (58 inches) and a residence time of 4.3 milliseconds. Figure 15 shows a sketch of the thrust chamber design.

Two types of combustion chamber designs were examined, regeneratively cooled and ablative. While either thrust chamber design could be incorporated into either of the booster design options, issues related to recoverability and reuse resulted in the grouping of high-cost components together. Thus, the regeneratively cooled thrust chamber design was only incorporated into the turbopump system design for cost and performance evaluation, and the ablative design was incorporated into both the turbopump and pressure-fed designs. As a result of our engineering trades, we selected the ablative design for our hybrid concept.

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3. Aerotherm Charring Material Thermal Response and Ablation Program, Version 3, Aerotherm Report No. UM-70-14, April 1970.

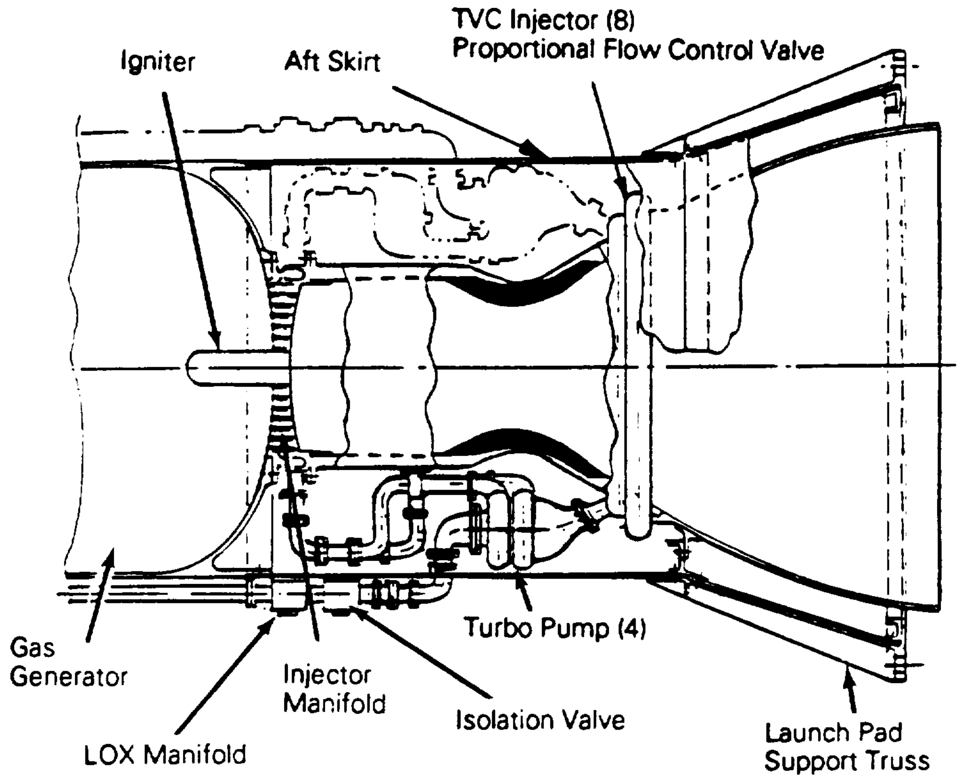

Figure 15. Thrust chamber design.

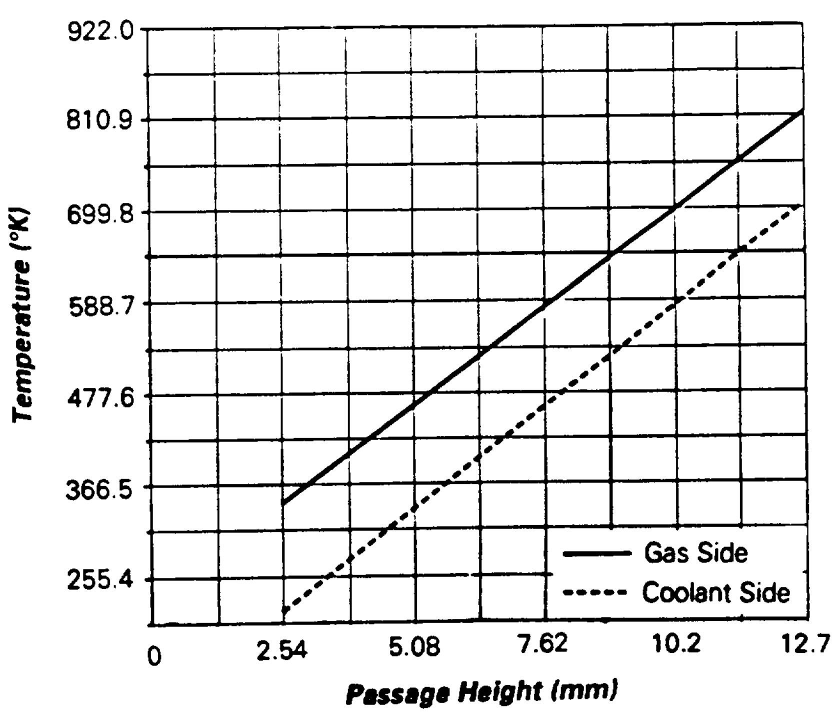

Figure 16. Throat wall temperature as a function of passage height. 0190-HYBRID-RPT

2.4.2.1 Regeneratively Cooled Design - The regeneratively cooled thrust chamber is a single piece machined from D6AC (low alloy) steel that incorporates channel wall architecture. ARC investigated the feasibility of cooling the thrust chamber with LOX. The cooling problem is complicated by the fact the oxidizer is throttled during the mission resulting in less available coolant. It was assumed the chamber would have 150 channels, and the cross-sectional area would vary axially along the thrust chamber. Figure 16 shows wall temperature as a function of throat passage height. If a passage height of 1.02 centimeters (0.4 inches) is selected, the corresponding gas sidewall temperature would be slightly under 700K (1,260°R). Figure 17 shows coolant passage pressure drop as a function of passage height. For a passage height of 1.02 centimeters (0.4 inches), the pressure drop (ΔP) in the coolant passage would be slightly under 1.38 MPa (200 psi).

Figure 18 shows coolant temperature-versus-chamber pressure for two different exhaust gas temperatures (100 and 75 percent of the uncooled temperatures). This plot shows that at 6.88 MPa (1,000 psia) chamber pressure, the LOX will be at a temperature of 136K (245°R) at a coolant passage pressure of 8.95 MPa (1,300 psia). The LOX would still be a liquid at this condition. As the thrust chamber is throttled to a chamber pressure of 3.79 MPa (550 psia), the coolant temperature is 139K (250°R) at a coolant passage pressure of 4.65 MPa (675 psia). At these conditions, the LOX is still a liquid; however, at slightly lower pressures, film boiling starts and the heat transfer coefficients would have to be determined experimentally to determine if it is still possible to cool the chamber.

Based on the previous thermal and hydrodynamic analyses, a single-pass regeneratively cooled thrust chamber was designed (Figure 19). The inlet manifold is located at the 9:1 expansion ratio, with LOX flow back to the injector manifold. From the 9:1 point out to an area ratio of 15:1, an uncooled braided carbon-carbon nozzle extension is used.

Channel wall construction was selected for the regenerative thrust chamber using a copper-based alloy plated on the D6AC steel to provide the required thermal conductivity. A number of large thrust chambers including the Space Shuttle main engine (SSME) use this approach. One possible method of construction is to start with a ring forging, spin the forging to the general shape, and then finish-machine to the required dimensions. The

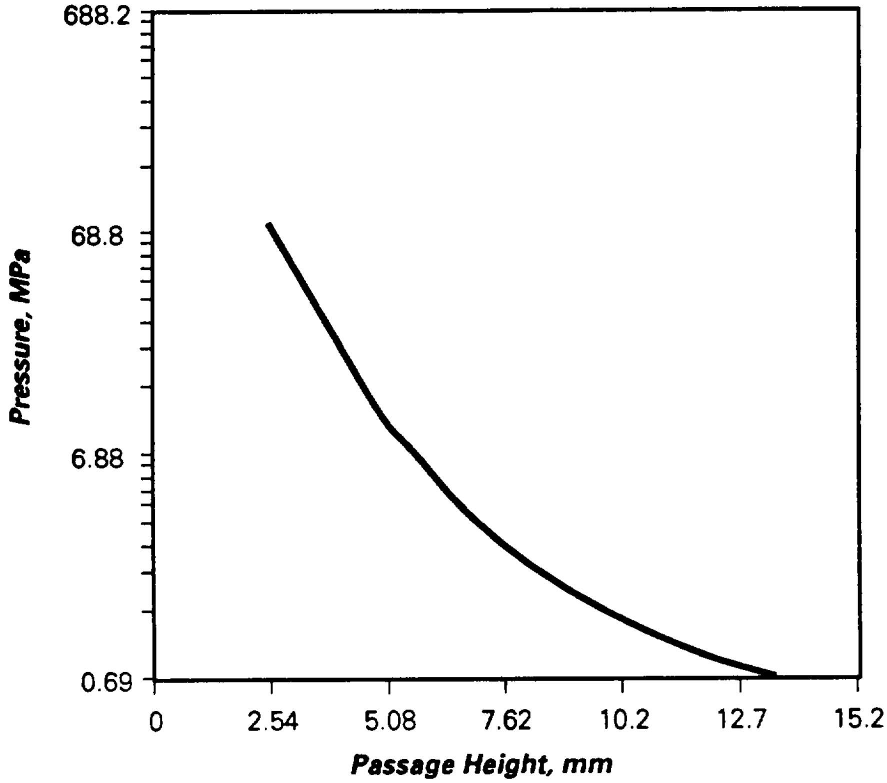

Figure 17. Coolant passage pressure drop as a function of passage height.

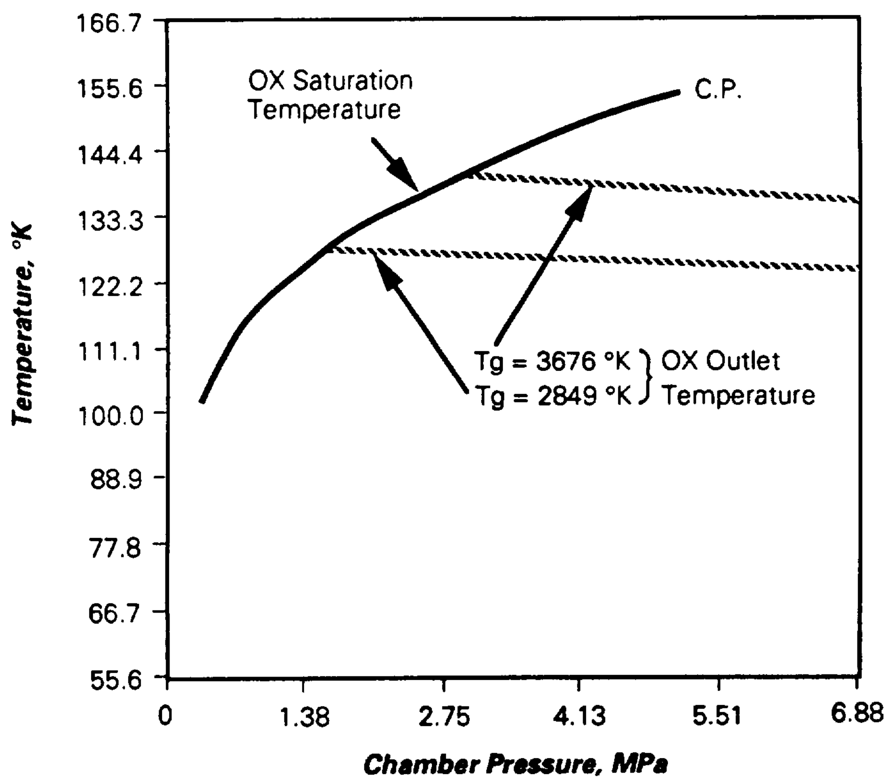

Figure 18. Relationship between coolant temperature and chamber pressure.

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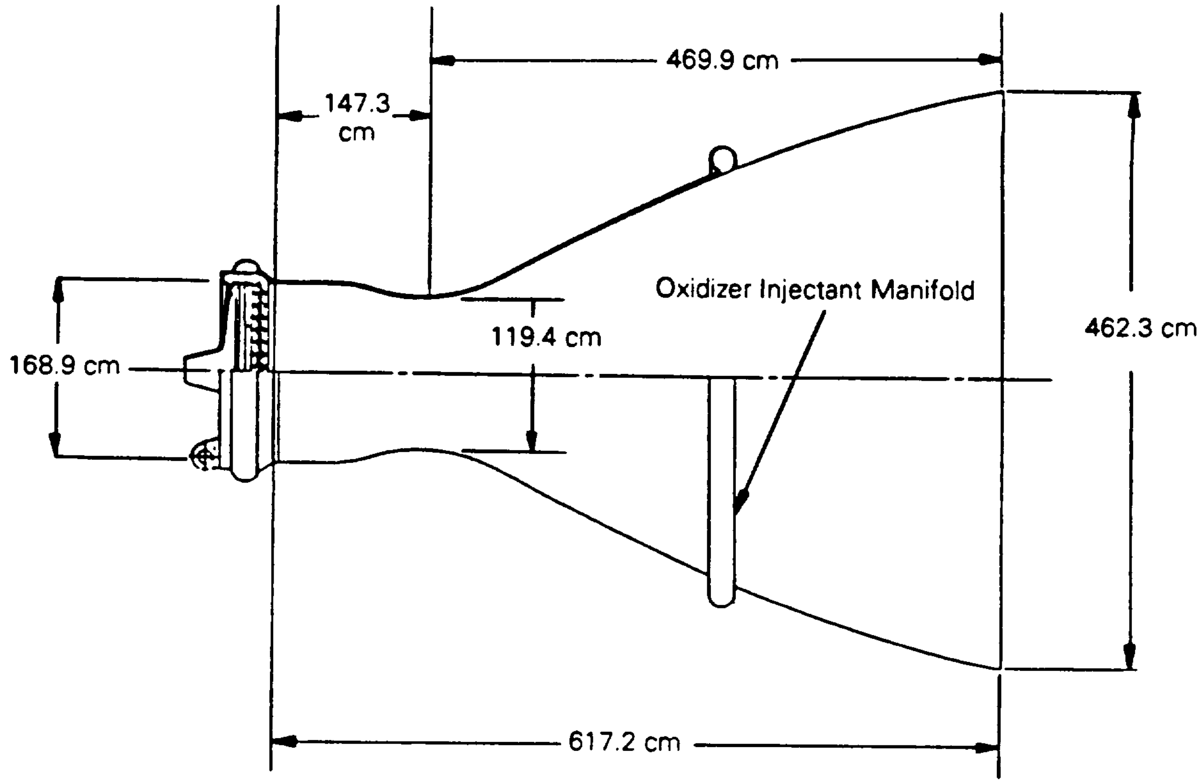

Figure 19. Single pass regeneratively cooled thrust chamber.

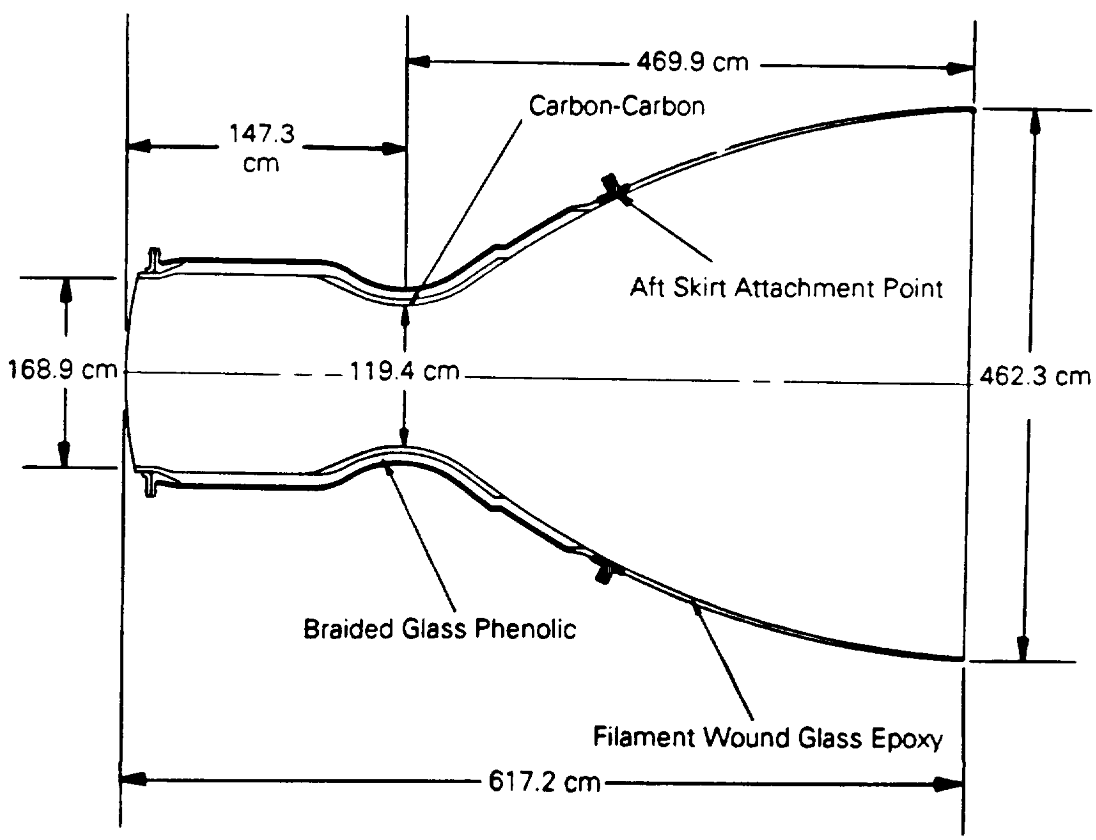

Figure 20. Ablative thrust chamber design.

channels are machined onto the outer surface of the thrust chamber and closed by electro-depositing nickel over the entire surface. It is also possible to fabricate a one-piece closure and slide it over the channels to complete the assembly.

The channel wall height at the throat is 1.02 centimeters (0.4 inches), the height at the injector end is 1.14 centimeters (0.45 inches), and the height of the manifold is 2.8 centimeters (1.1 inches). The thrust chamber was analyzed for buckling modes.

The BOSOR5 computer program for analysis of stress, stability, and vibration of segmented, ring-stiffened shells was used.[4] To meet the buckling pressure, a channel wall thickness of 1.3 centimeters (0.50 inches) was calculated.

Based upon the design, the weight of the thrust chamber assembly is as follows:

Regeneratively Cooled Portion	5,352 kg
Carbon-Carbon Nozzle Extension	998
Injector	1,134
Weight	7,484 kg (16,500 lb)

2.4.2.2 Ablative Design - The ablative thrust chamber design, Figure 20, incorporates a three-directionally (3D)-reinforced, glass-phenolic monolithic braided ablative (MBA) thrust chamber/nozzle with a 3D carbon-carbon throat insert. The MBA offers advantages over conventional laminated multi-ring designs typical of shuttle SRM nozzles in that (1) ply-lifting/delamination is eliminated via a 3D reinforced architecture, (2) leak paths due to multi-component interfaces and bondlines are reduced, and (3) manufacturing is simplified via automation, low raw material costs and reduced scrap due to near-net molding. Attachment to the injector manifold along with provision for the nozzle extension cone are integrally achieved with a filament wound overwrap of glass/epoxy.

At an expansion ratio of 5.7 aft of the throat, the flow environment is sufficiently benign to allow the glass/epoxy overwrap to perform as both

4. Buckling of Elastic-Plastic Complex Shells of Revolution Including Large Deflections and Creep, Lockheed Missiles and Space Company, Report No. LMSC-D407166, December 1974.

flame-surface and structure; therefore, the glass/epoxy is continued aft to an expansion ratio of 15. The total weight of the composite ablative thrust chamber is 8,165 kilograms (18,000 pounds).

Carbon/carbon, carbon phenolic, silica phenolic, a continuation of the glass/phenolic ablative structure, and a hybrid of silica and glass fibers were evaluated for performance in the nozzle throat region. The calculations show that carbon/carbon has better erosion resistance at the throat [1.3 centimeters (0.5 inches erosion)] than the glass phenolic MBA [10.2 centimeters (4.0 inches)] or carbon phenolic [8.9 centimeters (3.5 inches)]. The glass phenolic and silica phenolic erosion rates were unacceptably high due to the high temperature. Carbon phenolic was unacceptable due to the chemical environment resulting from an excess of free oxygen. The environment also impacts the performance of the carbon/carbon throat, but is offset by the reduction in flame temperature which has a direct effect on the kinetic reactions being modeled. The kinetic carbon reactions with water (H_2O), carbon dioxide (CO_2), and hydrogen (H_2) are directly modeled using the GASKET thermochemistry program.[5] The reaction rates are extremely sensitive to temperature. Our analyses show a two-order-of-magnitude reduction in total erosion will result at the throat when film cooling is assumed.

Boundary conditions in the thrust chamber were calculated using the results of the FLUENT computational fluid dynamics (CFD) analysis coupled with viscous flow boundary layer solutions calculated by the momentum energy integral technique (MEIT).[6,7] The CFD analysis was used to predict the reduction in the gas temperatures at the boundary layer due to annular fuel injection at the manifold. The results of the analysis show a significant reduction in the gas temperatures at the wall ranging from a 2,478K (4,460°R) reduction at the thrust chamber to a 1,144K (2,060°R) reduction at the nozzle throat.

Charring, material and ablation (CMA) analyses were performed at five locations in the nozzle and combustion mixing chamber. The oxygen content in

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5. Aerotherm Graphite Surface Kinetics Computer Program, Version B, December 1978, AFRPL-TR-78-77.

6. Creare Incorporated, "Fluent Manual," Version 2.9, TN-369, Rev. 3, 1987.

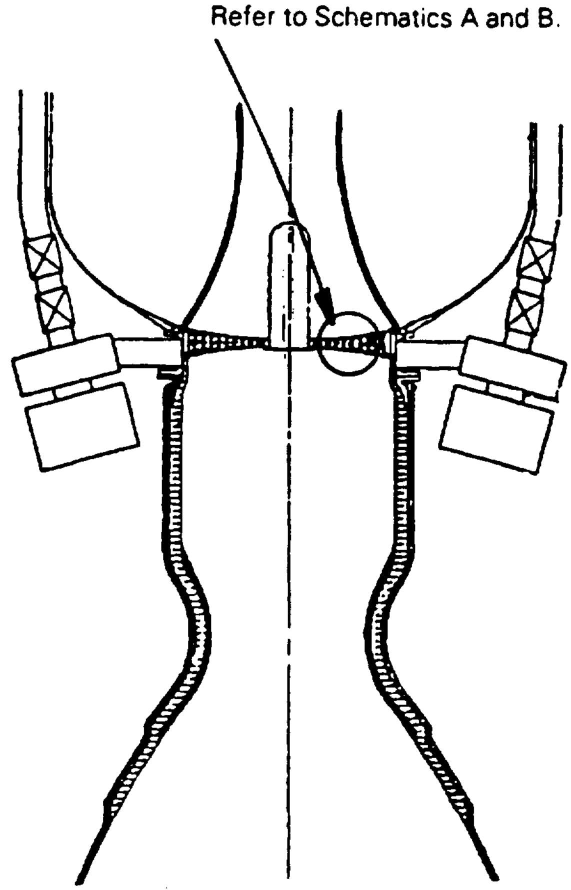
7. Momentum/Energy Integral Technique, July 1978, AFRPL-TR-78-53.

the combustion gasses is three times what is present in conventional solid propellants with flame temperatures in excess of 3,589K (6,460°R). In the absence of film cooling, wall component materials are subjected to a chemically reactive environment resulting in erosion of the glass fibers used in the MBA liner. The analysis performed in the combustion chamber shows that with film cooling and fuel injection, there will be minimal erosion of the glass MBA. Our point design is dependent on film cooling using unreacted gas generator effluent.

The effects of film cooling are no longer significant beyond an area ratio of 1.7; however, the static temperature drops sufficiently at an area ratio of 2.9 to allow transition back to the glass MBA. The composite overwrap forms the exit cone at an area ratio of 5.7. The minimum predicted thermal margin of safety of 3.15 occurs at the transition between the carbon/carbon insert and the quartz/phenolic MBA.

2.4.3 Injector Design

The injector (Figure 21) consists of the thrust chamber dome and the central dome segment of the gas generator, joined along their common perimeters by an oxidizer supply plenum. Fuel-rich combustion products pass from the gas generator into the thrust chamber via 500 injector tubes. Each tube is 3.9 centimeters (1.55 inches) in diameter and passes through both the upper and lower dome elements. The fuel ports are designed for a maximum pressure drop of 0.38 MPa (55 psi). In the thrust chamber dome, eight pairs (doublets) of oxidizer injectors are spaced about each of the fuel ports. Each oxidizer port is 0.33 centimeters (0.131 inches) in diameter and is designed for a maximum differential pressure of 1.72 MPa (250 psi). Each pair of oxidizer ports is angled for self-impingement of the streams for ligament breakup and atomization. In addition, the doublet pair is angled inward toward the stream of gases flowing out of the fuel injector port so that the atomized oxygen stream will impinge and mix with the fuel stream. Since the fuel stream is relatively warm [about 1,278K (2,300°R)] at the selected mixture ratio, the finely atomized LOX will vaporize and react with the fuel-rich gas stream. A preliminary evaluation of thermal loads on the injector indicate that at the specified mixture ratio and LOX pressure, the oxidizer will remain a liquid. A more detailed evaluation will have to be completed once the injector design has been finalized.



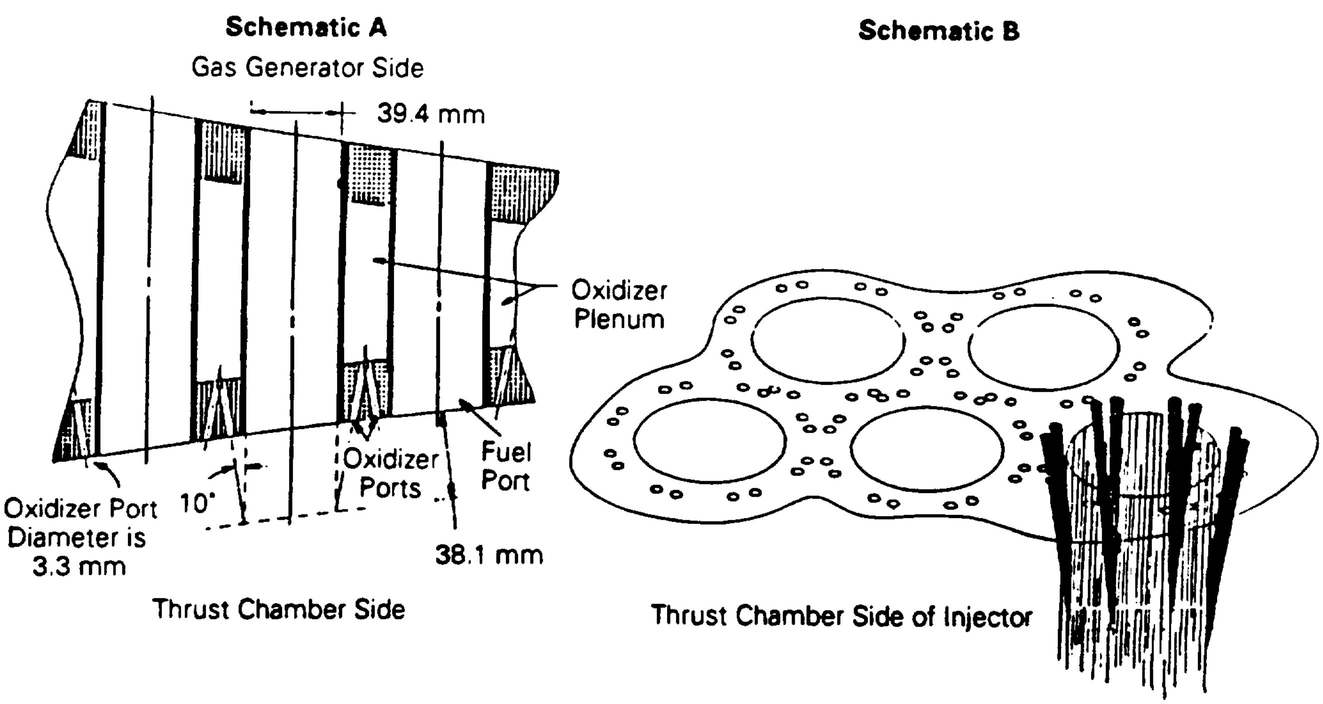


Figure 21. Injector manifold design.

The fuel ports at the outer periphery of the injector manifold will not be surrounded by oxidizer ports. This will provide a zone of combustion products along the wall of the thrust chamber. The resulting reduction of gas temperature and oxygen concentration near the wall will reduce cooling requirements for the regeneratively cooled chamber and erosion rates in the ablatively cooled chamber. Previous experience with film cooling of this type has demonstrated only minimal penalties in combustion performance.[8]

The flow of the gaseous fuel through the injector ports is designed to be subsonic. This unchoked injector allows pressure changes in the thrust chamber to be communicated to the gas generator. Since the gas generator burning rate is a function of pressure, the fuel flow rate is controlled by adjusting chamber pressure, which in turn is accomplished by varying the turbopump speed and, thus, the LOX flow rate.

The injector is fabricated from stainless steel. At the thrust chamber attachment area, the stainless is reinforced with a nickel alloy. The regeneratively cooled thrust chamber attachment area has a nickel coating deposited on the copper-based alloy. The injector is welded to the regeneratively cooled thrust chamber and bolted to a flange on the ablatively cooled thrust chamber. As configured, the injector manifold is estimated to weigh 1,134 kilograms (2,500 pounds) for both designs.

The proposed injector design offers two advantages: (1) the liquid rocket fuel injection development experience is applicable; and (2) injector development can be performed using subscale test motors, and then scaled up. Injector designs will be evaluated in the acquisition phase, Phase 2 of this program.

2.4.4 Combustion Stability

A preliminary evaluation of combustion stability was made to identify issues that need to be addressed. During the evaluation, four characteristics of the design were noted that will provide benefits:

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8. Liquid Rocket Engine Fluid-Cooled Combustion Chambers, NASA SP-8087, April 1972.

1. High-solids-loading in the thrust chamber is known to be an effective damping agent for high-frequency instabilities. The products of combustion of the fuel-rich gas generator propellant are approximately 50 percent-by-weight solid particulates with the particle size distribution ranging from 1 to 400 microns.
2. The free volume of the gas generator is larger than that for the thrust chamber. This minimizes the effects of pressure oscillations originating in the thrust chamber.
3. The injector fuel port area is smaller than the characteristic dimensions of the thrust chamber. This lower flow area will dampen the low frequency pressure oscillations between the gas generator and thrust chamber.
4. The oxidizer injection system has been designed for a 25 percent pressure drop across the injector face to minimize effects of thrust chamber pressure oscillations on oxidizer flow rate.

2.4.5 Oxidizer Delivery System

An engineering trade study was performed by ARC/Liquid Propulsion on eight systems for the storage and control of oxidizer for hybrid combustion. The study was of sufficient detail to make major feed system selections. Results of the trade study are discussed in detail in Appendix A. Components incorporated into the point design are presented below.

2.4.5.1 Pressure-Fed System - A Tridyne system was selected for pressurization of the oxidizer tank. Tridyne is a mixture of 91 percent helium and a stoichiometric ratio of hydrogen and oxygen. The Tridyne is stored at ambient temperature and at a pressure of 68.9 MPa (10,000 psia). When Tridyne is flowed through a catalytic bed, the hydrogen and oxygen react, producing a mixture of helium and water vapor at 667K (1,200°R). Parallel regulators, upstream of the catalytic bed, establish the head pressure on the oxidizer tank. The oxidizer flow rate is modulated by four throttling valves, one in each of the four 20.3 centimeter (8 inch) diameter supply lines. The lines are prefilled to the normally closed isolation valve located in each feedline and near the injector manifold. The isolation valves in the gas pressurization outlet lines are opened just before ignition to pressurize the oxidizer tank. Booster shutdown is accomplished by closing a normally open isolation valve located in the common oxidizer plenum at the base of the tank. The feed system has been sized to provide 100 percent of the required LOX flow, even with a failure of one of the four feedlines. Figure 22 shows a schematic of the delivery system.

A total of 3,601 kilograms (7,938 pounds) of Tridyne is required. The Tridyne tank is fabricated of IM-7 carbon fiber with an epoxy resin. The tank wall is 19.5 centimeters (7.68 inches) thick and includes a 0.08 centimeter (0.03 inch) aluminum liner. The total tank weight is 9,843 kilograms (21,701 pounds) (tank and liner), and the Tridyne feed system weighs 150 kilograms (330 pounds).

2.4.5.2 Turbopump Feed System - A schematic of the LOX delivery system is given in Figure 23. Four turbopumps were used, with each having a maximum operating capacity equal to 133 percent of the normal operating requirement. This permits delivery of the required LOX flow even if one of the four turbopumps fails. Fuel-rich gases from the gas generator are sent through parallel throttle valves to power the turbines. These throttle valves can be closed in the event of an emergency shutdown. The normally open isolation valve just upstream of the catalytic gas generator is also used for an emergency shutdown. The turbine exhaust is passed through a separate nozzle and expanded to ambient pressure conditions.

A Tridyne pressurization system was used for the turbopump feed system to provide a constant head pressure to the suction side of the pumps. The Tridyne is controlled by two isolation valves (Figure 23). Each valve is capable of handling full gas flow in case one isolation valve fails to open. A pressure transducer is provided so pressure in the tank can be monitored.

The Tridyne flows to a normally open isolation valve through a gas regulator to a catalytic gas generator where the oxygen and hydrogen react to heat up the helium. The products entering the LOX tank are heated helium and steam. A second regulator is provided in parallel with the first and is connected to a normally closed isolation valve. In case the first regulator malfunctions, this isolation valve can be opened, the isolation valve with the malfunctioning regulator can be closed, and the system will continue to operate. Regulators with built-in health monitoring systems will be used, and the switchover will occur automatically with no outside signals required.

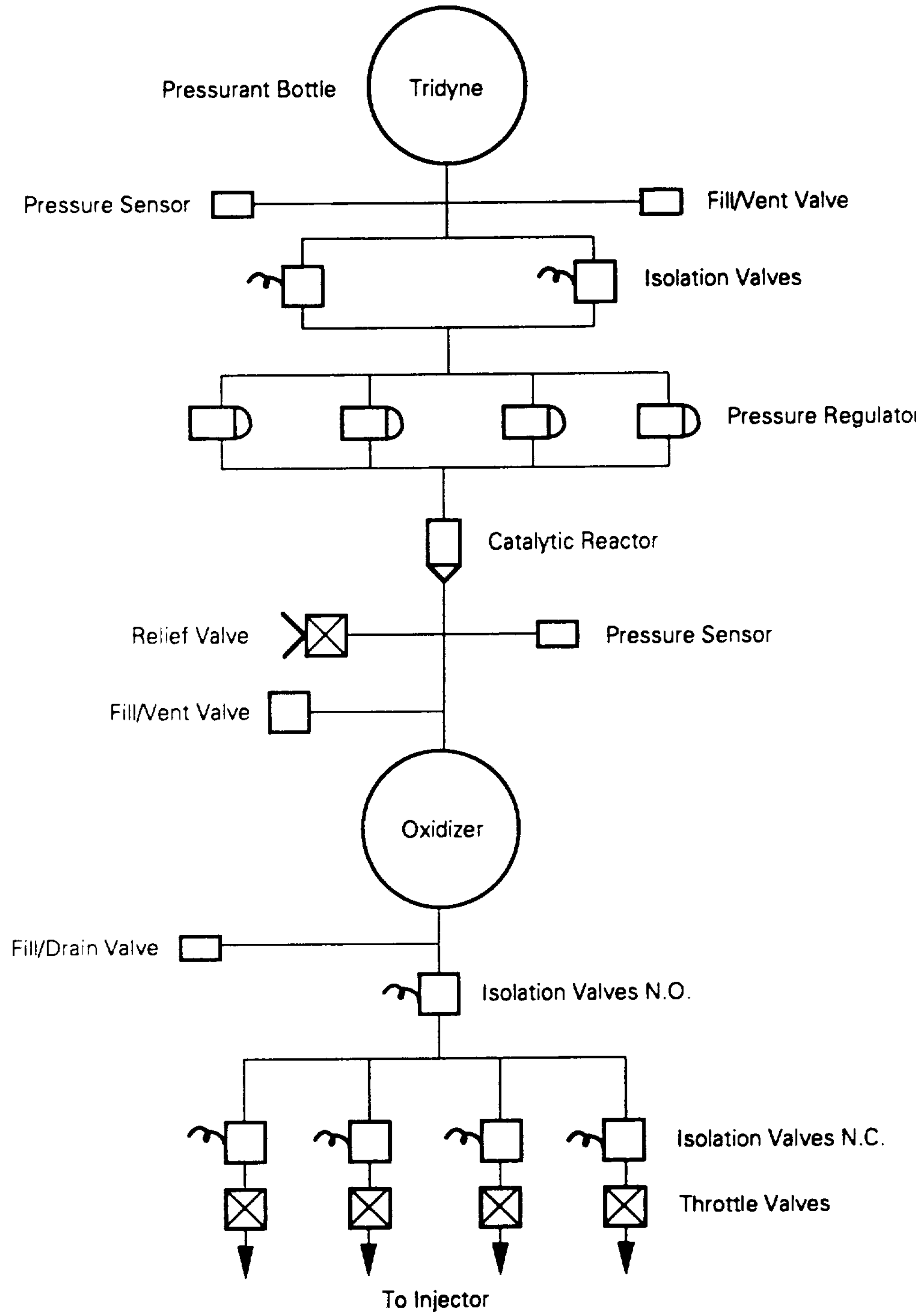

Figure 22. Catalytic warm gas pressurization system schematic for GG/LOX.

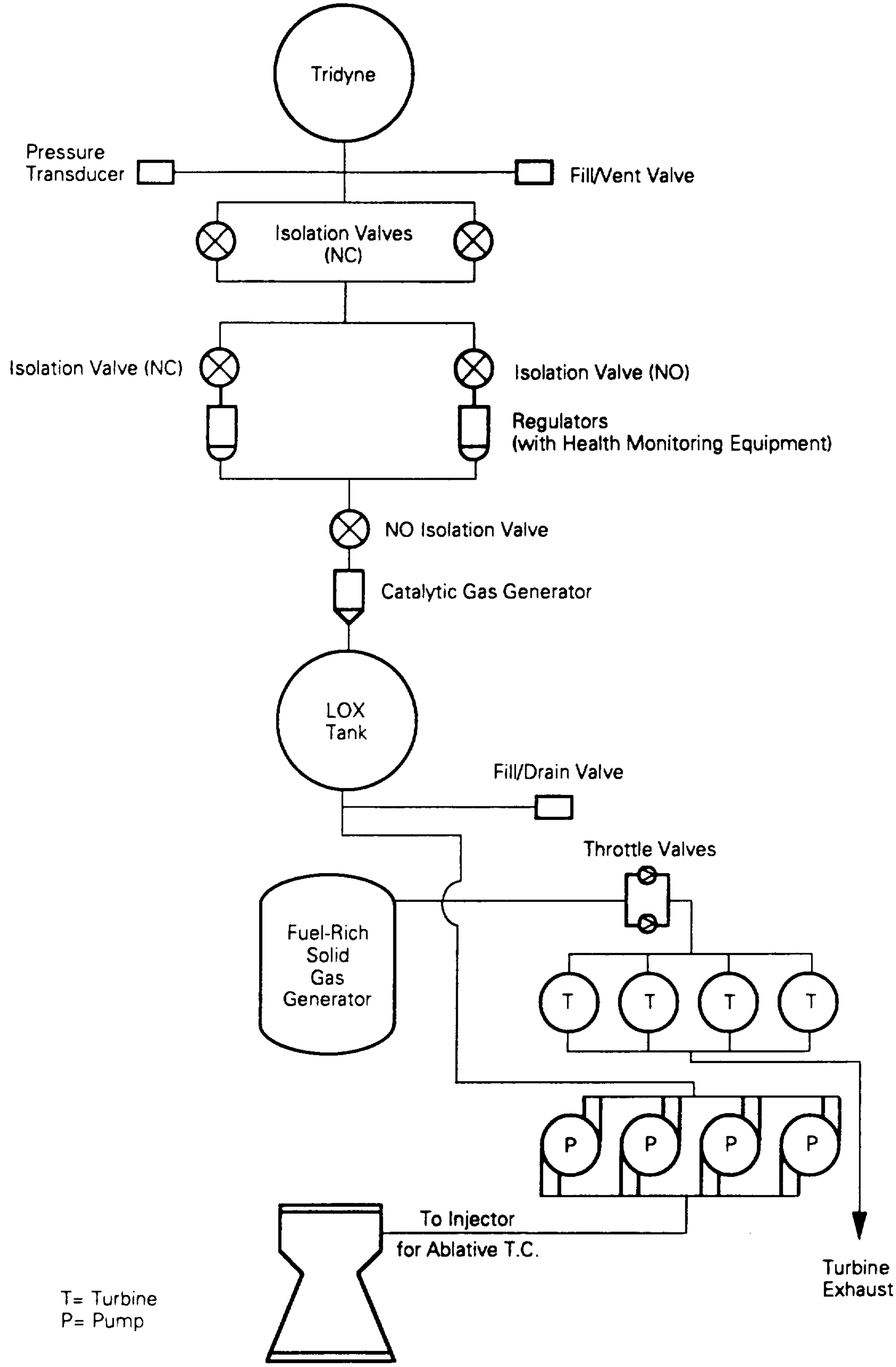

Figure 23. *Preliminary oxidizer pump-fed system schematic.*

Since the fluid to be pressurized is a cryogen, the steam generated will liquify and eventually freeze. This will not create any problems until the oxidizer tank is almost empty. We have increased our ullage volume and loaded more oxidizer to prevent entrainment of the water.

Oxidizer is routed directly to the pump inlet and from the pump outlet to either the injector (ablative thrust chamber) or to the cooling jacket inlet (regeneratively cooled thrust chamber).

A system pressure schedule is shown in Table 9. This schedule covers the ablative- and regeneratively cooled thrust chamber cases.

Our turbopump design, provided by Allied-Signal, is driven from the fuel-rich gas generator. The pumps were required to have a wide throttling range to supply the LOX flow rate throughout the burn and to accommodate a potential one-pump-out operating condition. A list of operating requirements is given in Table 10. The maximum pump outlet pressure is 9.46 MPa (1,375 psia) for the ablative thrust chamber and 11.5 MPa (1,675 psia) for the regeneratively cooled chamber. The higher delivery pressure for the regeneratively cooled thrust chamber is due to the pressure drop taken through the coolant channels.

Since the gas generator exhaust contains solid particulates, a method of separating the particulates from the gas stream was required to improve the turbopump reliability. Allied-Signal accomplished this by using a reverse pitot, inertial filter, developed and proven in cooling turbine applications. The reverse pitot, Figure 24, extends into the gas flow with the open end of the probe directed downstream. Flow entering the probe is forced to turn 180°. The momentum of the particles prevents them from being entrained, and they are separated from the flow. A well-designed probe will remove approximately 99 percent of the solid particulates. Four probes would be used, one feeding each of the four turbopumps. The probes would be made an integral part of the fuel injector manifold to simplify case construction, and would be fabricated from an austenitic stainless steel to survive the moderate effluent temperature.

The Allied-Signal turbopump is shown in Figure 25. The pump is a single-stage, mixed-flow design with a 22.9 centimeter (9 inch) impeller tip diameter [23.6 centimeters (9.3 inches for the regenerative option)]. The turbine uses a single-stage, impulse impeller with a 48.3 centimeter (19 inch) tip

Table 9. System Pressure Schedule.

	Pressure MPa
Tridyne Storage Pressure at 289K	68.8
Regulator Outlet Pressure	1.8
Catalytic Gas Generator Pressure	1.8
Tank Pressure*	0.8 (min)
	0.9 (max)
Inlet Pressure to Pump	0.4
Pump Outlet Pressure (Ablative)	9.5
Pump Outlet Pressure (Regen)	11.5

*Includes static head. Minimum tank pressure is 0.4 MPa.

Table 10. LOX Turbopump Operating Requirements.

Maximum Flow Rate (kg/sec)	3,144
Pump Inlet Pressure (MPa)	0.8
Pump Outlet Pressure (MPa)	9.5
	11.5[1]
Turbine Drive-Gas Flow Molecular Weight[2]	13.75
Turbine Drive-Gas Ratio of Specific Heats[2]	1.12
Turbine Inlet Pressure (Main GG Maximum Chamber Pressure) (MPa)	7.5
Turbine Inlet Temperature (GG Chamber Temperature) (K)	392
Turbine Discharge Pressure (MPa)	0.2
Minimum Flow Rate (kg/sec)	1,895
Minimum Chamber Pressure (MPa)	3.5
Four Turbopumps with Single Pump Out Capability	

1. Regenerative cooling version.
2. Assuming solids are filtered out using reverse pitot.

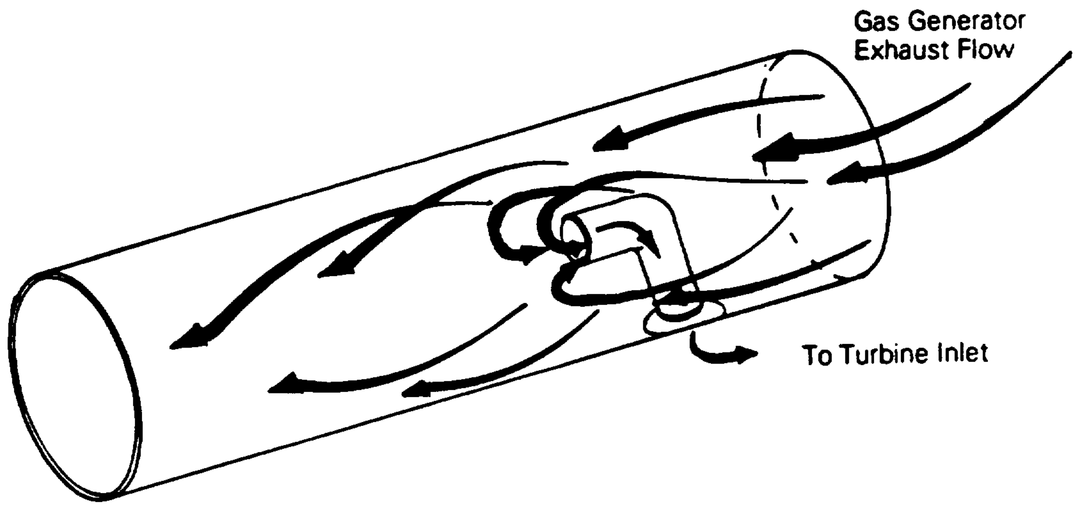

Figure 24. Reverse pitot.

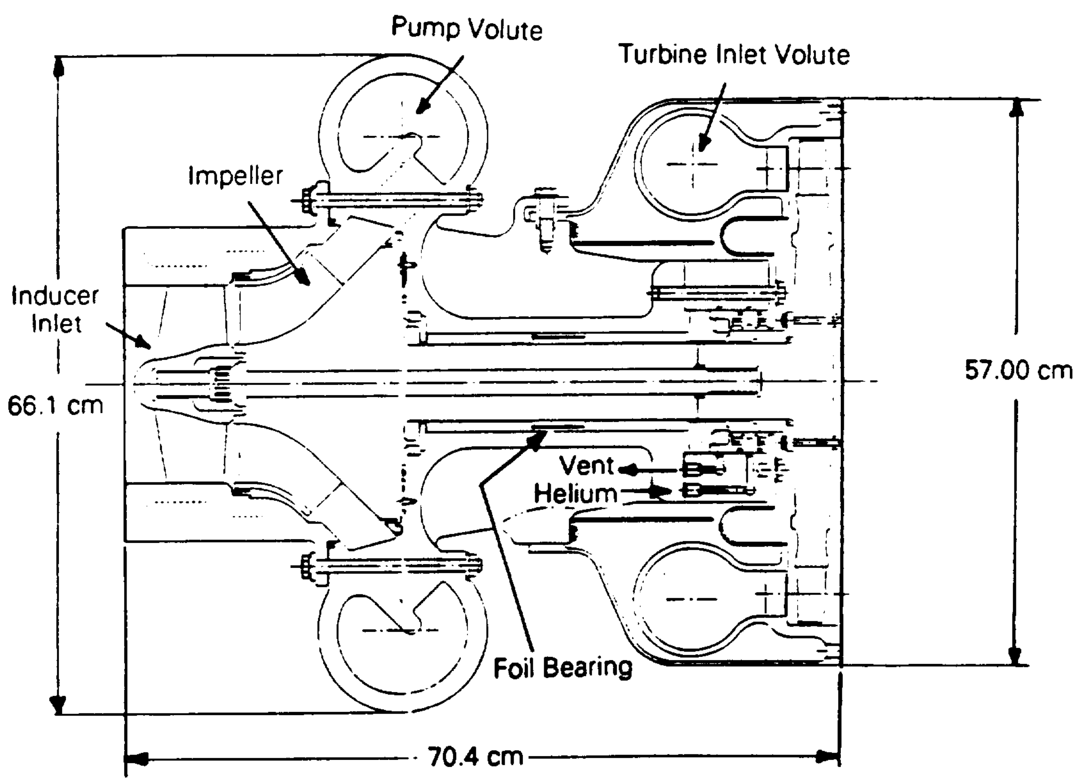

Figure 25. LOX turbopump.

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diameter. The pump, turbine housing, and turbine impellers are fabricated of Inconel 718 (77 percent N, 15 percent Cr, 0.2 percent CO, 7 percent Fe, 3 percent Al). The inducer is fabricated from Monel K-500 (63 percent N, 30 percent Cu, 2 percent F3, 4 percent s, 2.75 percent Al, 0.9 percent Mn) for its good erosion resistance. The estimated turbopump weight is 204 kilograms (450 pounds).

The design uses foil bearings rather than conventional ball bearings. Ball bearings have caused several failures in LOX turbopumps.[9] Foil bearings offer stable, high-speed operation at extreme temperatures where ordinary lubrication systems cease to function properly. In addition, foil bearings do not have the clearance and rotor stability problems associated with hydrostatic bearings, giving them unique advantages in the LOX turbopump application. Excellent reliability has been achieved for foil bearing machines used in other applications. The mean-time-between-failures for foil bearing cooling turbines is typically over 60,000 hours. The foil bearings are made of Inconel 750 with a Teflon coating. Silver plating is used wherever rubbing is likely to occur, such as the labyrinth seal and at the balance piston stationary lip areas.

There are several seals around the rotating assembly of the LOX turbopump to ensure efficient and safe operation. A labyrinth seal is used on the impeller shroud to control the leakage from the high-pressure outlet to the pump main stage inlet. The seal clearance is determined by considering the combined effects of static hydraulic unbalance load deflection, vibration runout, and differential thermal and centrifugal growth. The number of knife edges of the labyrinth seal control the amount of leakage. The stationary seal land is plated with silver, which offers good ignition resistance and reduces the danger of burnishing if localized contact occurs between the knife edges and the seal land.

A carbon face seal near the right journal bearing, inboard of the turbine wheel, is used as a spring-loaded static seal during chill-down. This seal prevents liquid oxygen from leaking into the turbine cavity during starts. During operation, this face seal lifts off and creates a finite clearance that

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9. Personal Communications, Dr. Alston L. Gu, Turbomachinery Systems, AiResearch Los Angeles Division, Torrence, California, August 1989.

controls the bearing cooling flow. Radial grooves may be utilized in the face seal to promote lift-off.

The bearing cooling flow is prevented from entering the turbine cavity by a drain between the face seal and a helium-purged, carbon, floating-ring seal. Another floating-ring seal is utilized to the right of the helium inlet to control the helium flow to the turbine cavity. The finite clearances of the floating-ring seals are determined by the desired leakage rates and the effects of differential thermal and centrifugal growth of the components.

The performance of the turbopump at the normal maximum flow, and during pump-out conditions is presented in Table 11. The flow rate for the pump-out conditions is 33.3 percent higher than that of the maximum flow point.

At the pump-out condition, the total gas generator chamber pressure of 7.5 MPa (1,085 psia) is used to drive the turbine; while at the maximum flow point, this pressure level is throttled to 4.3 MPa (618 psia) for the ablatively cooled version and 5.2 MPa (760 psia) for the regeneratively cooled version. Turbine efficiency is limited by the turbine tip speed. To achieve high reliability, the maximum turbine top speed allowed (17,440 rpm) is 457 meters/second (1,500 feet/second).

Turbopump performance was evaluated at four selected points in the booster duty cycle, Figure 26. The purpose of the evaluation was to ensure that the turbopump design had an adequate performance margin. Table 12 presents the study results. The available pressure to the turbine inlet from the gas generator bleed is above that necessary to deliver the required LOX flow rate. A throttling valve will be located in the turbine inlet to reduce pressure. The required gas bleed from the gas generator is estimated at 617 kilograms (1,360 pounds) for each turbopump. This translates to approximately 4,990 kilograms (11,000 pounds) of extra propellant to power the four turbopumps.

2.4.6 LOX Tank

LOX tank designs were developed for both the pressure-fed and turbopump booster options. The total LOX carried is 299,700 kilograms (660,725 pounds). A summary is:

Mission Requirement	281,681 kg
FITVC Requirement	12,143
2-Percent Reserve	5,876
Total	299,700 kg

Table 11. Maximum Flow and Pump-out Performance.

Pump	Maximum Flow	Pump-Out Condition
LOX Flow, kg/sec	786	1,048
Efficiency	0.84	0.76
Required Power, W ($x10^5$)	70.8 (87.6)[1]	104.4 (128.9)[1]
Turbine		
Turbine Inlet Pressure, MPa	4.3[2] (5.2)[1]	7.5
Turbine Flow, kg/sec	7.3 (9.0)[1]	10.4 (12.8)[1]
Efficiency	0.42	0.48
Speed, rpm	16,000	17,440

1. Regenerative cooling version.
2. Throttled down from GG chamber pressure of 7.5 MPa (1,085 psia).

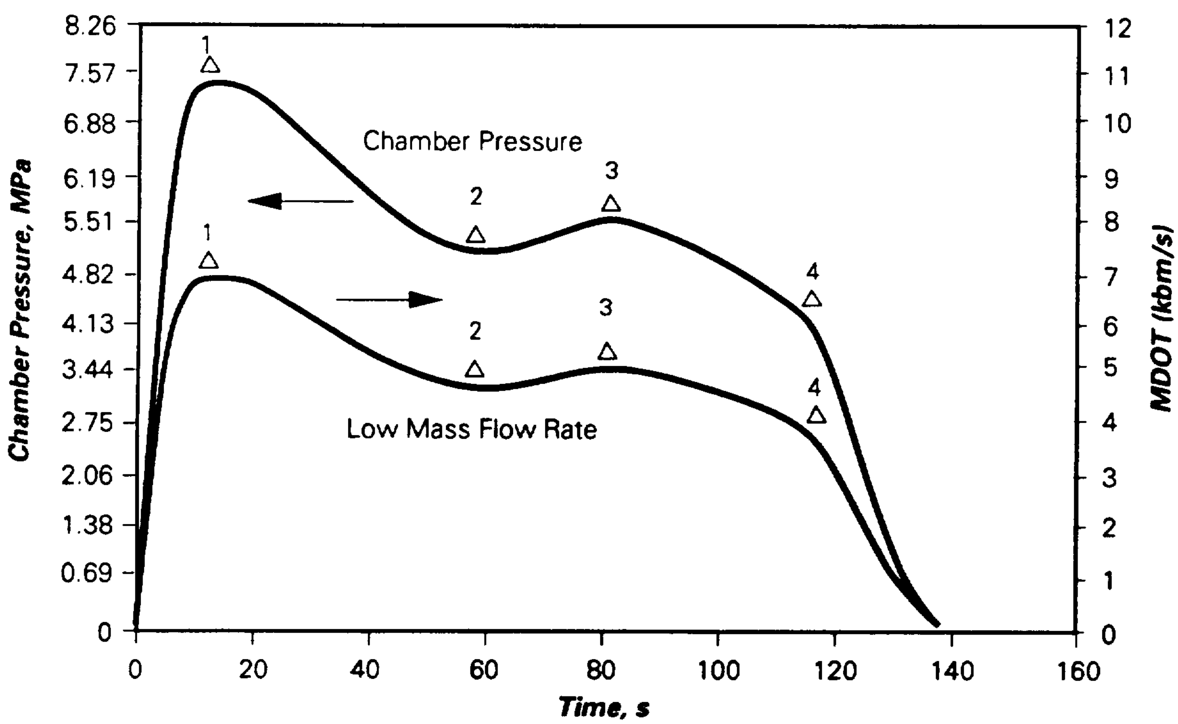

Figure 26. Selected turbopump operating points.

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Table 12. LOX Turbopump Transient Performance.[1]

Location on Duty Cycle	PT1	PT2	PT3	PT4
Flow Rate, kg/sec	3,144	2,177	2,359	1,905
Flow Rate Per Pump, kg/sec	786	544	590	476
Chamber Pressure, MPa	7.5	5.2	5.6	4.4
Pump Outlet Pressure, MPa[2]	9.5	6.5	7.1	5.6
Pump Efficiency	0.84	0.75	0.76	0.74
Speed, rpm	16,000	12,640	13,280	11,520
Required Power, W x 10^5	70.8	36.0	42.0	27.3
Turbine Inlet Pressure, MPa[3]	4.3	2.8	3.1	2.3
Turbine Efficiency	0.43	0.37	0.39	0.35
Turbine Flow, kg/sec[4]	7.3	4.8	5.3	3.9

1. Using solid propellant gas generator fluid to drive turbine. Ablative cooling version.
2. Assuming 26.7 percent higher than chamber pressure.
3. Throttled down from chamber pressure.
4. Total turbine flow for whole duty cycle is estimated to be 617 kilograms (1,360 pounds).

The LOX tank storage requirement is 255 cubic meters (9,003 cubic feet). This is calculated from the density of LOX at its storage temperature of 78K (140°R), a 3 percent allowance for ullage, and the assumption the the LOX feed manifold is prefilled to the isolation valves (2.0 cubic meters).

2.4.6.1 Pressure-Fed Option - A filament-wound composite tank was selected for the pressure-fed LOX system to minimize the system weight and, therefore, keep the life cycle cost competitive with the pump-fed systems. For the preliminary design, Hercules IM-7 carbon fiber [strength 5,402 MPa (785 ksi), modulus 275,283 MPa (40 msi), strain 1.85 percent)] was evaluated with two resin systems: epoxy-based EPON 826, and polyimide. Final selection of the materials will require additional engineering analysis and testing. A 419 centimeter (165 inch) tank diameter was selected to reduce the length to 1,981 centimeters (780 inches). The tank pressure was calculated to have a 12.3 MPa (1,793 psia) MEOP based on the pressure drop through the system. Structural analysis included the effects of bending loads at launch caused by the launch "twang" experienced by the shuttle and the loads imposed by the 12.3 MPa (1,793 psia) MEOP. Shuttle-type axial stiffness requirements were not applied because thrust reaction to the core vehicle is accomplished at the aft end of the hybrid booster rather than the forward end. The case thickness designed to accommodate the structural loads is 3.4 centimeters (1.36 inches), which yields a case weight of 9,740 kilograms (21,474 pounds). This is approximately 27 percent higher than a case not designed for shuttle-type bending stiffness requirements.

Several LOX tank liner materials were considered. Aluminum was a primary candidate, but it complicates the tank fabrication process and contributes significant weight for a nonstructural member. Several elastomeric materials such as Upilex, Teflon, and Kapton were also evaluated. Upilex, a polyimide film, was selected because it has good elongation properties and can meet the range of thermal requirements. A thickness of 0.008 to 0.01 centimeters of Upilex is estimated to be adequate. The liner will be layed up on the winding mandrel before manufacturing the oxidizer tank.

The composite tank will experience cryogenic temperatures down to 78K (140°R) due to the LOX storage. During the flight, pressurization gas at 667K (1,200°R) will replace the oxidizer. The tank wall temperature rise was calculated using ARC's trapped-gas thermal response model.[10] The predicted maximum temperature on the inner wall is 439K (790°R), which is well within the capabilities of the composite.

The composite LOX tank will be monolithic with steel polar bosses. The LOX feedlines will branch off a single exhaust port in the aft boss. Pressurization lines will enter the forward boss. Anti-slosh baffles will be integrally wound into the case.

2.4.6.2 <u>Turbopump Option</u> - The point design for the turbopump option incorporates an aluminum-lithium LOX tank. Aluminum-lithium offers weight and

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10. Spear, G. B., Developed by ARC in 1982 for gas generator modeling of variable heat loss due to mass flow.

strength advantages over conventional aluminum fabrication; however, it is also more expensive. The tank was sized using the same general dimensions and loads as the pressure-fed tank, with the exception that the pressure within the tank was assumed to be 0.5 MPa (75 psia). Structural analysis yielded a wall thickness at the top of the tank of 0.29 centimeters (0.11 inches) and 0.73 centimeters (0.286 inches) at the bottom. A 434 centimeter (171 inch) tank diameter was selected to improve the overall packaging of the components and reduce the booster length. The estimated tank weight is 4,213 kilograms (9,287 pounds). A liner is not required for this application.

For the manufacture of the cylindrical section of the tank, three monocoque options were identified as applicable based on Boeing's experience. The first option incorporates a spot-welded, internal "Z" stringer. The second option has a laser-welded internal "L" stringer. The final option uses spot-welded, external trapezoidal hat-section stringers, Figure 27.

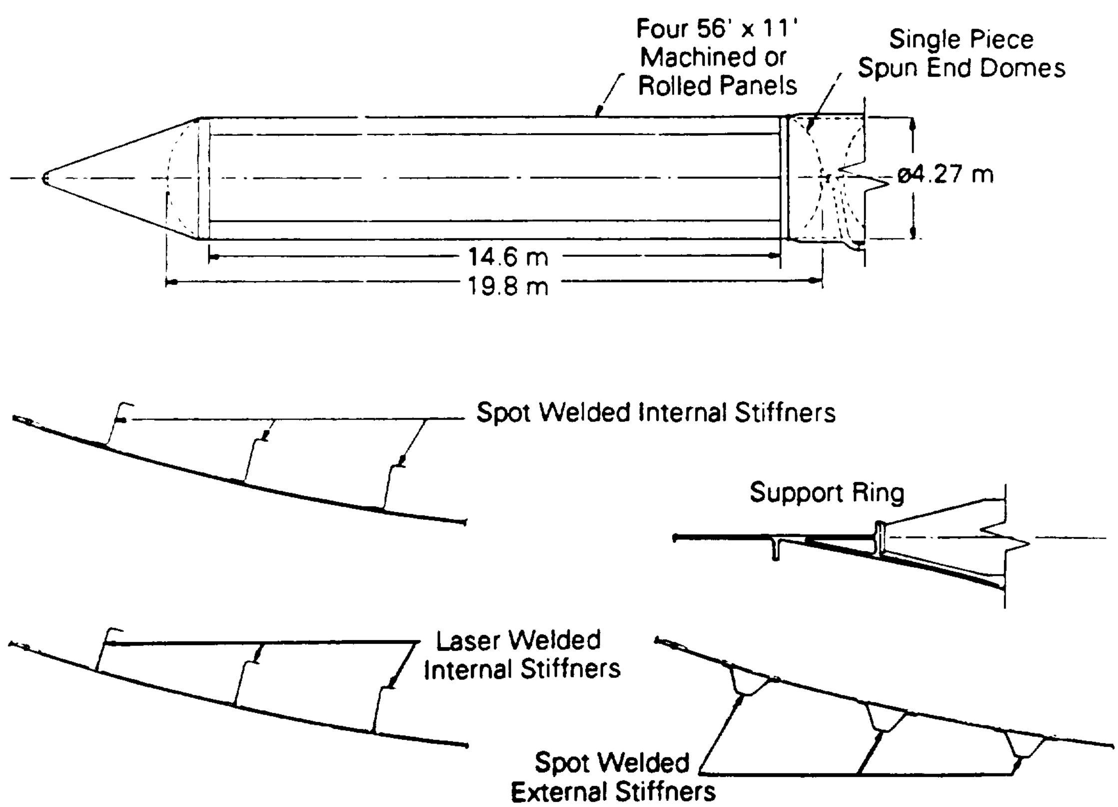

Figure 27. Aluminum-lithium LOX tank design.

2.4.7 Thrust Vector Control

Three methods of performing thrust vector control were evaluated for the hybrid booster design: (1) thrust chamber gimballing; (2) nozzle vectoring; and (3) fluid injection. Large vectoring angles (8 to 10 degrees) are often required to compensate for thrust mismatch between a pair of solid rocket boosters. Thrust mismatch between hybrid booster pairs could be corrected by differential throttling of the oxidizer in either booster. The remaining deflection requirement, driven by a number of factors such as core vehicle geometry and center-of-gravity (C.G.) shift, is in the two to three degree range.

Gimballing of the thrust chamber, commonly utilized in large liquid boosters, cannot be readily incorporated in the hybrid booster design. The flow of large volumes of hot gas from the gas generator to the thrust chamber complicates the design of a gimballed thrust chamber. Due to the complexity of the design, the gimballed approach was excluded from further consideration.

Vectoring of the nozzle is common practice in large SRBs such as the shuttle SRB, because it provides large (9 to 10 degree) deflection angle capability. While this approach is applicable to the hybrid booster design, it adds weight and cost to the design and reduces reliability based on historical data.

Fluid injection thrust vector control (FITVC) can provide 2 to 3 degrees of deflection angle and potentially provides higher calculated reliability than vectored nozzle designs. This concept was pursued as our baseline TVC approach.

Since the overall vehicle configuration, including core vehicle, is undefined, TVC requirements could not be absolutely defined at this time. For purposes of sizing an FITVC system, a set of requirements was established by reviewing typical shuttle SRB duty cycles and compensating for the elimination of thrust mismatch. The assumed TVC system design requirements are summarized in Table 13. These requirements were used as the basis for the conceptual design.

Table 13. Hybrid Booster TVC Design Requirements.

<u>Performance Requirements</u>

Maximum Thrust Deflection (deg)	3-5
Dynamic Response	
- Frequency Response (-3db) (Hz)	4
- Slew Rate (deg/sec)	5
Resolution (deg)	0.05
Duty Cycle (deg/sec)	150

<u>Program Design Priorities</u>

1. Safety/Reliability
2. Cost
3. Performance

Early emphasis in the study centered on the choice of injectant to be used for the system.[11] Three injectant candidates were evaluated for feasibility: (1) LOX bled off the turbopump outlet; (2) solid fuel exhaust bled off the gas generator; and (3) a hybrid approach that combines LOX bled from the turbopump and solid fuel exhaust bled from the gas generator at a fixed mixture ratio. Figure 28 shows the injectant usage estimates for the three candidates assuming a total hybrid booster mass flow rate of 3,969 kg/sec (8,750 lbm/sec), corresponding to 80 seconds into the duty cycle [1,588 kg/sec (3,500 lbm/sec) fuel flow, 2,381 kg/sec (5,250 lbm/sec) LOX flow, 1.5:1 MR)].

Although all three injectants are effective, the hybrid FITVC approach is the most efficient in terms of propellant usage, followed by LOX-only and fuel-only, respectively. The propellant usage estimates are based on empirical data for secondary injection systems.[12,13] Figure 29 shows the ratio of

11. W. G. Koch, "Design Concepts for Liquid Injection Thrust Vector Control, Part 1 - System Considerations," Hydraulics and Pneumatics, September 1965.

12. Personal Communication, Burgunder, A. T., Fluid Systems Division, Allied-Signal Aerospace Company, Tempe, AZ, August 1989.

13. Nogues, P., and Mazond, M., "Values Asservies Pour le Pilotage d'un Engin Par Injection Secondaire Liquide".

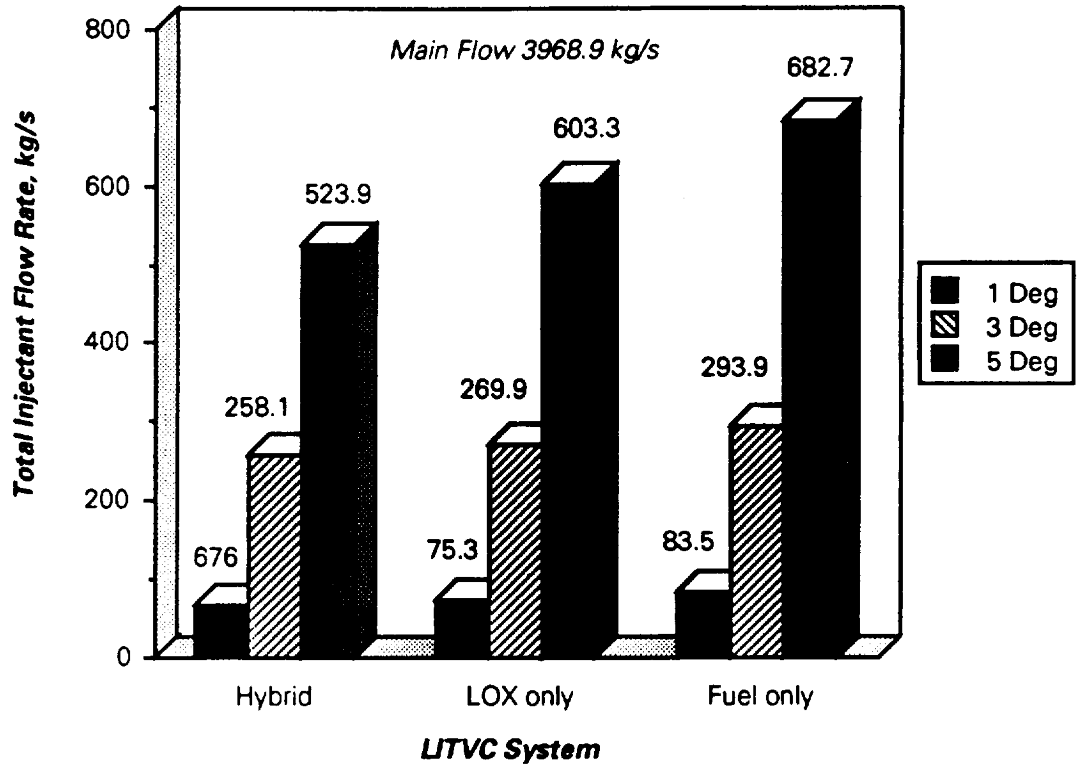

Figure 28. Comparison of injectants with three maximum nozzle deflections.

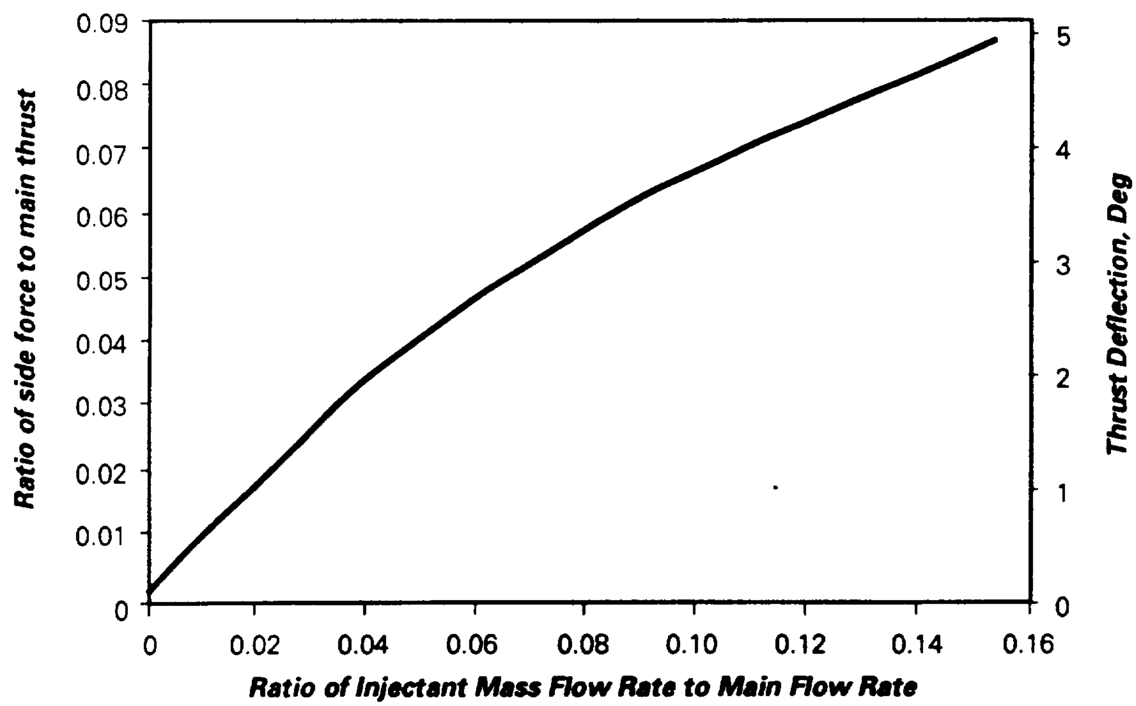

Figure 29. Hybrid engine FITVC performance (liquid oxygen injectant).

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side-force-to-main thrust versus injector mass flow rate-to-total flow for the LOX injectant. This relationship between the thrust deflection angle and force ratio illustrates the increasing inefficiency of an FITVC system at larger thrust deflection angles. This characteristic impacts the injectant system design in two ways: (1) at large deflection angles, the flow rate may become too large to have only a single port at each injector location; and (2) when the deflection angle is 5 degrees, the injectant flow approaches 16 percent of the LOX requirement for the main propulsion system, which is impractical for TVC application. Thus, a practical deflection angle using LOX for the hybrid application is 3 degrees.

The injectant trade study is shown in Table 14. The fuel-only and hybrid approaches require hot-gas control valves resulting in additional complexity, cost, and design risk. In addition, the hybrid system would require a second control valve for LOX and a means of monitoring the fuel/LOX mixture ratio, both of which impact reliability.

LOX appears to be the best choice for an injectant because it offers a system that is simple and low cost, with performance (injectant usage) close to the hybrid FITVC approach.

The major design implementation decisions necessary in defining an FITVC point design for this study are summarized in Table 15. Each of these decisions must be reviewed as the system requirements become better defined.

Table 14. FITVC Injectant Trade Study Summary.

System	Hybrid	LOX	Fuel
Injectant Usage (Relative to Hybrid)	1.0	1.05 to 1.15	1.14 to 1.30
Dry Weight	High	Low	Medium
Reliability	Low	High	Medium
Cost	High	Low	Medium

Conclusions

1. Fuel-only system can't win.
2. Lower flow rate of hybrid probably doesn't offset LOX-only system advantages for reliability and cost.
3. Focus on LOX-only system design.

Table 15. Conceptual Design Implementation Decision Summary.

Item	Design Feature	Decision	Rationale
1	Maximum Thrust Deflection	3° Max	FITVC system injectant usage efficiency reduced significantly past 3 degrees.
2	Number of Injectors	8	Injectant usage reduction (i.e., 4-injector system uses 1.31 times more fluid than 8-injector system).
3	Number of Feedlines	4	Weight, redundancy.
4	Injector Flow	355 kg/sec, 6.88 MPa	Worst case flow requirements for 3-degree system at maximum engine flow conditions.
5	Feedline Flow Requirement	390 kg/sec	Worst case flow requirements for injectors on a common requirement.
6	Feedline Sizing	12.7 cm ID	Minimize head loss.
7	Injector Metering Configuration	Shaped Nozzle	Pintle loads are more linear, collimator implementation.
8	Single vs. Multiple Injection	Single Port	Simple, more reliable approach.
9	Injector Location	Area Ratio = 0.33	Empirical data nominal location, additional analysis required to optimize.
10	Injector Angle	90°	Empirical data nominal location, additional analysis required to optimize.
11	On-Off vs. Proportional	Proportional	Less severe pintle impact design considerations.
12	Control Feedback	Pintle	Easily implemented without LVDT or other position sensor using "Follower Servo" approach.

Figure 30 is a schematic representation of the FITVC system. The system consists of eight independently controlled injectors supplied through four feedlines by the fuel injection manifold. Each of the injectors is supplied with a constant source of LOX at a controlled pressure by the four (primary) turbopumps. It was assumed that the primary turbopumps have the capability of supplying the required LOX to each of the injectors at design flow conditions.

The FITVC conceptual design sizing and performance summary is shown in Table 16. A sketch of the conceptual FITVC integrated with the nozzle is given in Figure 15. Key features of the design include:

- LOX is used as the servo actuator working fluid.
- Simple one-piece pintle/actuator piston design.
- Stepper motor-controlled servo-actuator produces 150:1 force amplification.
- Integrated pintle/slide valve design reduces package size/weight.
- Head loss minimized with toroid feed manifold and collimators.
- Self-housed injector assembly easily integrated with nozzle.

A weight estimate for the FITVC system is shown in Table 17. The weight estimates assume an average density of 2.8 gms/cm^3 (0.10 lbm/in^3) (aluminum bronze) for the injector assembly. The feed line weight estimate assumes a 7.8 gms/cm^3 (0.28 lbm/in^3) density. Electrical power requirements are estimated to be 40 watts per injector, and the accuracy of the system is estimated at 0.1 degrees. The LOX feedlines will be taken off the injector manifold. The total LOX requirement for the baseline duty cycle of 150 degrees-seconds is about 12,143 kilograms (26,770 pounds).

The injectors were designed using a pintle valve controlled by a slide valve "follower servo" approach. The large pintle flow forces typically experienced in an FITVC system dictate the use of a servo mechanism to actuate the injector pintles. Traditionally, hydraulic actuators have been used in these applications because of their inherent high-force/low-electrical power capability. In this application, high-pressure LOX can be used as the actuator working fluid because of its availability. This approach simplifies the injector design and eliminates the need for a separate actuator power source.

Several servo design approaches were examined for feasibility using LOX. An "open center" valve design was considered for simplicity (the "open

Table 16. FITVC System Sizing/Performance Summary.

* Injectant	Liquid Oxygen
* Maximum Thrust Deflection (deg)	3 at maximum thrust
* Mass Flow Capability (kg/sec) (at 3°)	355 (per injector) 391 (system)
* Pintle/Seat Design	
- Seat Diameter (cm)	9.7
- Maximum Stroke (cm)	2.8
- Piston Diameter (cm)	13.7
- Pintle Loads (N)	22,686 (maximum)
* Nozzle Design	
- Seat Angle (deg)	60 (includes angle)
- Nozzle Diameter (cm)	7.9
* Collimator Design	
- Number of Holes	12
- Hole Diameter (cm)	2.0
* Slide Valve Design	
- Diameter (cm)	2.54
- Stroke (cm)	± 0.06
- Valve Load (N)	± 44.5
* Stepper Motor Design	
- Step Size (deg)	15
- Acme Screw Lead (cm/rev)	0.5
- Stepping Speed (steps/sec)	216
- Stepping Torque (Nm)	0.11
- Electrical Power (w/injector)	40
* Feedline Diameter (ID) (cm)	12.7
* Injector Axial Location	Nozzle Area Ratio = 0.33
* System Performance	
- Actuator Force Output (N)	± 38,788 @ 5.5 MPa supply
- Frequency Response (Hz)	4
- Slew Rate (deg/sec)	5
- Accuracy (deg)	0.1

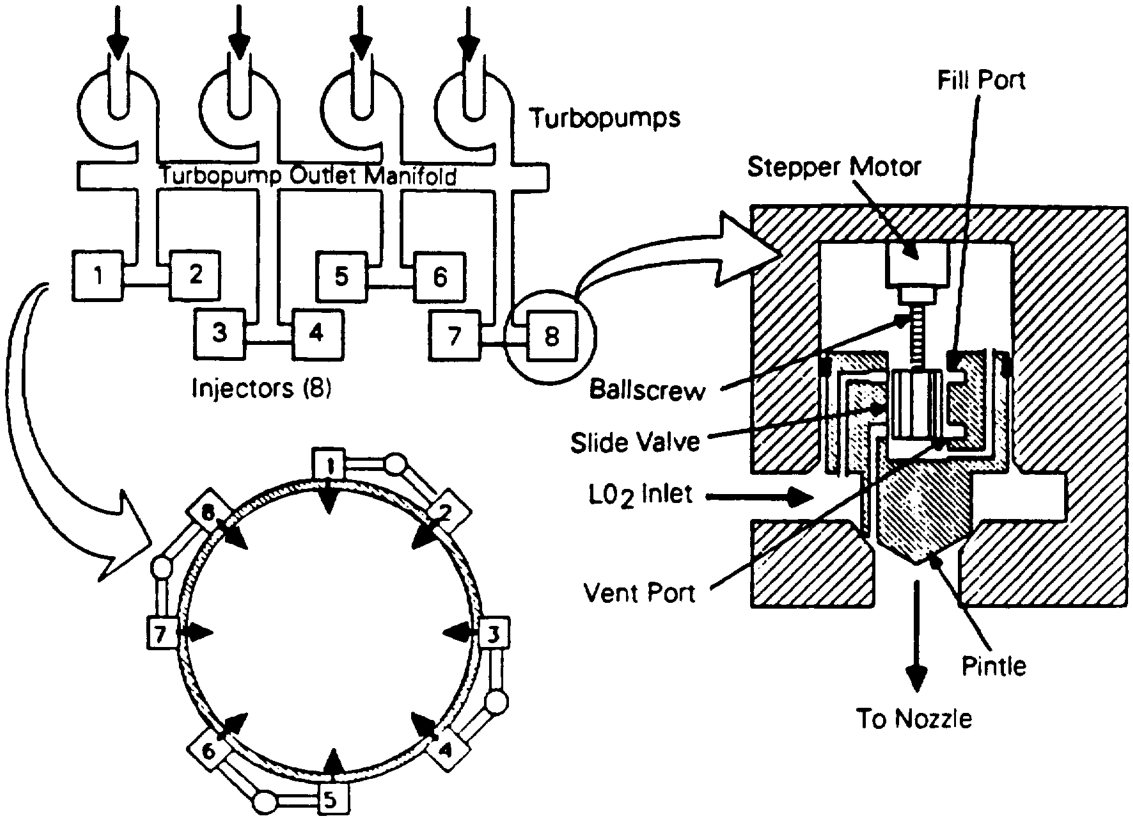

Figure 30. Hybrid booster program FITVC schematic.

Table 17. FITVC Weight Estimate Summary.

Item	Weight (kg)	Quantity	Total (kg)
Feedlines/Mounting Flanges	130	4	519
Injectors	41	8	330
Misc. Mounting Hardware	43	-	43
System Total			892

center" approach requires no seals in the valve porting area). The quiescent valve leakage could be used to cool the pintle during closed conditions. A stepper-motor-driven "follower servo" using a linear slide valve was finally selected for the conceptual point design because of the low electrical power usage. This approach can be implemented without the use of a pintle position feedback transducer, further simplifying the design.

2.4.8 Ancillary Components

2.4.8.1 Launch Pad Support Truss - A truss-type launch pad support was chosen to provide structural efficiency and the ability to retract from the rocket motor exit cone during launch. Because the support system can be retracted and remain on the pad, launch weight is reduced.

The truss structure was sized with a top inner diameter of 386 centimeters (152 inches), and an outer diameter at the bottom of 635 centimeters (250 inches). This provides ample clearance for the exit cone (see Figure 15, Page 33). Each strut is 254 centimeters (100 inches) in length, angled 45 degrees relative to the motor centerline axis, and 60 degrees relative to adjacent struts. The strut support was sized to withstand 13.3×10^6 Newtons of thrust with a safety factor of 1.6 or 27,216 kilograms (60,000 pounds) on each strut in the structure. The analysis assumed each strut was made of 1,514-MPa (220-Ksi) D6AC steel or an equivalent-strength steel or composite. The truss required 3.8 centimeters^2 of material cross-sectional area to withstand the required load. The Euler buckling equation was solved for the strut radius to ensure buckling did not occur. This resulted in a minimum strut radius of 7.8 centimeters (3.08 inches). A check on mode buckling showed that a 15.2 centimeter (6 inch) diameter strut with a 0.2-centimeter (0.08-inch) wall was acceptable.

2.4.8.2 Aft Skirt and Thrust Transfer Ring - The aft skirt and thrust transfer ring must withstand the truss load from the launch pad support truss and transfer this load into the composite wall of the booster. The ring must also withstand the out-of-plane loads introduced by the attachment of each booster to the core vehicle. The booster-to-core vehicle attachment must transmit axial, radial, and circumferential loads.

The ring is designed as a fitting fabricated from D6AC steel, and is bulky to accommodate the stress concentrations associated with the attachment of struts and due to the geometry needed to make this attachment with sockets and pins. Each individual strut applies 27,216 kilograms (60,000 pounds) of force in the vertical direction in line with the strut. A 2.54-centimeter (1-inch) pin is required for this application. It provides a calculated safety margin of 3.14 with a 1,101 MPa (160 Ksi) pin strength.

The I-beam-shaped portion of the ring is sized to withstand the booster-to-core attachment loads. These are based on ASRM and are 98.7 Newtons radial at 8.8×10^6 Newtons of axial load, and 26.7×10^4 Newtons circumferential load relative to the booster centerline. The I beam was sized using the 98.7 Newtons radial load which is the dominant load on the interface ring. The calculated bending moment for this load, using Roark's formulation, is 451,600 Newton-meters.

For an I beam with a 20.3-centimeter (8-inch) depth, the required moment of inertia for the section is 184 centimeters (72.6 inches). The I beam portion of the ring fitting will have a flange thickness of 1.1 centimeters (0.45 inches) and a web thickness of 0.8 centimeters (0.3 inches). The weight of this portion will be 46.2 kilograms per circumferential meter. The weight of the fitting portion of this ring (struts from pad and core vehicle) must be added to this weight.

2.4.8.3 Recovery System - The option of recovering some or all of the booster components is motivated to reduce LCC by reusing high-cost, refurbishable components. Based on examining component costs, the items on a hybrid booster worth recovering include turbopumps, the regeneratively cooled thrust chamber, and heavywall metal tanks.

To investigate recovery options, a saltwater landing was selected with a terminal impact velocity of 12.2 meters/second (40 feet/second). Impact loads, floatation system(s), and saltwater contamination were all included in trade decisions; reliability was not evaluated. After analysis of all potential options for recovery of pump and pressure-fed concepts, a set of options was assembled, Table 18.

Recovery of a composite tank was judged to be unacceptable. Even if the laminate was strengthened sufficiently to take the impact loads, the effort involved with inspecting/refurbishing the tank for delamination and/or water absorption would probably exceed the cost of a new tank.

Table 18. Recovery Options.

	Pressure-Fed		Pump-Fed	
	Composite	Metallic	Composite	Metallic
Full		X		
Partial			X (Recover Engine Component Only)	X (Recover Engine Component and Gas Generator)
None	X	X	X	X

The only item of value in the pressure-fed system worth recovery would be the thickwalled metal oxidizer tank, if it was to be used in the design. The cost advantages of this approach can be modeled using the shuttle SRB cases.

The pump-fed systems use low-pressure oxidizer tanks. In the absence of large pressure loads, these tanks tend to be lightweight and not capable of the sustaining impact loads. The high-value components (turbopumps, regeneratively cooled thrust chamber) would be worth recovering. Our design philosophy was to physically group these components together for recovery and discard the rest.

Several recovery concepts were explored. A recovery technique for reuse of the grouped high-value items, illustrated in Figure 31, provides a method of keeping the reusable components dry, and avoids complex valves, bladders, and seals associated with some concepts. The booster is slowed by a series of parachutes housed in the nose cone after hybrid burnout. Risers, which are structurally tied to the aft end of the booster, reorient the booster to impact in a nose-down attitude. Solid retrorockets in the aft skirt are fired to slow the impact velocity to less than 12.2 meters/second (40 feet/second). Ports are opened in the oxidizer tank; these are designed to rapidly flood the tank. The resultant center-of-gravity/center-of-buoyancy locations yield a stable floating configuration with the aft end well-above the water line. The recovery ship would then either tow the entire vehicle or lift the aft end while the tank is separated and sunk. Recovery system weights for the different options are shown in Table 19.

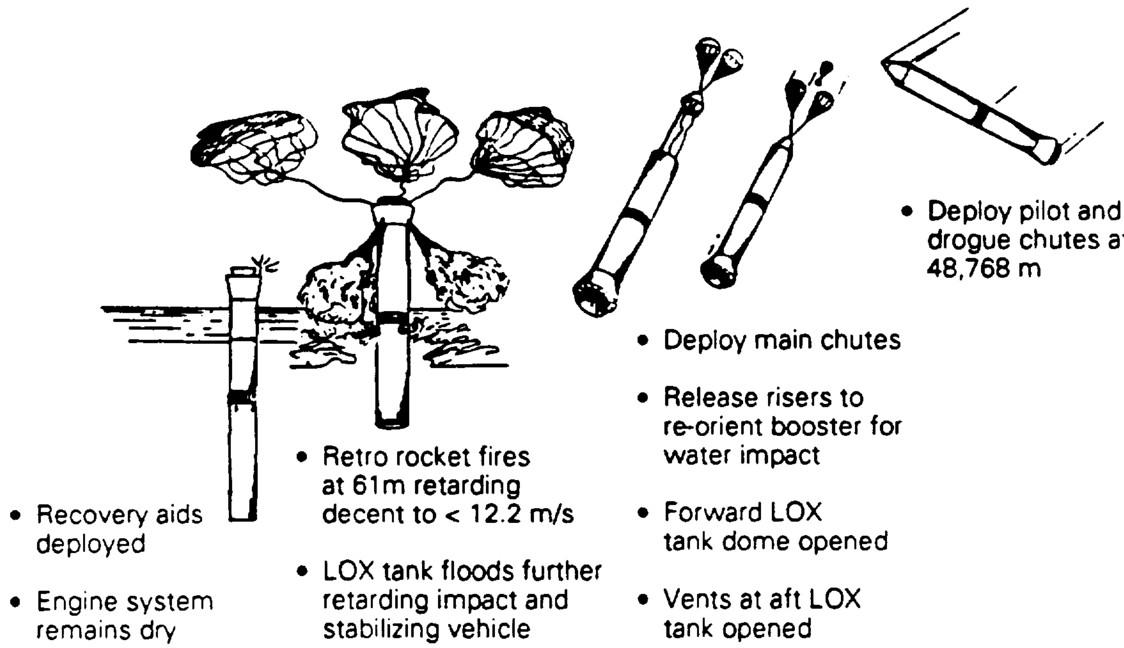

Figure 31. *Recovery scenario.*

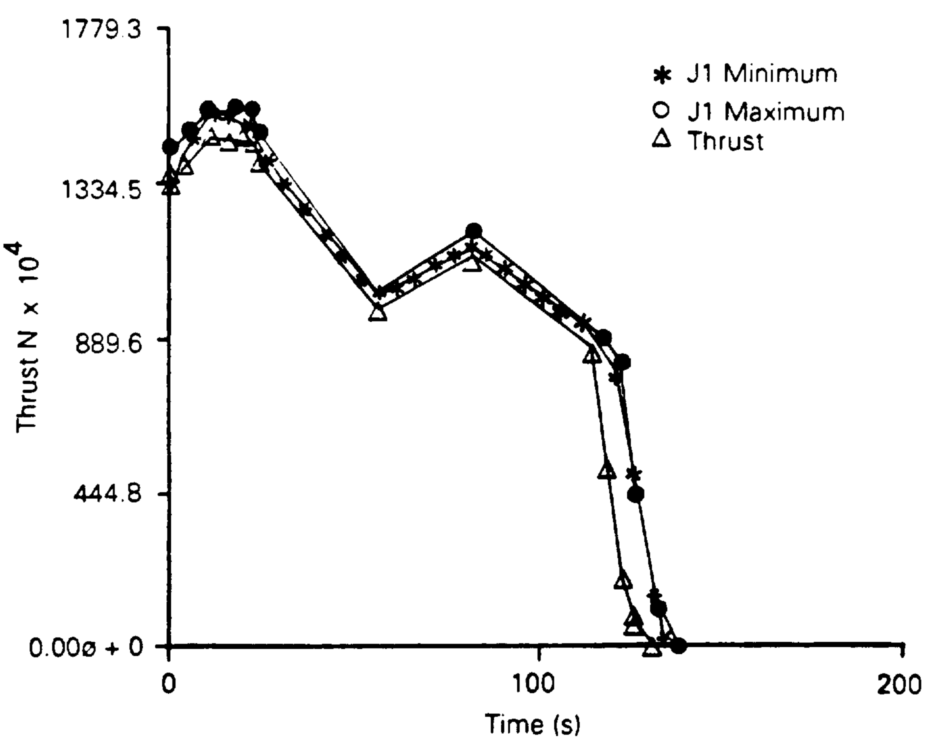

Figure 32. *Predicted thrust trace full-size booster.*

Table 19. Recovery Systems Weights.

Item	Metallic Tank Pressure-Fed	Composite Tank Pump-Fed	Metallic Tank Pump-Fed
Pilot chute (kg)	13.6	13.6	13.6
Drogue Chutes (kg)	997.9	263.1	385.6
Main Chutes (kg)	3991.6	1043.3	1814.4
Retrorockets (kg)	861.8	226.8	385.6

2.4.9 Performance Predictions

Each full-size booster was designed to provide 15.1×10^6 Newtons of thrust over the 120-second burn time. Performance of the propulsion system was predicted using the TRANSV computer model developed by ARC for solid propellants.[14] The model was modified for the hybrid to simulate: (1) burning rate sensitivity to chamber pressure; (2) instantaneous burning surface area; (3) LOX flow rate interaction; and (4) pressure drop across the injector. Thrust calculated by the model is given in Figure 32 and compared with the minimum and maximum values provided by the SOW (3 percent variation). Model predictions for chamber pressure, mixture ratio, and I_{sp} are shown in Figures 33 through 35. The maximum expected operating pressure, MEOP, was determined by adding a 3 percent manufacturing variation to the prediction calculated at 306K (551°R). This variation is dominated by burning rate associated with fuel batch-to-batch processing; it is also comprised of variations associated with grain dimension, throat area, and fuel properties. The mixture ratio and theoretical I_{sp} are maintained at near-optimal values throughout the flight using a combination of grain geometry tailoring and throttling of LOX flow rate.

The gas generator grain is a center-perforated configuration with a 78.7-centimeter bore (31 inches) and eight 10.2-centimeter-wide slots equally spaced around the circumference. The slots are overcast with a

14. TRANSV: Transient Internal Ballistic Prediction program; ARC developed; provides pressure, mass flow rate, thrust predictions in three calculation phases, ignition, steady state combustion, tailoff.

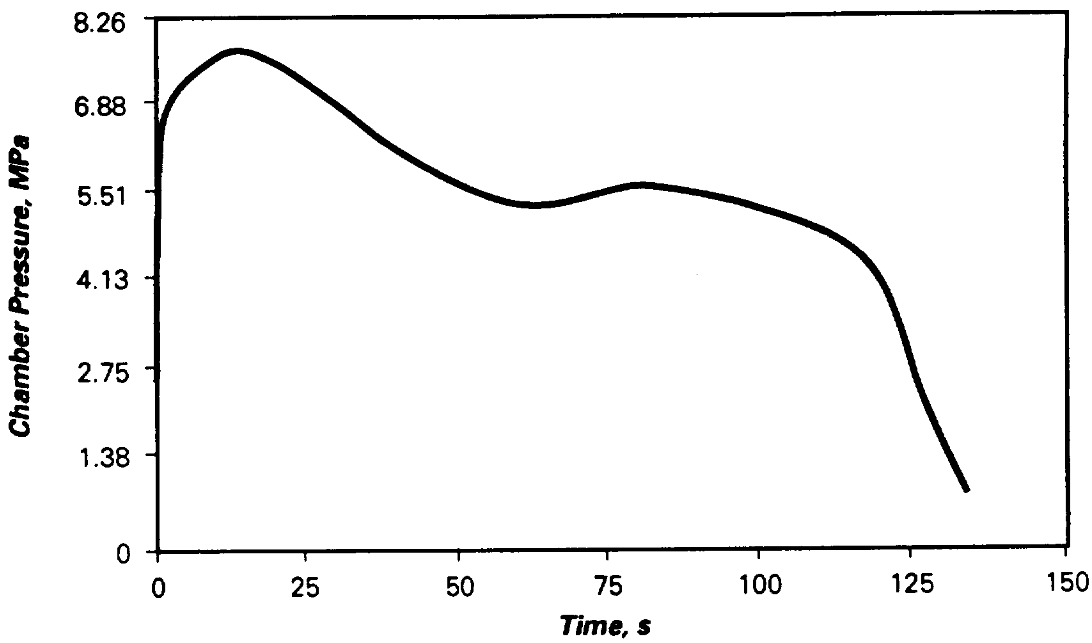

Figure 33. The predicted chamber pressure for the full-size booster grain design.

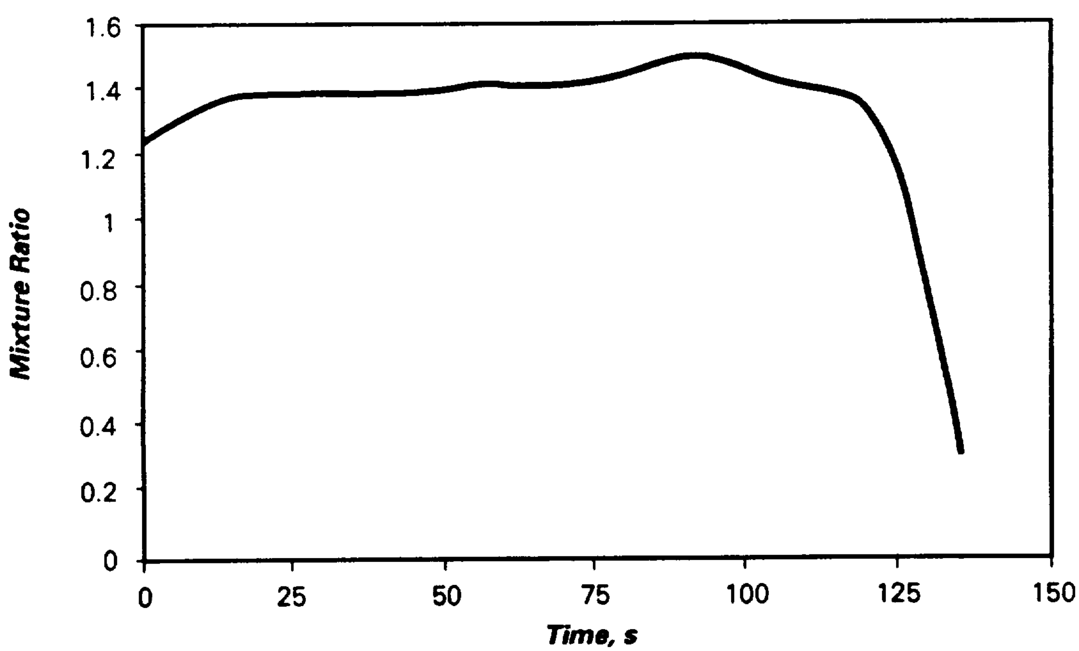

Figure 34. The predicted mixture ratio for the full-size booster grain design.

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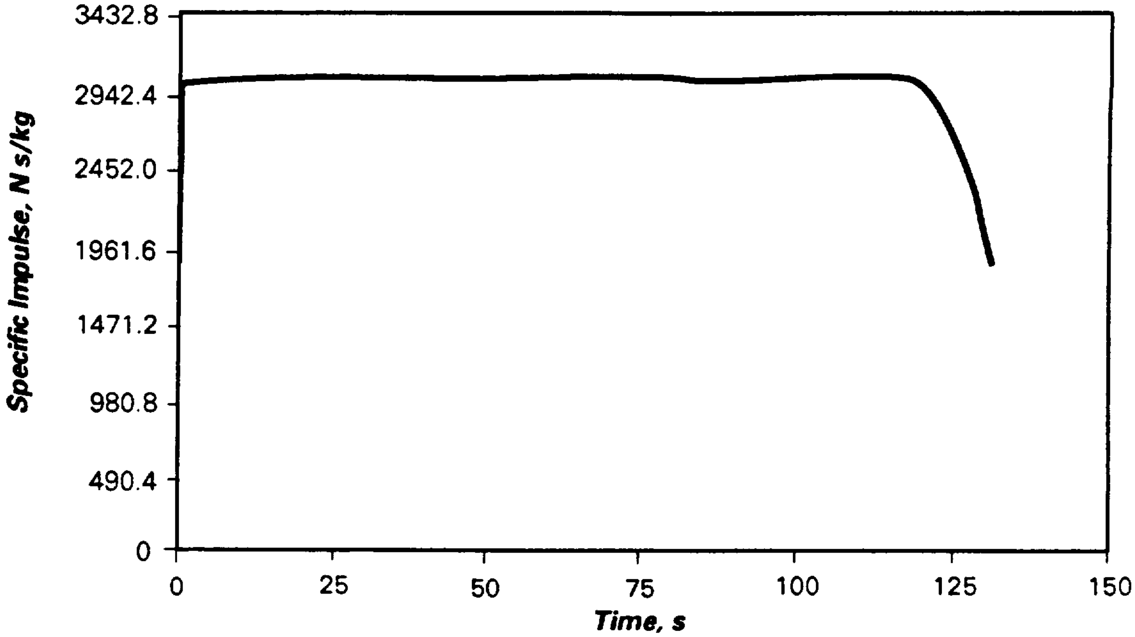

Figure 35. The predicted vacuum specific impulse for the full-size booster grain design.

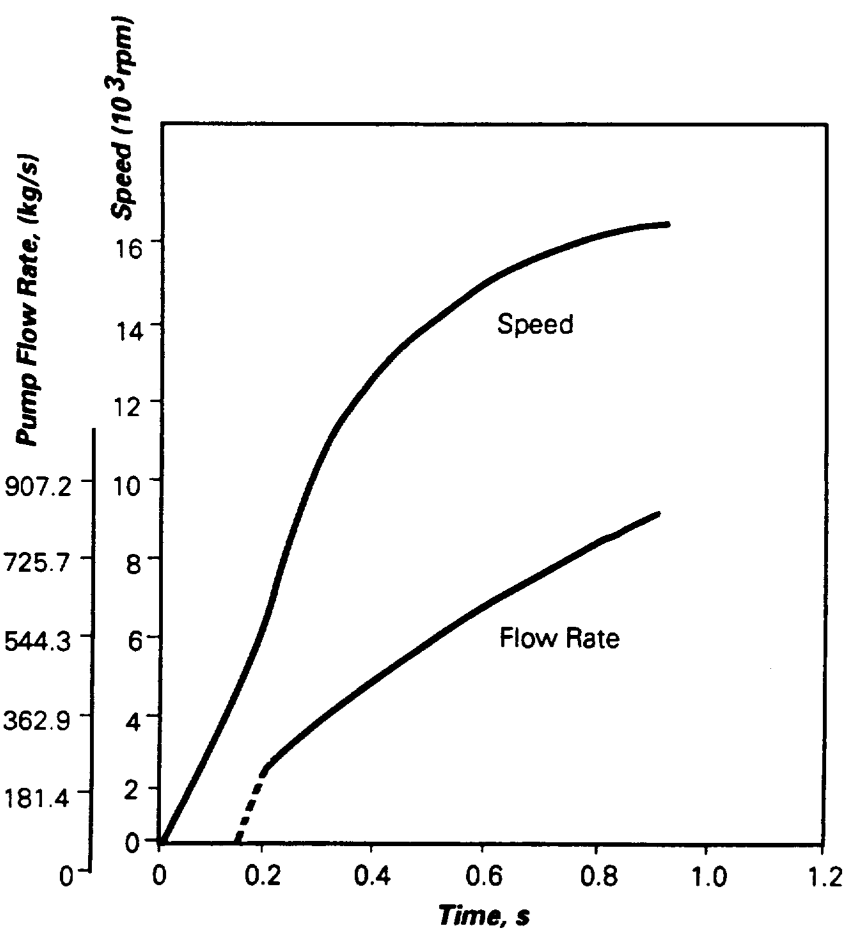

Figure 36. Turbopump start-up characteristics.

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2.5-centimeter-thick web of igniter propellant with a burning rate of 2.5 centimeters/second. To ensure that LOX flow could be established quickly to stabilize combustion of the gas generator fuel, ARC used the combustion model to examine the ignition and start-up transients. LOX flow rate at start-up is shown in Figure 36. The curve was calculated using the turbine inlet pressure, pump speed, and head pressure at the injector. This predicted LOX flow rate was input to the TRANSV model to predict the start-up thrust and pressure given in Figures 37 and 38. Steady-state thrust and pressure are established in approximately 1 second. The chamber pressure exceeds the 2.1 MPa (300 psia) extinguishment limit of the gas generator less than 0.3 seconds after ignition.

During normal operation, gaseous fuel flow through the injector is subsonic, allowing pressure changes in the thrust to be transmitted to the gas generator. Further, the gas generator pressure level is only slightly higher than the thrust-chamber pressure. When LOX flow is terminated, pressure in both the gas generator and thrust chamber decreases. The predicted gas generator pressure with and without LOX flow (assuming the gas generator would still burn without oxidizer flow at pressures below the extinguishment pressure) is shown in Figure 39. Proper sizing of the cumulative fuel injector port flow area will result in a gas generator pressure that is below the extinguishment limit of the fuel. Thus, the fuel ceases to burn without oxidizer flow. Fuel extinguishment was demonstrated under corporate IR&D funding. Figure 40 shows the burning rate of the ARCADENE 399 formulation tested as a function of pressure. This formulation was not tailored to meet the hybrid requirements but extinguished below 3.4 MPa (500 psi).

Emergency shutdown of the booster was simulated at a number of points in the flight using the hybrid computer model. The termination of LOX flow was assumed for this analysis to be instantaneous (turbopump spool-down was not considered because it could not be quantified for our design). In every instance, this termination resulted in the immediate and total termination of thrust and gas generator combustion. The results from one of these shutdown simulations are provided in Figures 41 and 42. While the termination of gas generator combustion is not a requirement under this program, it was addressed to meet the pad-abort requirements: the booster will automatically shut down on the pad if LOX flow rate levels are not established by the time the start-up grain is exhausted.

FOLDOUT FRAME

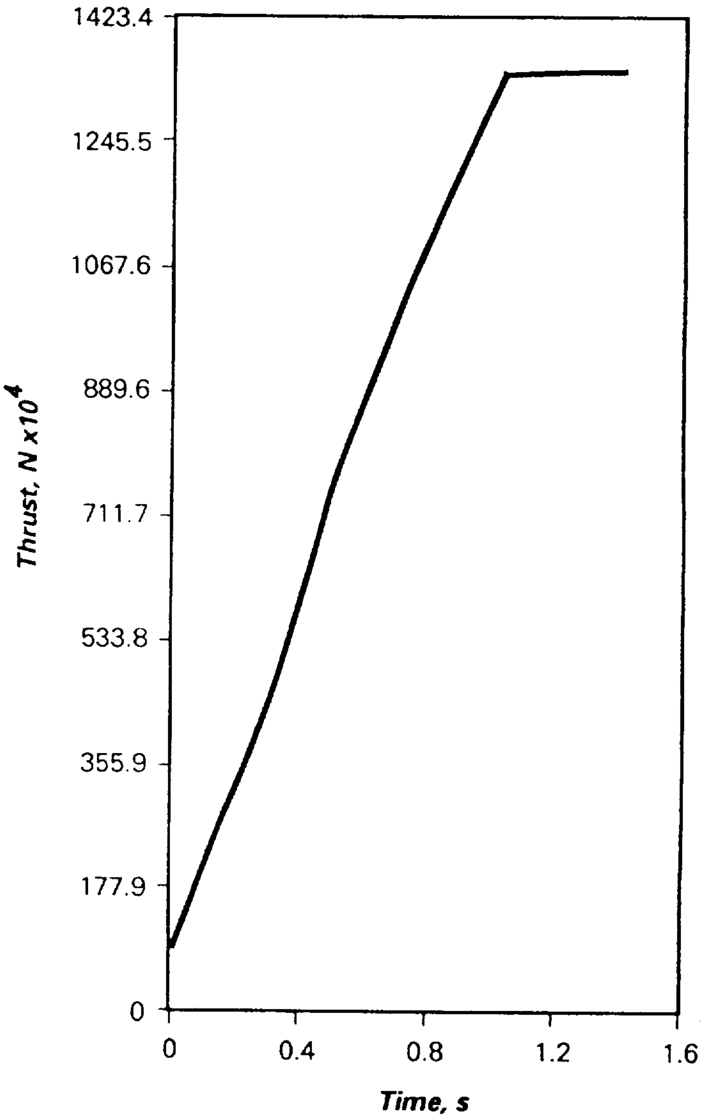

Figure 37. Predicted start-up transients full-size booster.

FOLDOUT FRAME 2

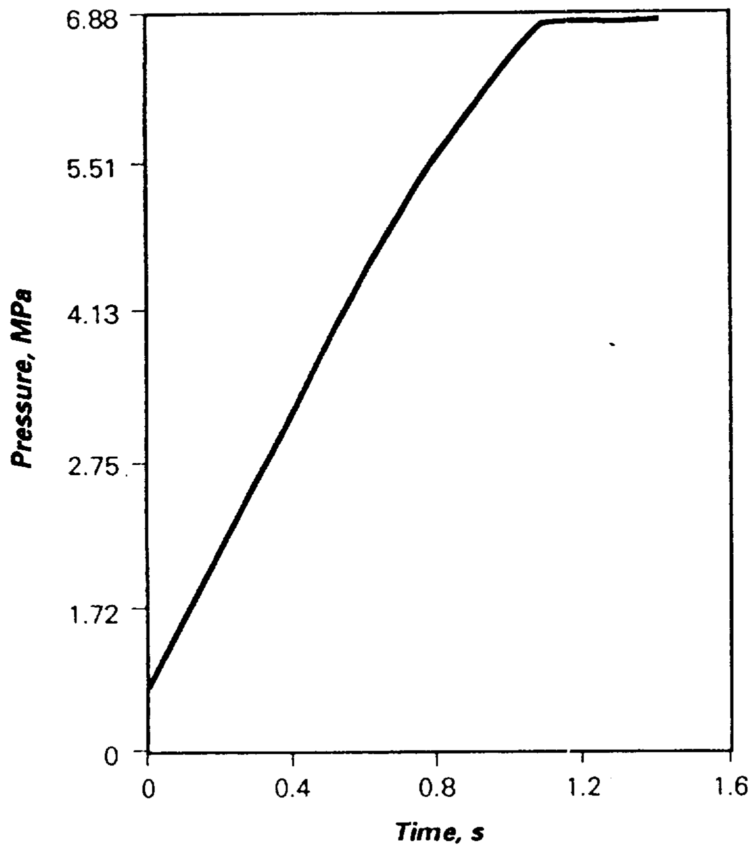

Figure 38. Predicted start-up pressure transient full-size booster.

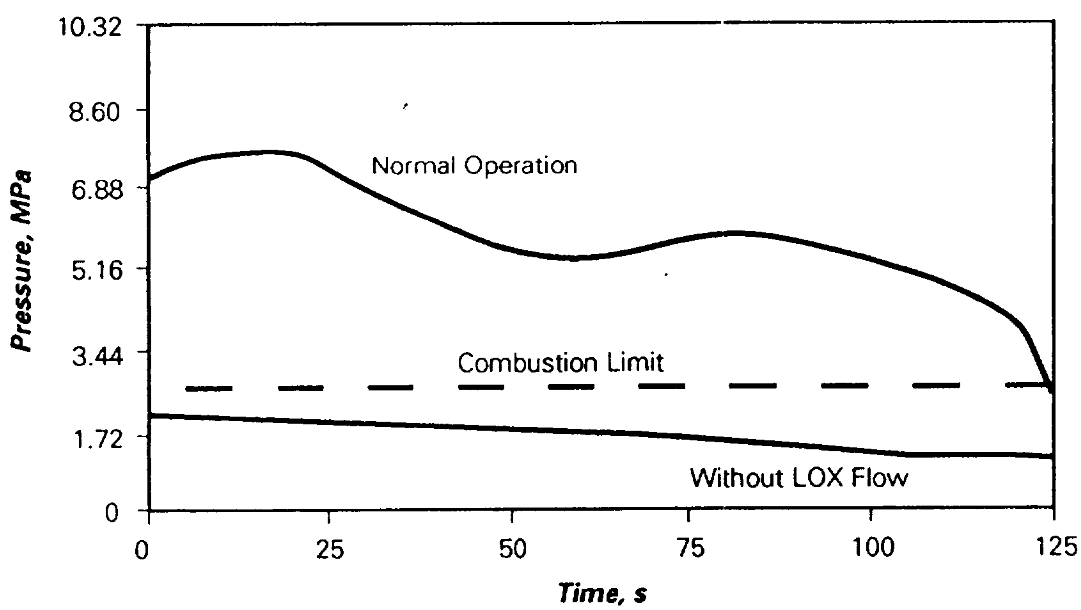

Figure 39. Gas generator pressure.

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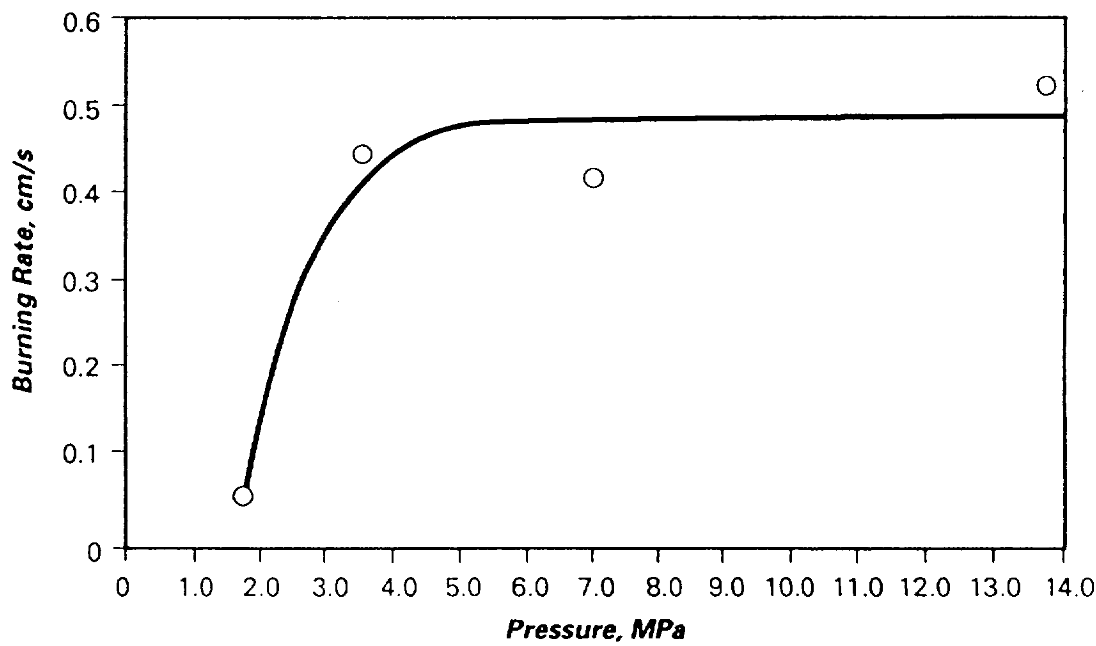

Figure 40. Burning rate vs pressure.

FOLDOUT FRAME 2

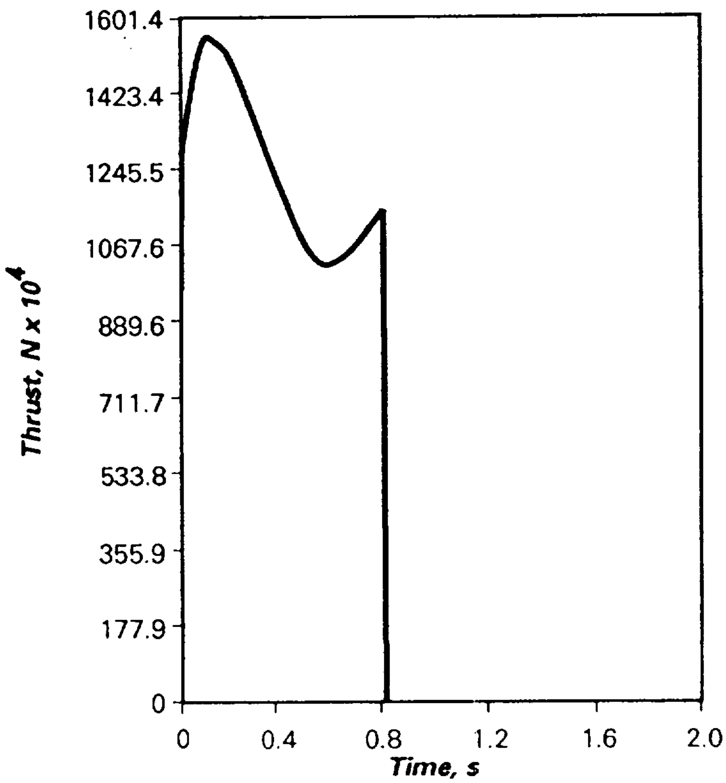

Figure 41. Emergency shutdown thrust.

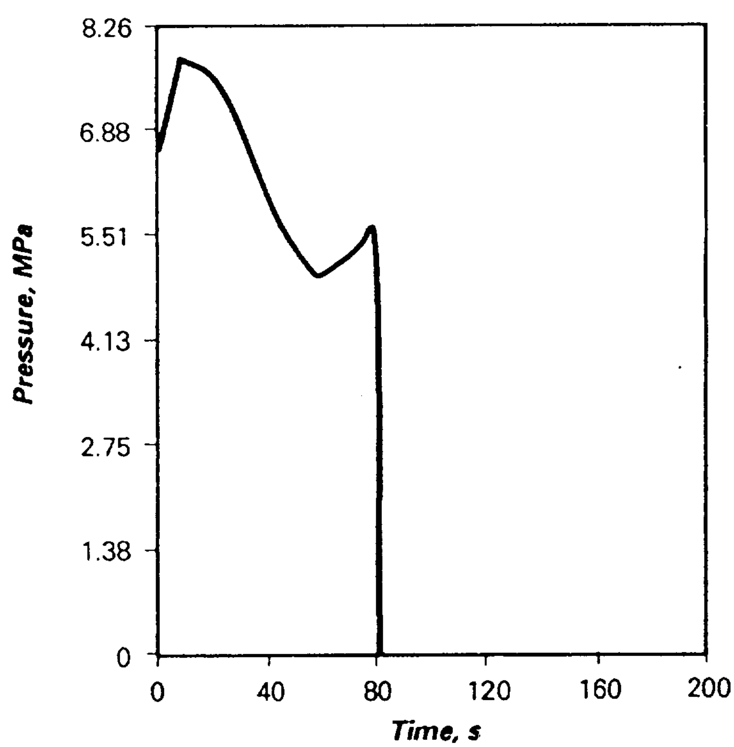

Figure 42. Emergency shutdown pressure.

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To conclude the full-size evaluation, Boeing "flew" the turbopump-fed gas generator hybrid. The hybrid booster was nominally 414 centimeters in diameter and 4,681 centimeters in length, with a gross lift-off weight of 564,859 kilograms. The booster had a carbon/epoxy (IM-7/EPON 826) gas generator case, aluminum-lithium LOX tank, silica phenolic, monolithic-braided ablative nozzle with fluid injection thrust vector control (LOX injectant). The booster was "flown" to their separation point and the shuttle and external tank were "flown" to low earth orbit (150 nautical miles at 28°E).

2.5 Quarter Size Point Design

Besides the full-size point design developed above, ARC was required to develop a point design for a booster having one-fourth the thrust of the full-size booster. In this configuration, eight boosters are mounted around a core vehicle. To provide a comparison between the quarter- and full-size booster designs, chamber pressure, design mixture ratio, and nozzle expansion ratio were held constant between the two sizes. This approach simplified the design effort since many of the major design parameters remained unchanged while others scaled directly. Many features of the full-size booster were retained for the quarter-size design. The differences between the two designs are: (1) the core vehicle supports its own weight and the weight of the eight hybrid boosters on the pad; (2) the launch pad support truss is not required; (3) bending stiffness requirements for the booster case are relieved since there is no launch "twang"; (4) a single diameter was selected for the entire booster; (5) only one turbopump is used per booster (no redundancy); and (6) the propellant burning rate and pressure exponent were reduced to compensate for the reduced grain web of the smaller booster.

Layout drawings for both the pressure-fed and turbopump options are given in Figure 43. A list of the component weights for the pressure-fed and turbopump options is provided in Table 20.

2.5.1 Gas Generator

The gas generator fuel formulation was identical to that used in the full-size booster, but the burning rate was reduced to 0.81 centimeters/second (0.32 inches/second) at 6.88 MPa (1,000 psia) from 1.27 centimeters/second (0.50 inches/second). This is accomplished by tailoring the fuel formulation such as changing oxidizer particle size, decreasing burning rate catalysts, or

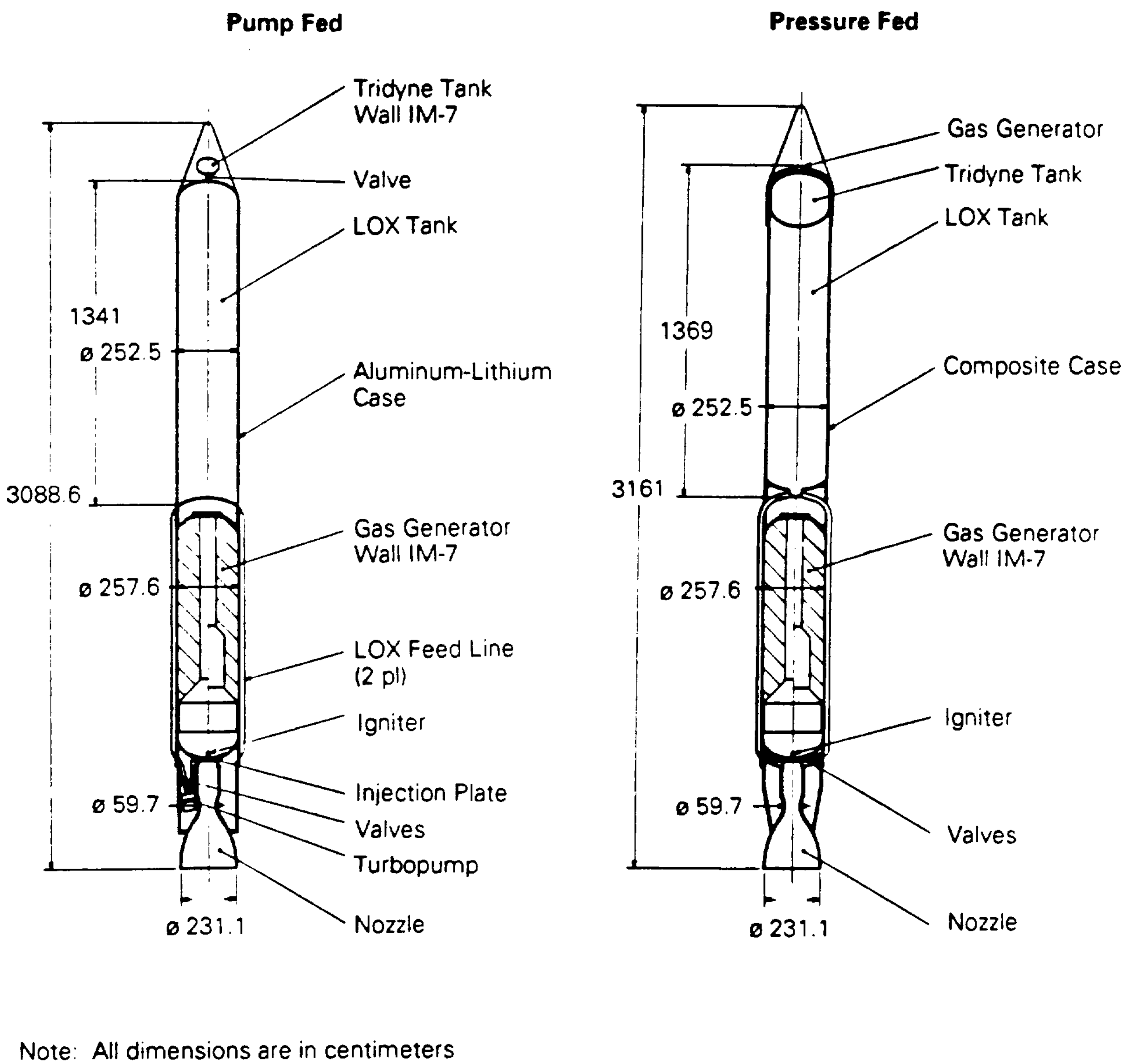

Figure 43. Quarter size booster designs.

Table 20. Quarter-Size Vehicle Weight Breakdown (Pressure-Fed).

Subsystem	Element	Pressure-Fed Weight (kg)	Pump-Fed Weight (kg)
Gas Generator	Propellant	52,478	53,725
	Case	1,227	1,247
	Liner/Insulation	249	252
	Igniter	11	11
Oxidizer Delivery System	LOX	74,925	74,925
	Tank	2,946[1]	943[2]
	Feedlines	73	24
Pressurizing System	Tridyne	891	278
	Tank	2,136	
	Liner	41	
	Catalyst Bed	75	980
	Plumbing and Valving	34	
Thrust Chamber	Injector Manifold	204	204
	Chamber	1,877	1,458
Ancillary Components	TVC	249	249
	External Insulation	1,004	1,004
	Interstage	304	113
	Nose Cone	298	298
	Skirt	726	726
Total Weight		139,748 (308,091 lbs)	36,437 (300,792 lbs)

1. IM-7/EPON 826 carbon-epoxy tank
2. Aluminum-lithium tank.

using burning rate suppressants. A 259 centimeter (102 inch) diameter grain was selected to maintain similar length-to-diameter ratio between the two vehicle sizes. An impulse efficiency of 92.5 percent and a sliver fraction of 2 percent were assumed in the design. The grain design was tailored by ballistic analysis to achieve the required thrust throughout the flight while maintaining an optimum mixture ratio. The predicted thrust and mixture ratio are given in Figures 44 and 45, respectively. The resulting grain geometry is similar to the full-size grain design; the center port diameter is 68.3 centimeters (26.9 inches), and the grain length is 889 centimeters (350 inches). Figure 46 shows the geometry of each of the two sets of aft slots. The total gas generator propellant weight for the pressure-fed option is 52,478 kilograms (115,694 pounds). An additional 1,225 kilograms (2,700 pounds) is required to drive the turbopumps for the pump-fed option. Starter propellant grain segments may be overcast, or a separate cartridge may be used. The gas generator igniter weighs 11.3 kilograms (25 pounds) and is scaled down from the full-size igniter.

The gas generator case was designed for an MEOP of 8.6 MPa (1,253 psia) and a safety factor of 1.6. The filament-wound composite case thickness used IM-7 carbon fiber and epoxy resin was 1.1 centimeters (0.45 inches) [(weight of 1,227 kilograms (2,704 pounds)]. The case is insulated with ARCTIP (HTPB filled with glass microballoons). Insulation requirements were the same as the full-size booster since the environment and burn time are the same.

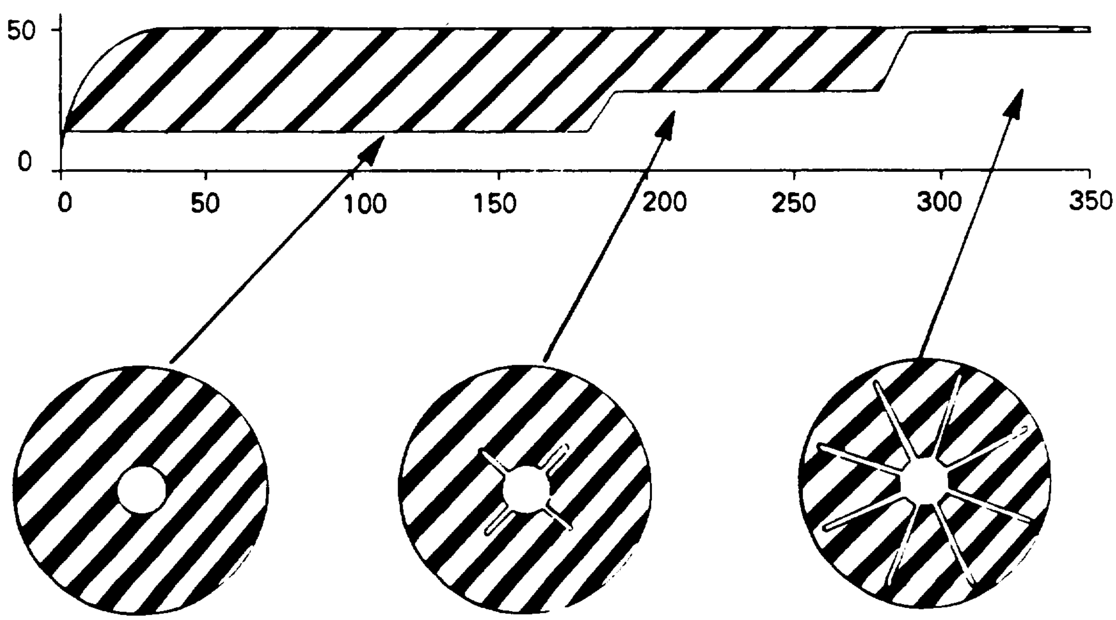

Figure 46. Quarter size grain design.

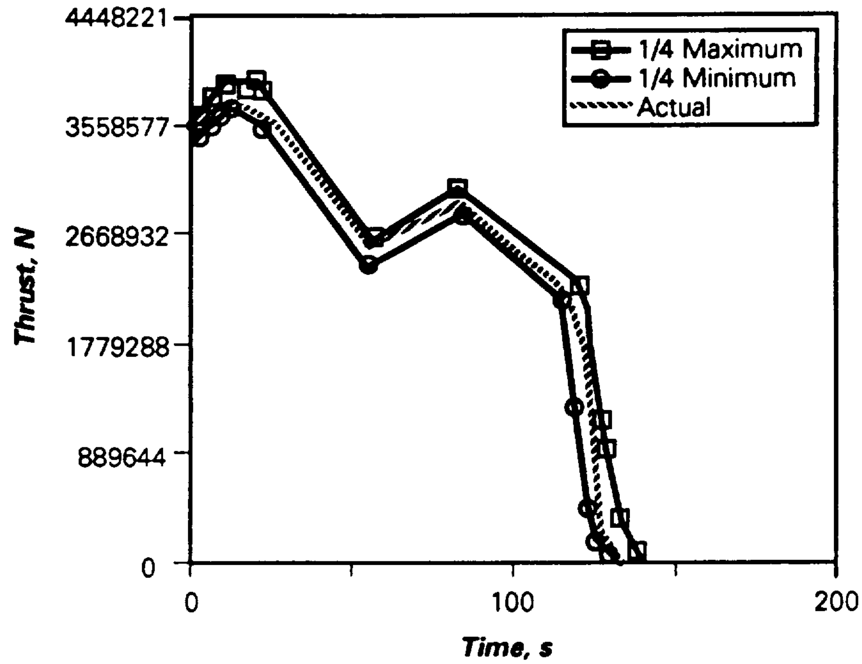

Figure 44. Quarter-size booster thrust trace.

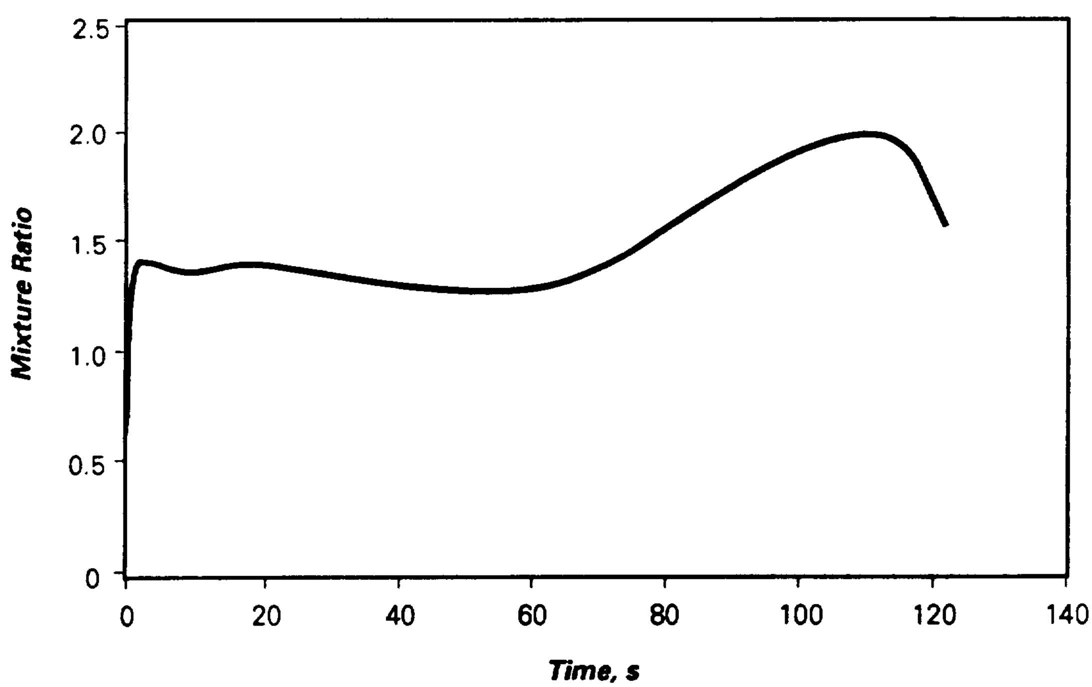

Figure 45. Quarter-size booster mixture ratio.

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2.5.2 Thrust Chamber

The quarter-size booster was considered an expendable system; for this design, therefore, only an ablative thrust chamber was considered. The general characteristics of the thrust chamber are similar to the full-size booster. Many of its major dimensions are scaled down; the throat and exit areas are one-fourth of those in the full-size booster. This results in a throat diameter of 59.7 centimeters (23.5 inches) and a nozzle exit diameter of 231 centimeters (91 inches). The physical length of the cylindrical portion of the chamber is 150 centimeters (59 inches) to maintain a design L* value of 305 centimeters (120 inches) and a residence time of 4.3 milliseconds.

The fuel injector manifold is a direct scale-down from the full-size booster with 125 fuel ports (one-quarter that of the full-size booster). Each fuel injection port has eight pairs of oxidizer ports, the same as the full-size booster. This is one of the major advantages of the gas generator approach: many of the basic features of the booster are directly scaleable from one vehicle size to another.

2.5.3 Thrust Vector Control

Fluid injection TVC is used for the quarter-size booster. LOX flow rates and total LOX consumed scale directly from the full-size design to perform the same duty cycle. Total LOX requirements are estimated to be 3,093 kilograms (6,820 pounds) including a 2 percent reserve. The total inert weight of the TVC system is 249 kilograms (550 pounds).

2.5.4 Oxidizer Tank

The oxidizer tank for the pressure-fed design is filament-wound with IM-7 carbon fiber. Two resin systems were evaluated, EPON 826 epoxy and polyimide. EPON 826 was selected for the quarter-size point design. Sizing of the tank for structural loads benefits from the absence of a major bending stiffness requirement. Tank thickness was calculated to be 1.7 centimeters (0.65 inches) which yields a tank weight of 2,946 kilograms (6,495 pounds). The tank is lined with Upilex elastomeric material.

2.5.5 LOX Delivery System

The pressure-fed LOX delivery system resembles the design of the full-size system. Tridyne is used to pressurize the oxidizer and is stored in a filament-wound composite tank fabricated from IM-7 carbon fiber. The two resin systems evaluated were EPON 826 epoxy and polyimide. EPON 826 was selected for the point design. The tank weighs 2,177 kilograms (4,800 pounds) and includes a 0.008 centimeter (0.003 inch) aluminum liner. The feedlines for the pressurizing gas are 2.5 centimeter diameter stainless steel. LOX is fed through two 11.8 centimeter (4.65 inch) diameter lines.

2.6 Life Cycle Cost Trade Studies

2.6.1 Introduction and Summary

In order to assess the impact of component selection and design on cost, reliability and performance for the full-size and quarter-size boosters, ARC and Boeing Aerospace Company (BAC) used an integrated design model to conduct trade studies. Cost parametrics were developed for the following: (1) pump-fed oxidizer versus pressure-fed; (2) classical hybrid versus gas generator; (3) reusable versus expendable boosters; (4) oxidizer-to-fuel mixture ratio; (5) nozzle expansion ratio; (6) chamber pressure; (7) body diameter; (8) thrust deflection; (9) reserve propellant; (10) design safety factor; (11) gas generator grain radius; (12) redundant pump capability; and (13) cost-optimized design variables.

An integrated model was used to conduct the conceptual trades. BAC, under corporate IR&D, developed the model, a hypervelocity aerospace vehicle conceptual design (HAVCD) program, which uses a wide array of cost experience from launch vehicle programs, spacecraft/probes, upper stages, tactical/strategic missiles, and commercial aircraft and specialized design subroutines to perform optimization analysis. The integrated model synthesizes a booster, calculates the life cycle cost, and predicts payload performance and system reliability. The hybrid booster is synthesized from input specifications, component weights and volume algorithms. LCC is calculated using cost algorithms comprised of: (1) cost estimating relationships (CERs) to predict hardware engineering design costs and manufacturing costs; (2) support costs not related to hardware (system engineering, software test and tooling); and (3) facilities, operations and support. The system reliability is predicted based on the specified components and component failure rates. The low-cost solid propulsion study life cycle cost model, STACEM, was the source of the gas generator CERs. The CERs for liquid booster components, vehicle and launch operations nonrecurring costs, and launch operations recurring costs, were provided by BAC.

Hybrid life cycle costs were calculated for two mission models. Both mission models assumed a 4-year period of linear flight growth rate, followed by a 10-year operational period at a constant flight rate. Two constant flight rate calculations were provided: one flight per month totaling 150 missions (Mission I), and one flight per week totaling 650 missions (Mission II). Two full-size or eight quarter-size boosters were required per mission. This resulted in total production quantities of 300 or 1,300 full-size and 1,200 and 5,200 quarter-size boosters for Missions I and II, respectively. Life cycle costs were calculated in constant dollars and were not discounted.

New launch and production facilities were assumed to be required for the booster and operations nonrecurring costs. Facilities costs were included in design, development, test and evaluation (DDT&E).

A learning curve of 90 percent was assumed for all component costs. A 95 percent learning curve was assumed for propellant processing. A 100 percent learning curve was assumed for operations recurring costs. The cost of unreliability was not included in the cost calculations.

Since the point design trades and LCC analysis were performed concurrently, the hybrid booster LCC were calculated for a reference vehicle similar, but not identical, to the point design. The reference booster was synthesized by the integrated hybrid booster model and was designed to provide the specified vacuum thrust.

The point of reference for the full-size synthesized design is: mixture ratio of 1.5; initial chamber pressure of 6.88 MPa (1,000 psia); LOX tank diameter of 4.27 meters (14 feet); gas generator diameter of 3.96 meters (13 feet); and a nozzle expansion ratio of 15. The point of reference for the quarter-size synthesized design is: mixture ratio of 1.5; initial chamber pressure of 6.88 MPa (1,000 psia); LOX tank and gas generator diameter of 2.56 meters (8.4 feet); and a nozzle expansion ratio of 15.

The full-size booster includes a flexseal nozzle/TVC, carbon-epoxy (IM-7/EPON 826) LOX tank with aluminum liner, carbon-epoxy gas generator case, and an ablatively cooled PAN fiber/phenolic thrust chamber. The quarter-size booster includes the same components, but fluid injection TVC replaced the flexseal nozzle. The weights of the range safety system, booster separation system, aft skirt, and igniter and those for the expendable-versus-reusable trade study and recovery system, were assumed to be consistent with the current shuttle boosters. The reference full-size boosters utilized redundant pumps for the pump-fed designs, and both sizes utilized cold-gas helium-pressurization of the LOX tank.

A summary of the LCC estimates for both mission models and both booster sizes is shown in Table 21. LCC is broken down into four general categories: recurring vehicle costs, recurring operations costs, vehicle non-recurring costs (DDT&E), and operations non-recurring costs (DDT&E). The large booster provides lower LCC than the quarter-size booster for both mission models.

Vehicle recurring cost is the primary LCC element. The weighting of vehicle recurring and non-recurring LCC increases with the increased number of missions. A breakdown of the vehicle LCC elements is shown in Table 22. These categories are further broken down as shown below.

- Oxidizer supply includes LOX tank, pumps, pressurization, piping, and valves.
- Thrust chamber includes injector, combustion chamber, insulation and the nozzle.
- Integration assembly and checkout includes subsystem integration, subsystem assembly, and final assembly and checkout.
- Structures includes nose cap, aft skirt, and attachments.
- Solid fuel includes the gas generator propellant, gas generator case, gas generator insulation and liner.
- Electronics and instrumentation (E&I) includes avionics, batteries, instrumentation and wiring.
- Miscellaneous includes range safety system and miscellaneous booster items.

The relative weighting of the cost elements is constant for the two mission models; differences are a result of learning effects.

The major difference between the two booster sizes is in the cost of structures and E&I. The difference in the cost of structures is due to the

Table 21. Hybrid booster LCC breakdown summary.

	Large Booster		Quarter Size	
Missions	**150**	**650**	**150**	**650**
Vehicle (%)	53.0	68.8	59.1	72.5
Operations (%)	26.7	16.6	22.8	13.5
Vehicle DDT&E (%)	8.8	10.2	8.2	10.5
Operation DDT&E (%)	11.5	4.4	9.9	3.6
LCC (Billions)	$11.43	$30.21	$13.24	$37.07
LCC/Mission (M)	$76.2	$46.5	$88.3	$57.0

Table 22. Vehicle LCC breakdown summary.

	Large Booster		Quarter Size	
Missions	**150**	**650**	**150**	**650**
Oxidizer Supply (%)	22.8	22.8	21.6	21.6
Thrust Chamber (%)	18.7	18.8	19.2	19.2
Integration Assembly Checkout (%)	17.5	17.4	17.3	17.2
Structures (%)	12.1	12.1	8.3	8.4
Solid Fuel (%)	8.6	8.6	7.8	7.8
Separation Sys (%)	6.2	6.2	4.5	4.5
TVC (%)	5.1	5.1	6.0	6.0
Electronics & Instrumentation (%)	4.8	4.8	12.0	12.0
Misc. (%)	4.3	4.3	3.3	3.3

aft skirt. The full-size booster was assumed to have the same aft skirt as the shuttle SRBs. The quarter-size booster does not include an aft skirt weight allocation. The difference in E&I is due to the assumption of constant E&I requirements for the two sizes.

The breakdown of DDT&E costs is shown in Table 23. The increase in vehicle facilities, tooling, and special test equipment is due to the increased production requirements for the one-flight-per-week mission model. The costs of design and support engineering decrease with increasing missions, and are functions of the design cost of the vehicle. The costs of operations facilities and ground support equipment (GSE) are assumed to be a function of the booster stage weight. Therefore, the weighting of the operations non-recurring costs decreases with the increased number of missions. This assumption may not be valid for the one-launch-per-week mission. This model may require more than one launch site, and should be reevaluated if the one-launch-per-week mission is retained.

The breakdown of operations recurring costs is consistent for both sizes and mission models and is shown in Figure 47.

2.6.2 Conceptual Studies

Early conceptual studies were conducted to address the selection of one of the hybrid concepts and one of the oxidizers. Preliminary point designs were developed for pump-fed and pressure-fed oxidizer systems using LOX and 95-percent H_2O_2 to estimate components/weights used in the LCC model. LCC and LCC/pound of payload estimates were calculated (Figures 48 and 49) to select a single concept and oxidizer. The pump-fed gas generator hybrid with LOX as the oxidizer is shown to provide the lowest cost ($11.4 billion), and the pressure-fed classical hybrid with H_2O_2 provided the highest cost ($22.5 billion). Life cycle cost and LCC/pound of payload for the configurations is shown below.

Configuration	LCC (%)	LCC/Payload (%)
Gas Generator Hybrid, Pump-Fed, LOX	100.0	100.0
Classical Hybrid, Pump-Fed, LOX	111.4	120.8
Gas Generator Hybrid, Pump-Fed , Peroxide	117.3	132.0
Classic Hybrid, Pump-Fed, Peroxide	120.9	139.9
Gas Generator Hybrid, Pressure-Fed, LOX	133.6	152.2
Gas Generator Hybrid, Pressure-Fed, Peroxide	166.5	253
Classic Hybrid, Pressure-Fed, LOX	168.0	280
Classic Hybrid, Pressure-Fed, Peroxide	189.0	313

Table 23. DDT&E breakdown summary.

	Large Booster		Quarter Size	
Missions	**150**	**650**	**150**	**650**
Vehicle Facility Special Test Equipment (%)	28.9	62.4	37.5	70.0
Operation Facilities and Ground Support Equipment	56.7	30.2	54.6	25.2
Design (%)	9.1	2.7	3.0	1.3
Support (%)	5.3	4.7	4.9	3.5

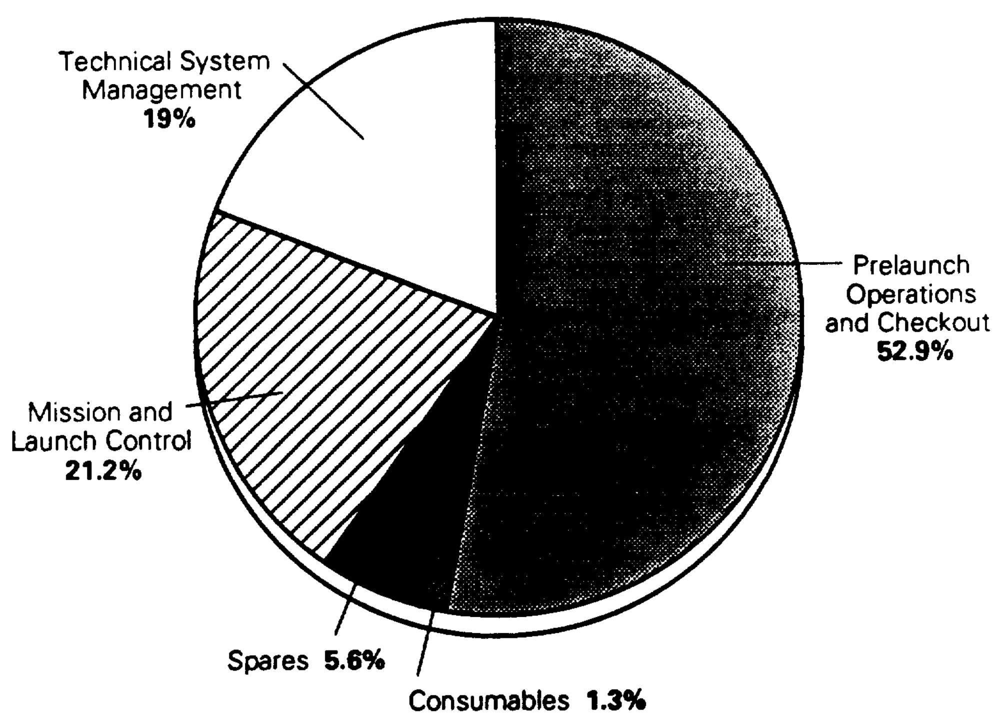

Figure 47. Operations LCC breakdown.

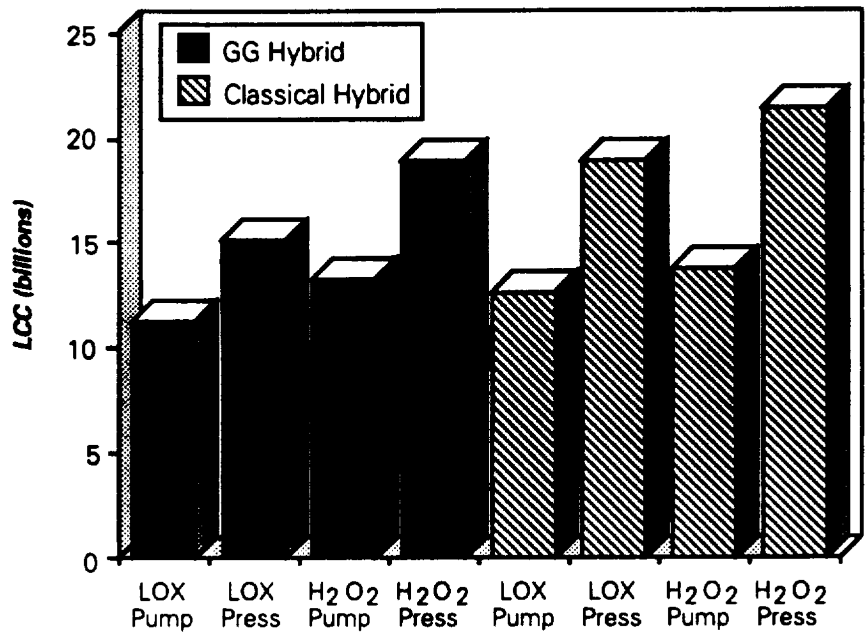

Figure 48. Hybrid concept LCC comparison.

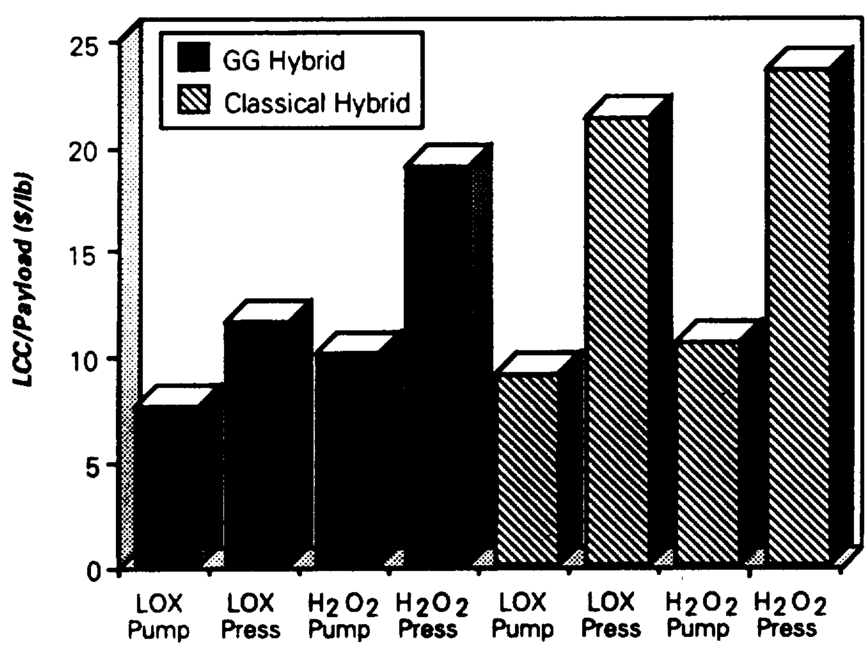

Figure 49. Hybrid concept comparison, LCC/payload.

Because the gas generator hybrid and LOX oxidizer provided the lowest cost, they were selected for the final point design trades. To calculate the costs of the eight conceptual designs, certain assumptions had to be made to estimate system weights and complexity factors used in the cost algorithms. The assumptions used are:

- Classical hybrid utilized gaseous oxidizer injection.
- LOX was preburned to 667K using propane or methane.
- Turbopumps utilized propane or methane.
- H_2O_2 was decomposed by a catalyst bed.
- Fuel utilization for the gas generator was 98 percent, and the classical hybrid was 95 percent.
- Turbopump system had pump-out capability.

2.6.2.1 Pump-Fed Versus Pressure-Fed - The cost drivers for the pressure-fed evaluation are the cost of the oxidizer tank and pressurization system. The pressure-fed oxidizer tank operates at a pressure 5.44 MPa (800 psi) greater than the thrust-chamber pressure and results in a tank design that is heavier and more complex. In addition, the pressurization system has to be larger with sufficient expulsion capability to empty the LOX tank. The pump-fed design provides head pressure to the pumps. This system operated at 6.37 MPa (925 psi) less than the thrust-chamber pressure and required a smaller expulsion system to generate this pressure. As a result, the cost of the pressure-fed tank and expulsion system exceeded the cost of the turbopumps.

Oxidizer-to-fuel mixture ratio has a greater impact on the evaluation of pressure-fed systems than on pump-fed. The optimum LOX/gas generator hybrid mixture ratio is 1.5; for the peroxide/gas generator it is 4.0; for the classical hybrid/LOX it is 2.75; and the classical hybrid/H_2O_2 mixture ratio is 6.5. Higher mixture ratios increase the pressurizing gas requirements and, therefore, cost and weight of this system. Low mixture ratios are preferred for pressure-fed systems to keep the cost competitive with the turbopump designs.

2.6.2.2 Classical-Versus-Gas Generator Hybrid - The difference in the LCC of the pump-fed classical hybrid and the pump-fed gas generator hybrid is the cost of the methane or propane system required to drive the turbopumps and preburn the LOX. The weight of the preburner system and weight of sliver reduces the payload performance of the classical hybrid compared to the gas generator hybrid.

Hydrogen peroxide offers a higher system density, but reduced I_{sp} compared to LOX. This results in a smaller, heavier hybrid booster which costs more than the LOX system.

2.6.2.3 Reusable-Versus-Expendable Hybrid Boosters - A reusable hybrid booster was synthesized from the reference expendable booster configuration by adding a recovery system. The weights of the reusable components were increased by 20 percent to compensate for the higher safety margins and the design complexity factor for reusable components was increased 40 percent.

The following components were assumed to be reusable: flexseal nozzle, TVC, aft skirt, attachments, interstage, recovery system, electronics and instrumentation, pumps, piping, injector, valves and the igniter housing.

The refurbishment cost of solid rocket components was obtained from the STACEM code. The refurbishment cost of liquid components was assumed to be 25 percent of the theoretical first unit cost. The design life of reusable components was baselined at 10 reuses with an attrition rate of 10 percent. All composite materials were assumed to be expendable. The number of boosters required was assumed to be equal to: (total quantity of boosters required) * (units per booster)/(design life)/(attrition rate). A cost of recovery equipment and facilities was assumed to be $100 million for Mission I (1 flight/month, 150 missions) and $200 million for Mission II (1 flight/week, 650 missions).

The LCC of the reusable booster was calculated for the two mission models. The design life and attrition rate assumptions were varied to determine the LCC sensitivity. In addition, the number of flights per year was varied to determine when the expendable booster provided lower LCC than the reusable hybrid booster. The results of this study are shown in Figure 50. The cost drivers of this trade study are the mission model, the design life of reusable components, the recovery attrition rate, and the recovery system DDT&E.

The major cost driver is the mission model. The difference between the expendable and reuseable designs ranges from 0 percent at 50 missions to 12.3 percent at 650 missions.

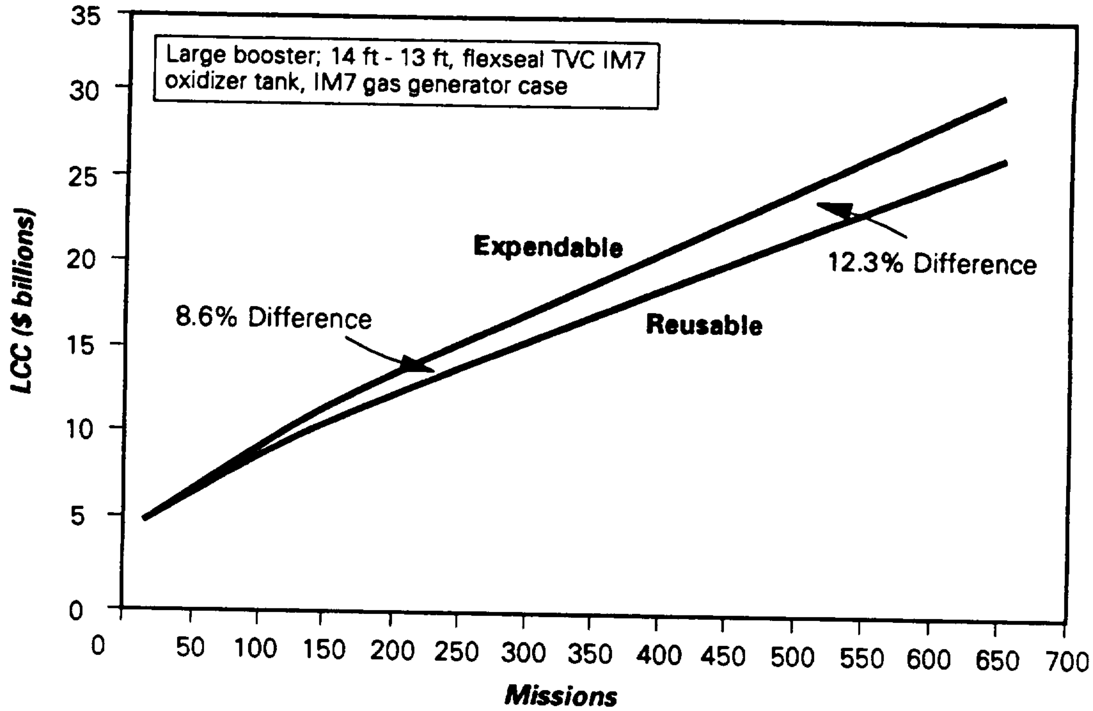

Figure 50. Life cycle cost vs number of missions.

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If the design life of the components was increased 100 percent, the LCC of the reusable system would decrease 2.5 percent; cutting the attrition rate in half results in a decrease in the reusable booster LCC of 1.7 percent; and doubling the recovery DDT&E increases the reusable booster LCC less than 1 percent.

For the specified mission models, the reusable hybrid booster is predicted to have lower cost. It must be recognized that these reference vehicles are not optimized, and that apparent LCC advantage of the reusable system may be decreased by the following:

- Reduction in the number of flights per year.
- The reduced payload capability of the reusable design (approximately 3 percent) reduces the LCC/pound of payload advantage.
- Advanced nozzle and thrust chamber technology.
- Increased recovery system DDT&E.

2.6.3 Reference Design Trade Studies

The reference design was used with the Boeing model to parametrically determine the impact of mixture ratio, nozzle expansion ratio, chamber pressure and body diameter on LCC with Mission I (150 missions). In this study, a single operating condition was varied over a range of values and the effects on weight, payload, and costs were calculated for a composite (carbon-epoxy) gas generator case and LOX tank, and repeated for a number of other material combinations, summarized in Table 24.

Table 24. Configuration and Material Parametrics.

	Full-Scale		Quarter-Scale	
	Pump-Fed	Pressure-Fed	Pump-Fed	Pressure-Fed
LOX Tank				
Carbon-Epoxy (IM-7/EPON 826)	X	X	X	X
Aluminum	X	X	X	X
Aluminum-Lithium	X	X	X	X
Gas Generator				
Carbon-Epoxy (IM-7/EPON 826)	X	X	X	X
D6AC Steel	X	X	X	X

NOTES: Operating conditions varied:

Mixture Ratio:	1.3 - 2.9
Chamber Pressure:	4.13 - 15.14 MPa (600 - 2,200 psia)
Nozzle Expansion Ratio:	6 - 22
Body Diameter:	3.05 - 5.49 meters (10 - 18 feet)

2.6.3.1 Large Booster, Pump-Fed Trades

Mixture Ratio - The LCC and LCC/pound of payload-versus-mixture ratio calculations are shown in Figures 51 and 52. There is a large difference in the gas generator case and LOX tank operating pressures for the pump-fed designs. The selection for the optimum mixture ratio becomes a trade between weight and cost of the gas generator case and LOX tank and performance changes.

The sensitivity of LCC to mixture ratio depends on the component raw materials cost and manufactured component weight. Composite components,

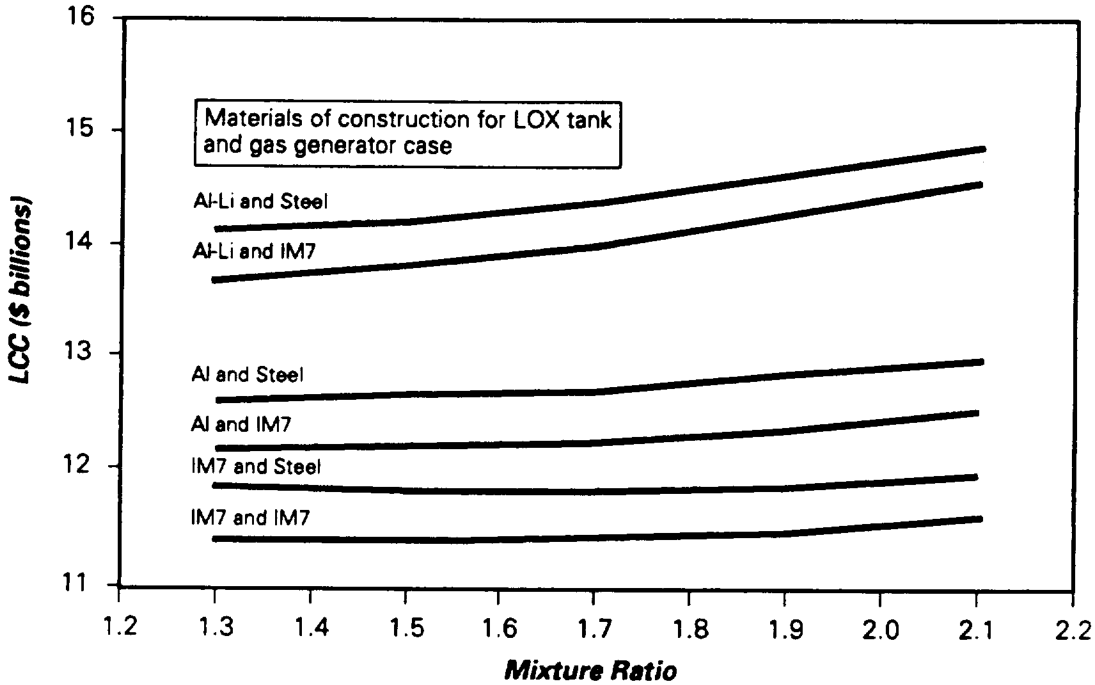

Figure 51. Mixture ratio vs LCC for the pump fed large booster.

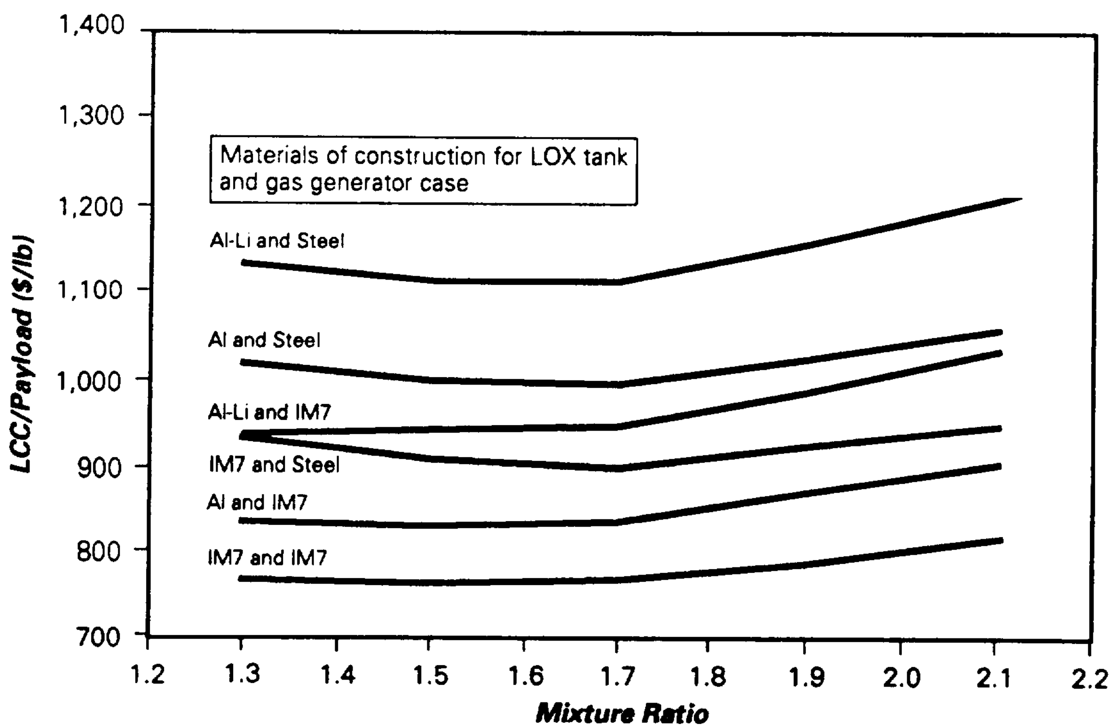

Figure 52. Mixture ratio vs LCC/payload for the pump fed large booster.

manufactured from carbon fiber (IM-7), are the least sensitive because of the high cost of the fiber. On the basis of LCC/pound of payload, the optimum mixture ratio for the D6AC steel gas generator case is 1.7. The optimum mixture ratio for the composite gas generator case is 1.5.

Expansion Ratio - LCC and LCC/payload results-versus-expansion ratio are shown on Figures 53 and 54. The LCC decreased with increasing expansion ratio due to the design criteria of constant vacuum total impulse and increase in vacuum I_{sp} with expansion ratio. LCC/pound of payload calculations account for increased inert weight of the nozzle-versus-performance improvement. The decreased sea level thrust is due to overexpansion of the nozzle. The optimum expansion ratio for LCC/pound of payload is in the range of 10 to 14.

Chamber Pressure - LCC and LCC/pound of payload results versus chamber pressure (Pc) are shown in Figures 55 and 56. This trade is driven by the gas generator case material. For a composite gas generator case, LCC and LCC/pound of payload decrease with increasing pressure. For a steel case, LCC and LCC/pound of payload are a minimum between 6.88 MPa (1,000 psi) and 9.63 MPa (1,398 psi) and increase with increasing pressure.

Body Diameter - LCC and LCC/pound of payload results versus body diameter are shown in Figures 57 and 58. The selection of a body diameter is a trade between inert weight and performance due to increased drag reference area and the change in booster length. Other potential problems associated with the booster body diameter, such as interface problems with the core vehicle and launch equipment, and transportation were not considered in this trade.

For each material system, LCC decreases uniformly with increasing diameter; however, the cost/pound of payload does not follow the same trend. Each material system has an optimum diameter ranging from 3.66 meters (12 feet) (all carbon-epoxy) to 4.88 meters (16 feet) (aluminum-lithium oxidizer tank and carbon-epoxy gas generator).

2.6.3.2 Large Booster, Pressure-Fed

Mixture Ratio - The LCC and LCC/pound of payload-versus-mixture ratio calculations are shown in Figures 59 and 60. The results indicate that a composite gas generator case and LOX tank have the lowest LCC and LCC/pound of payload. Costs are driven by the LOX tank materials of construction. Other materials will have a higher cost, and aluminum-lithium has the highest cost.

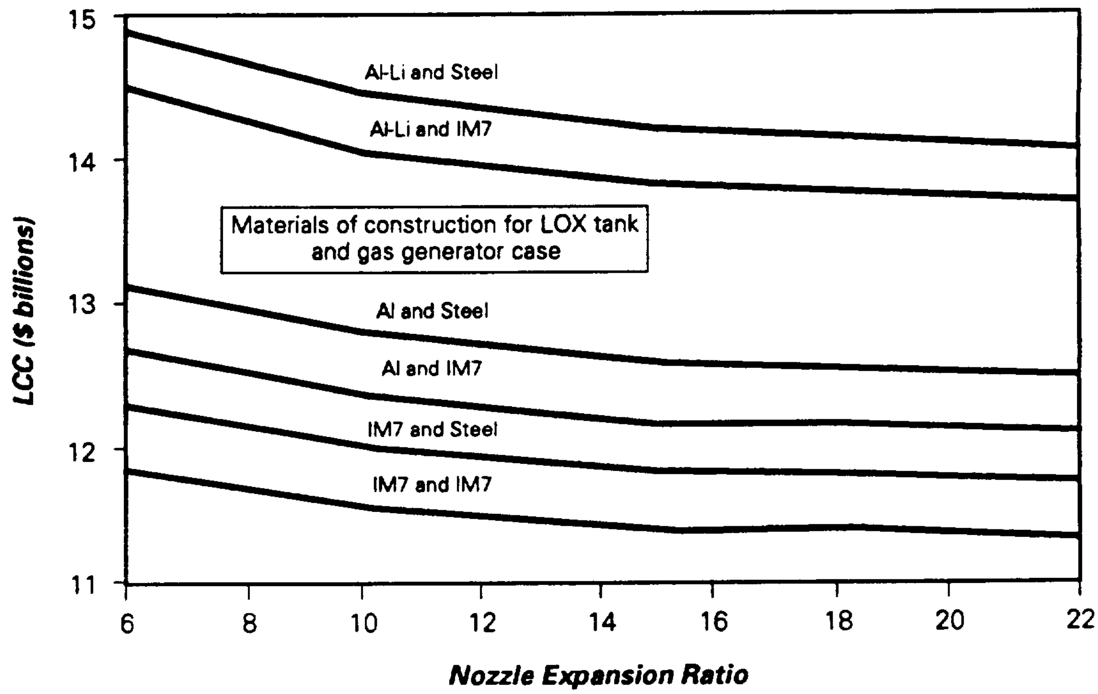

Figure 53. LCC vs expansion ratio for the pump fed large booster.

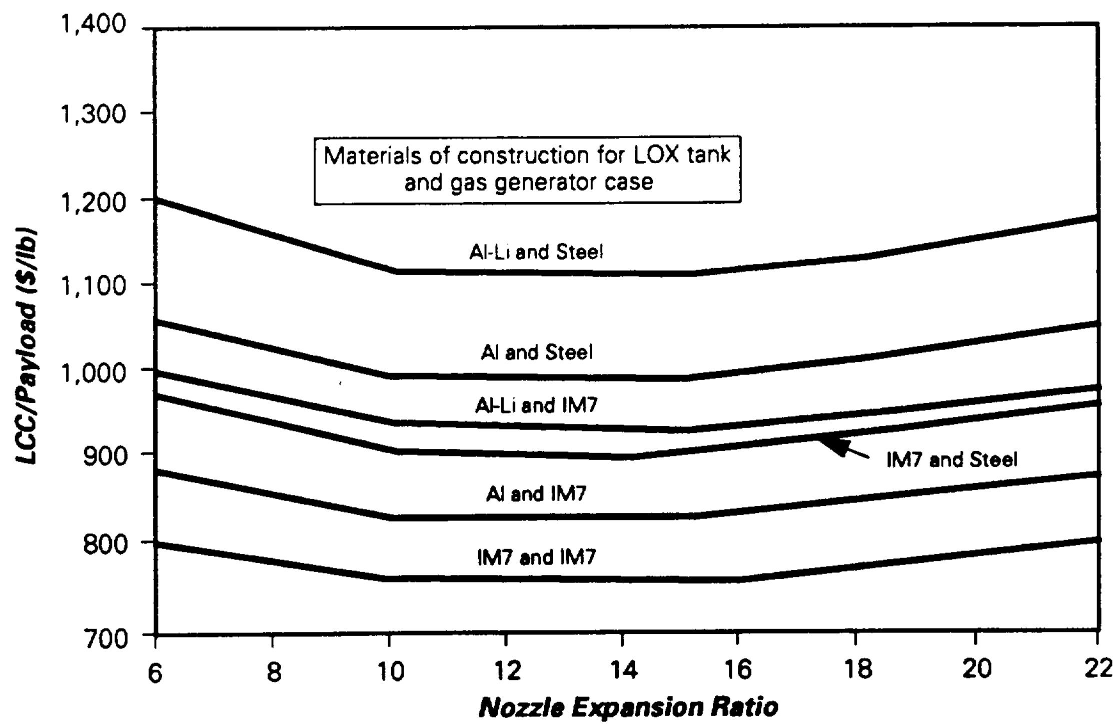

Figure 54. LCC/payload vs expansion ratio for the pump fed large booster.

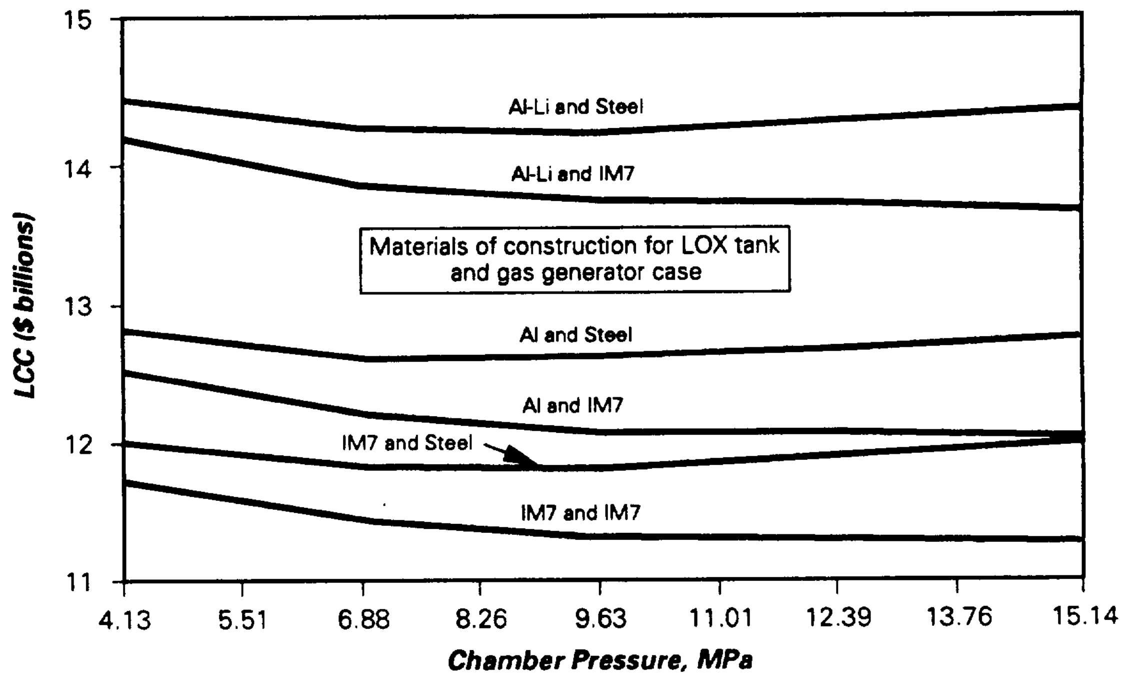

Figure 55. LCC vs chamber pressure for the pump fed large booster.

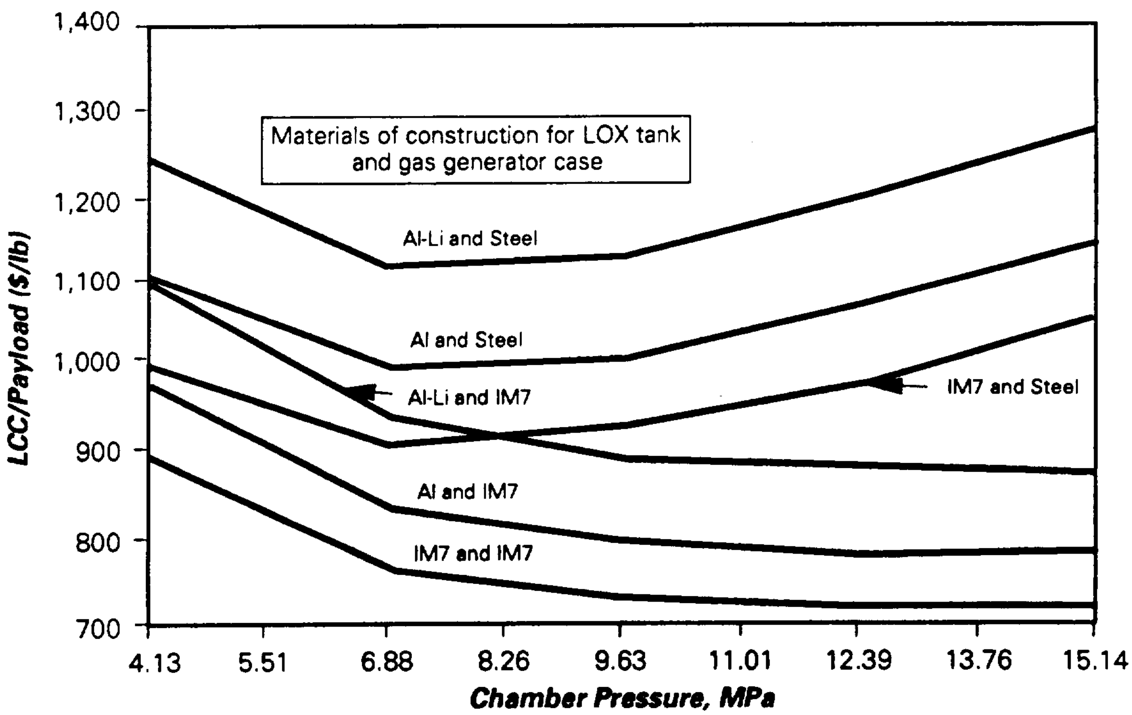

Figure 56. LCC/payload vs chamber pressure for the pump fed large booster.

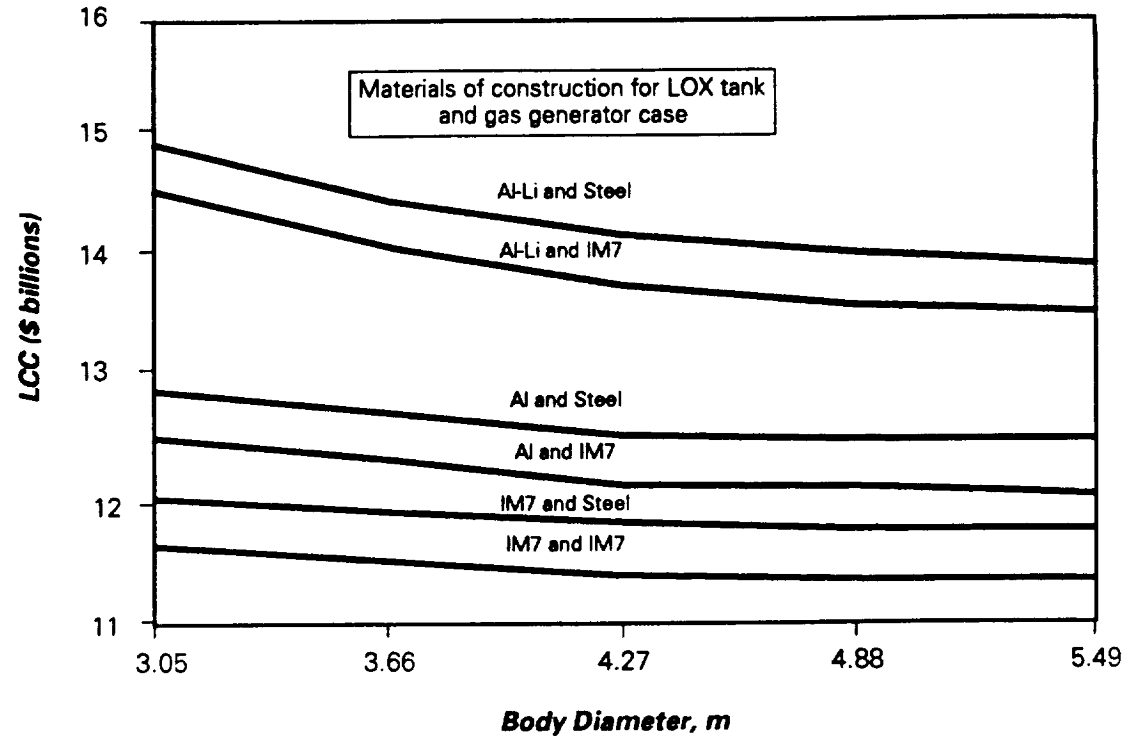

Figure 57. LCC vs body diameter for the pump fed large booster.

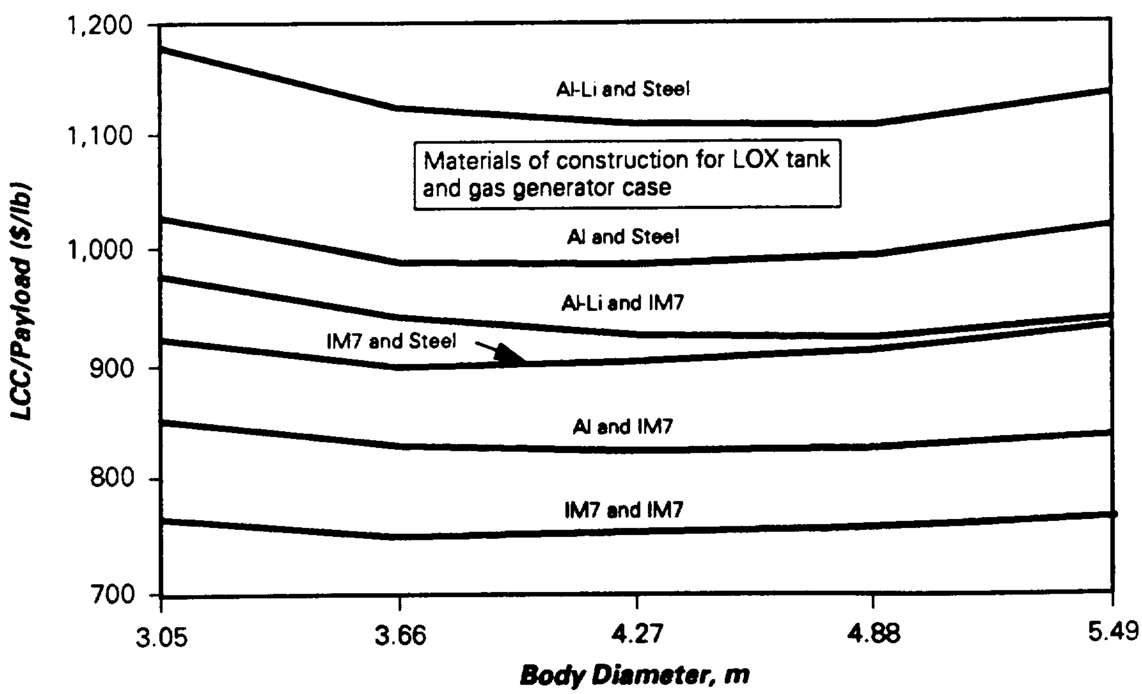

Figure 58. LCC/payload vs body diameter for the pump fed large booster.

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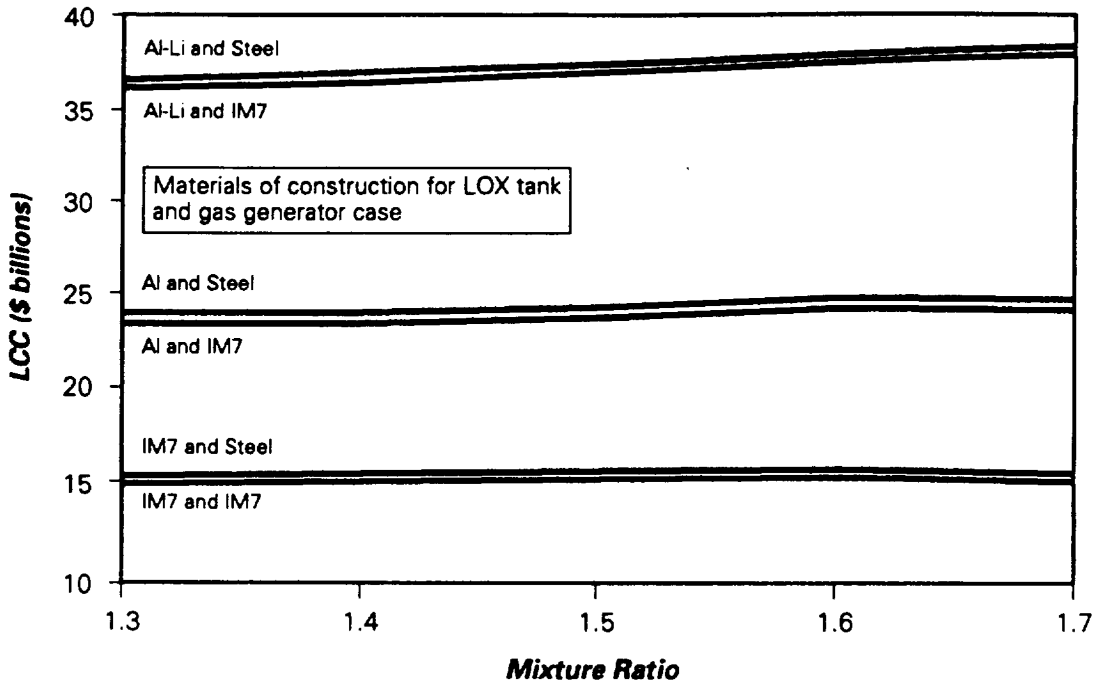

Figure 59. Mixture ratio vs LCC for the pressure fed large booster.

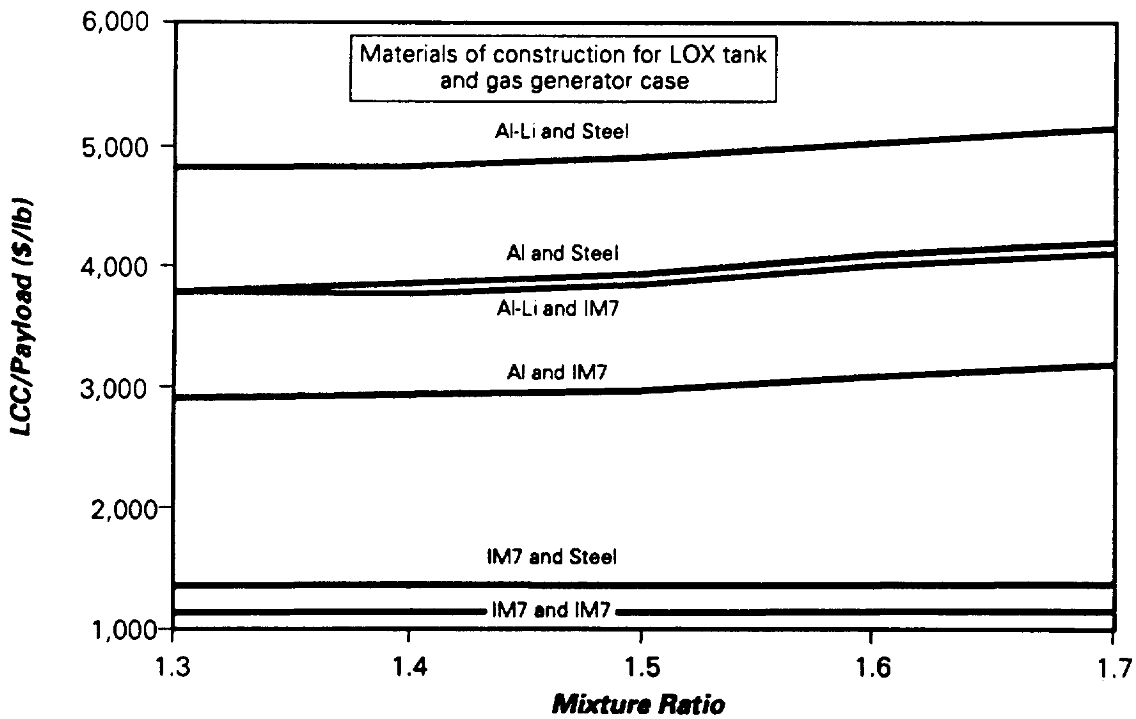

Figure 60. Mixture ratio vs LCC/payload for the pressure fed large booster.

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Expansion Ratio - LCC and LCC/payload results versus expansion ratio are shown in Figures 61 and 62. LCC decreases with increased expansion ratio due to the increase in vacuum I_{sp}, and design criteria of constant vacuum total impulse. LCC/payload accounts for an increased weight of the nozzle versus performance and decreased sea level thrust due to overexpansion of the nozzle. The optimum expansion ratio for LCC/pound of payload is in the range of 10 to 14.

Chamber Pressure - LCC and LCC/pound of payload results versus chamber pressure (Pc) are shown in Figures 63 and 64. LCC increases significantly with increasing pressure. LCC/pound of payload is optimum at 6.88 MPa (1,000 psi) for a composite gas generator case, and 4.13 MPa (600 psi) for a steel gas generator case.

Body Diameter - LCC and LCC/pound of payload results versus body diameter are shown in Figures 65 and 66. The pressure-fed design cost, like the pump-fed case, decreases with increased diameter. Minimum LCC/pound of payload is obtained at diameters greater than 3.66 meters for the composite gas generator case and approximately 3.66 meters for the steel gas generator case.

2.6.3.3 Quarter-Size Booster - LCC results for the quarter-size were consistent with the full-size. Payload performance of the quarter-size vehicle was not determined. Summary tables of the quarter-size booster results are included in Appendix B.

2.6.4 Additional Booster Design Studies

To complete the LCC evaluations, additional design complexities were investigated to determine the impact on the reference booster cost. The items evaluated using the Boeing model were: (1) thrust vector control; (2) propellant reserve; (3) design margins; (4) volumetric loading of gas generator propellant; and (5) oxidizer pump-out capability.

2.6.4.1 Thrust Vector Control - Fluid injection thrust vector control (FITVC) offers the potential for improved reliability and reduced life cycle cost compared to the flexseal nozzle and actuation system presently used on the shuttle booster.

To define the cost benefit of FITVC, a series of reference boosters was synthesized. We assumed a deflection requirement of 1, 3, and 5 degrees to calculate the mass of fluid required. We assumed duty cycles for the TVC of

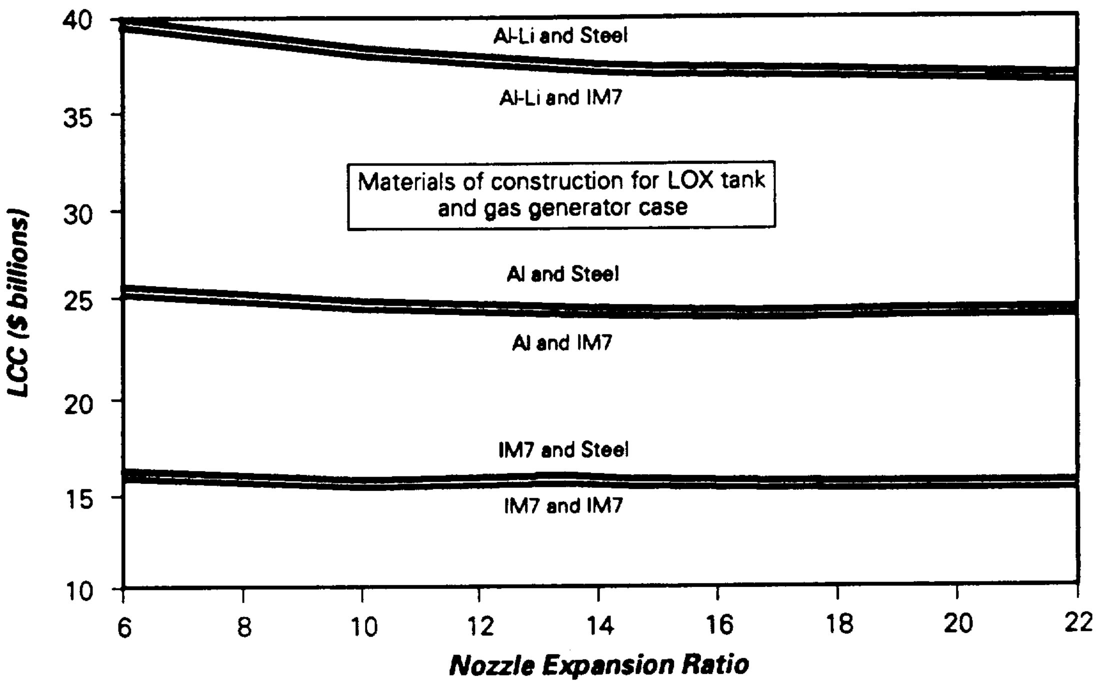

Figure 61. LCC vs expansion ratio for the pressure fed large booster.

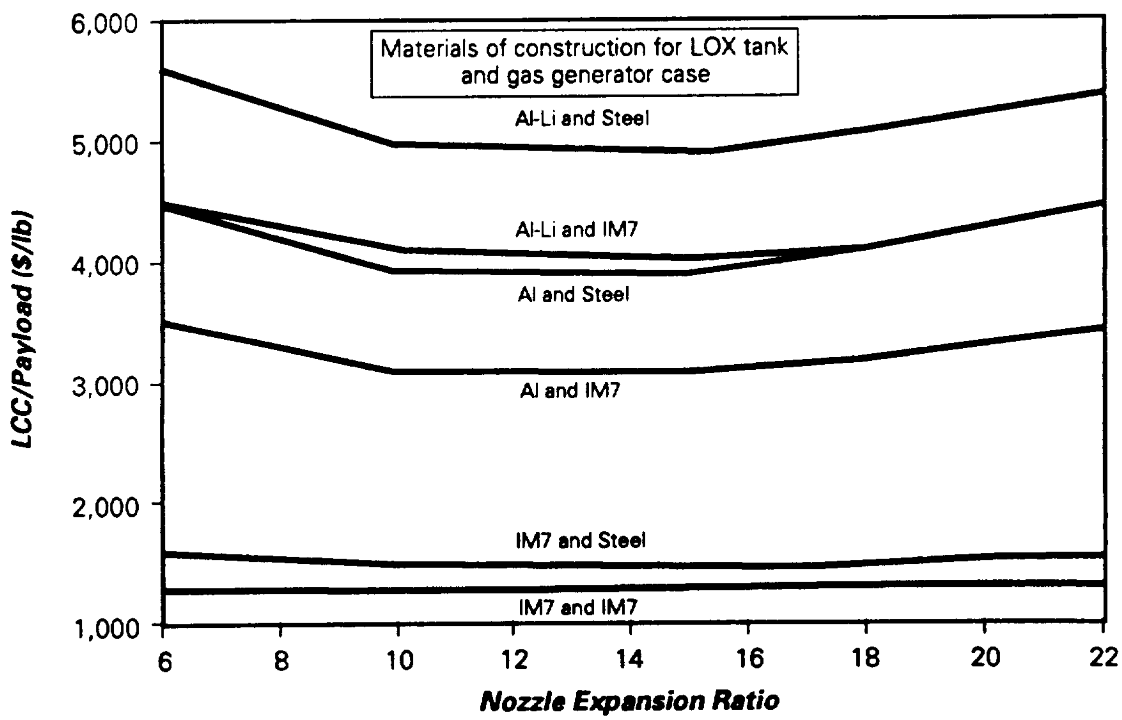

Figure 62. LCC/payload vs expansion ratio for the pressure fed large booster.

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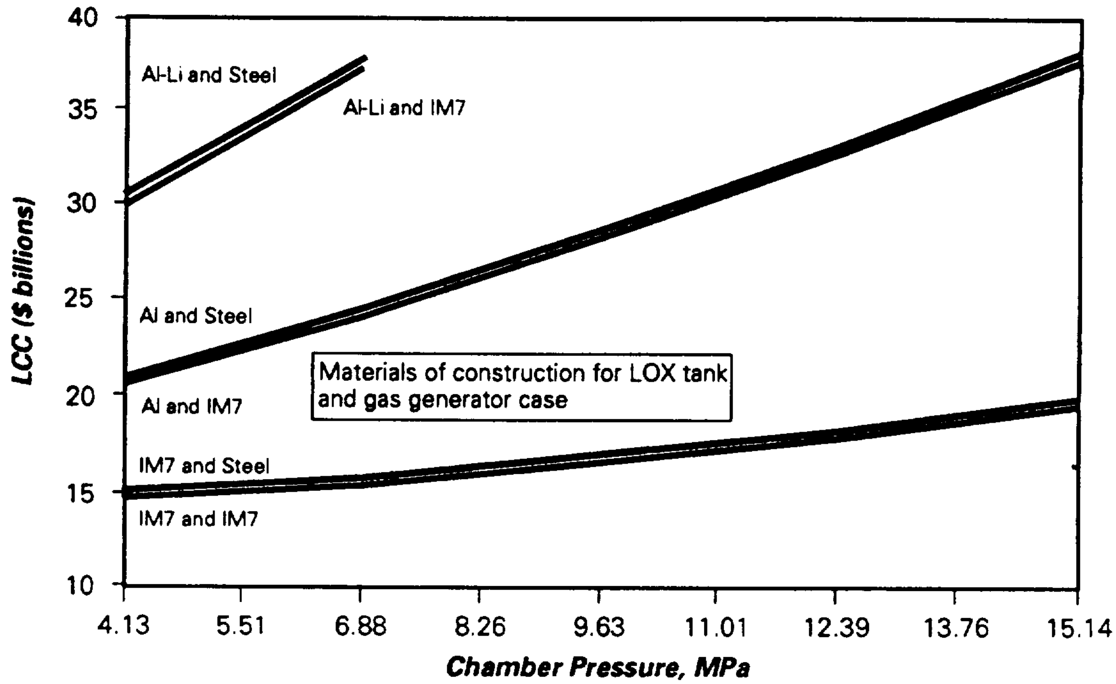

Figure 63. LCC vs chamber pressure for the pressure fed large booster.

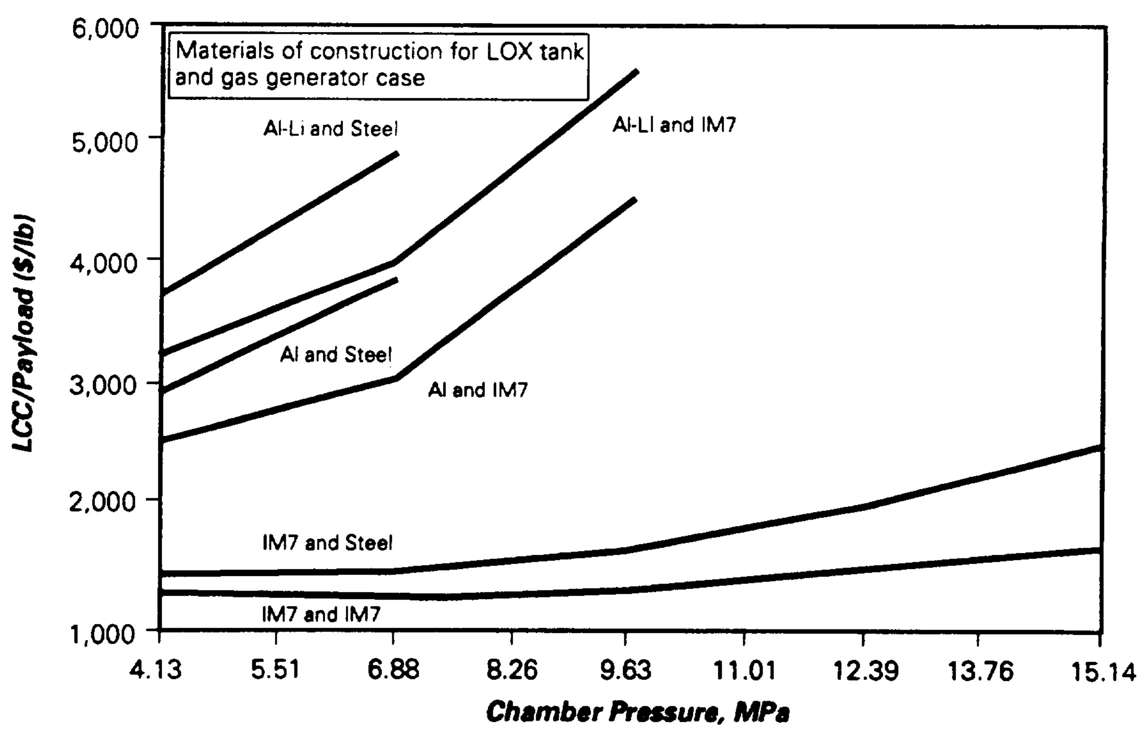

Figure 64. LCC/payload vs chamber pressure for the pressure fed large booster.

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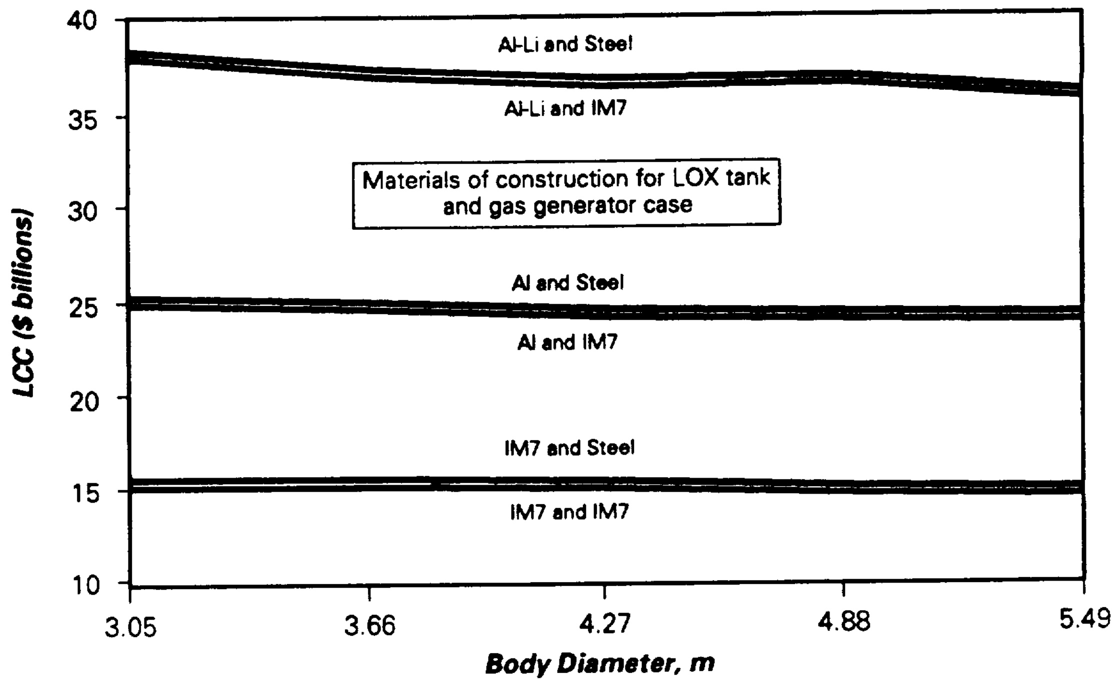

Figure 65. LCC vs body diameter for the pressure fed large booster.

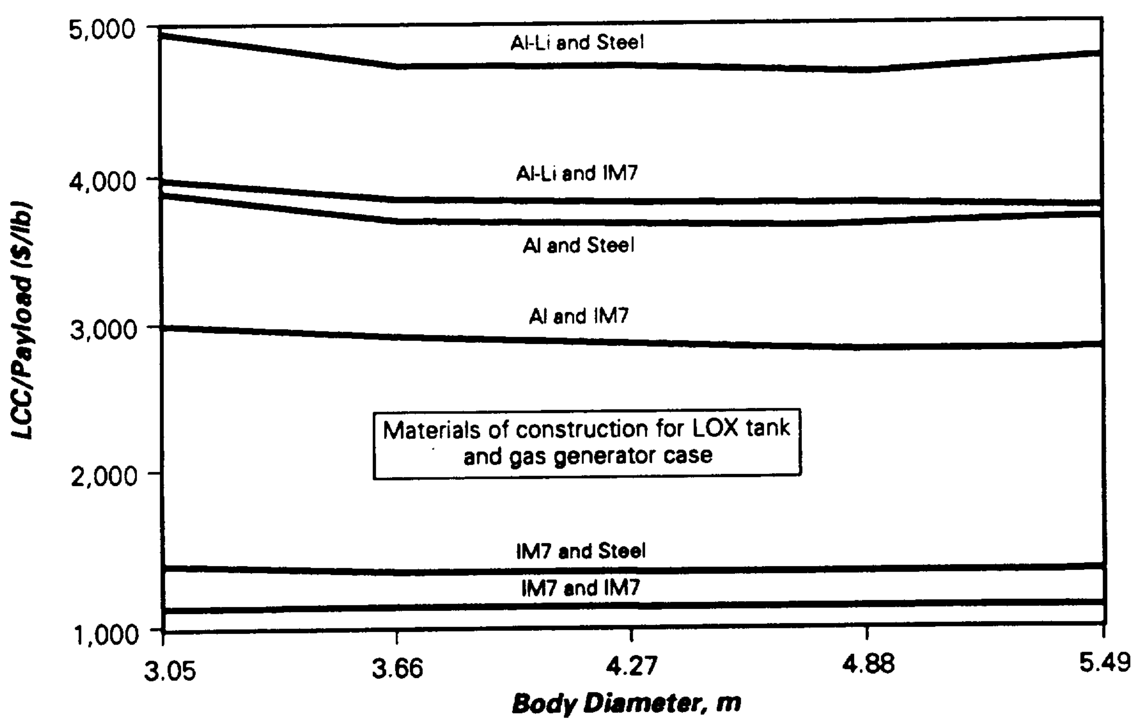

Figure 66. LCC/payload vs body diameter for the pressure fed large booster.

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20, 40, and 60 percent of the total burn time (135 seconds) at each deflection. This was done for both the full-size and quarter-size boosters.

The large booster LCC and LCC/pound of payload FITVC results are shown in Figures 67 and 68. The quarter-scale FITVC results are shown on Figure 69. Over the range evaluated, FITVC offers a lower LCC than the reference case ($11.4 billion full-size; $13.2 billion quarter-size). However, on a basis of LCC/pound of payload, the break-even point is at 3° deflection and 60 percent duty cycle. The LCC/pound of payload break-even point would be increased by the use of an advanced nozzle technology such as the MBA nozzle.

2.6.4.2 Propellant Reserve - The ability to extinguish the hybrid booster through the termination of the oxidizer flow allows propellant reserve to be designed into the booster. Reserve propellant improves the booster reliability through the elimination of propellant failure modes associated with variable burning rates and combustion efficiency.

Hybrid boosters were synthesized with propellant reserve increased to 5 percent. The impact of reserve propellant on LCC and LCC/pound of payload

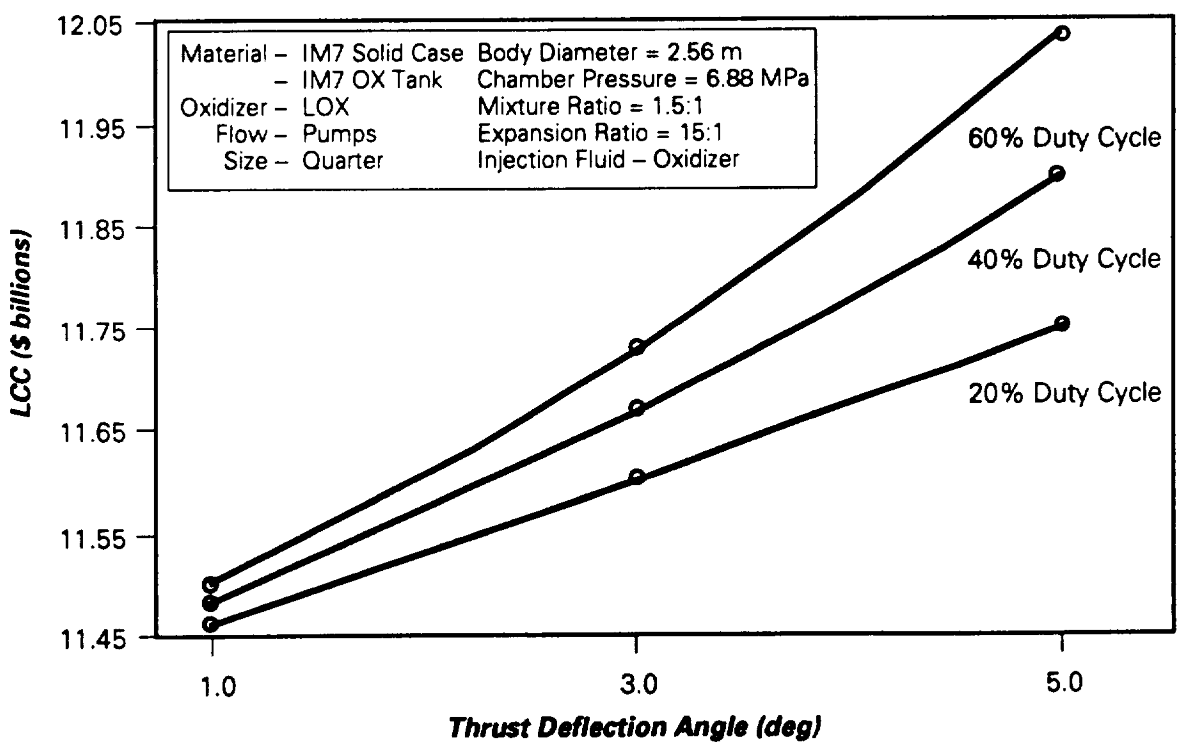

Figure 69. <u>*Life cycle cost vs thrust deflection for the quarter scale booster.*</u>

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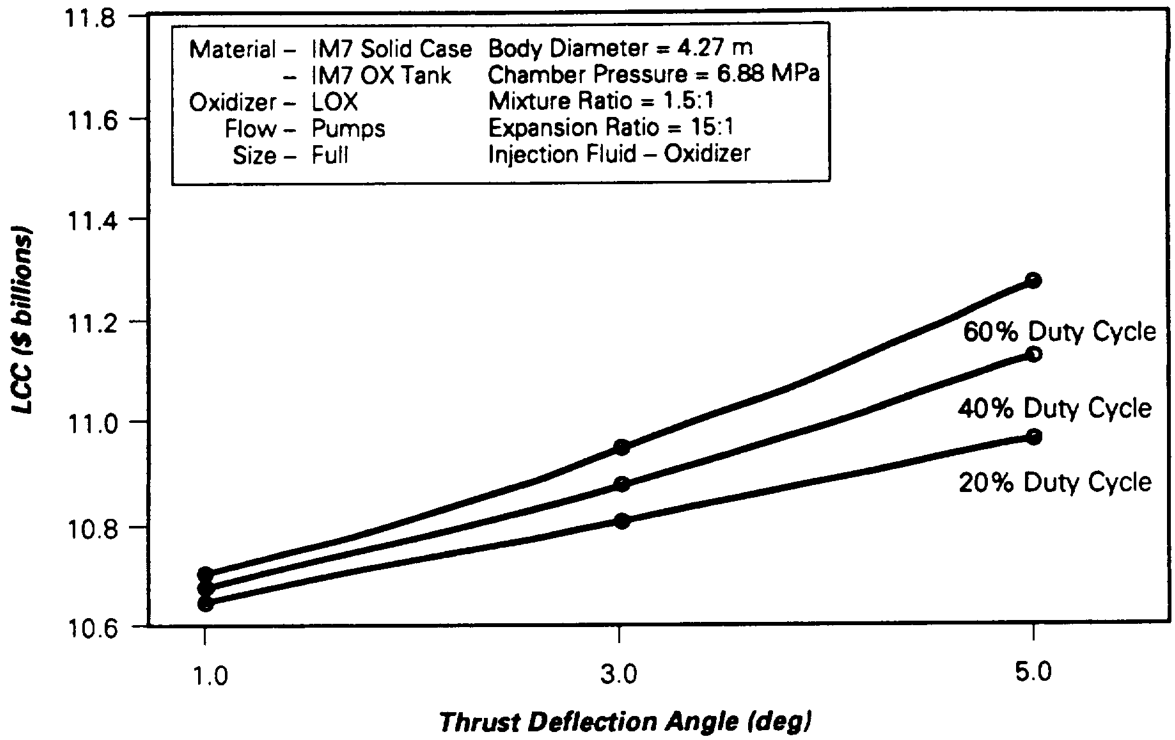

Figure 67. Life cycle cost vs thrust deflection for the large booster.

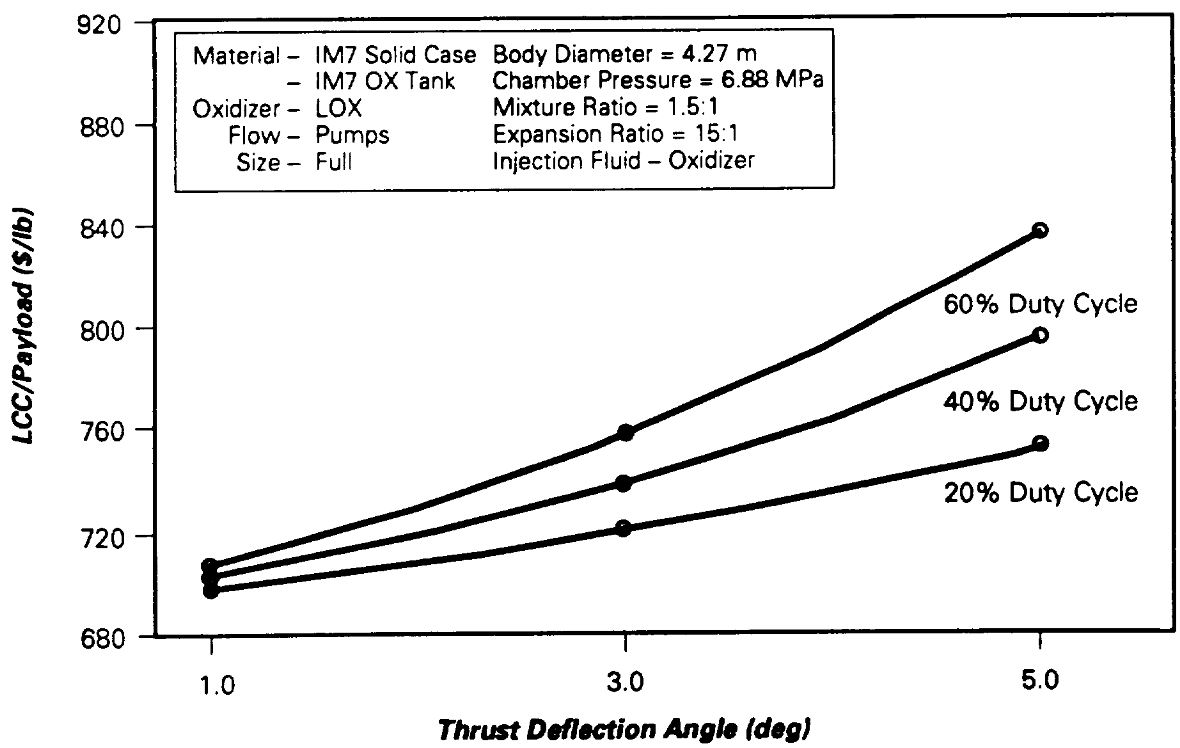

Figure 68. Cost per pound of payload vs thrust deflection for the large booster.

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is shown in Figures 70 and 71. The increased inert weight to store the reserve propellant resulted in a decrease of 4,400 kilograms (9,700 pounds) of payload. LCC increased 1.2 percent ($140 million) and LCC/pound of payload increased $200/kilogram ($91/pound).

2.6.4.3 Design Margins - Increased design margins may be used to improve reliability. Hybrid boosters were synthesized with the structural safety margins of the gas generator case and oxidizer tank increased from 1.6 to 1.9. The impact on LCC and LCC/pound of payload are shown in Figures 72 and 73. Inert weight is increased approximately 1,225 kilograms (2,700 pounds) as a result of the increase in design margin resulting in a payload decrease of 363 kilograms (800 pounds). LCC was increased by less than 1 percent ($100 million), and LCC/pound of payload increased 1.5 percent [$26/kilogram ($12/pound)].

2.6.4.4 Volumetric Loading of Gas Generator Propellant - Lower volumetric packing of the gas generator case may provide processing cost reductions or may be required due to burning rate limitations of scavenged clean propellants. To document the cost associated with changes in gas generator case volumetric loading, a hybrid booster was synthesized with different grain port radii to reflect volumetric loadings of 75 to 95 percent. The results in terms of LCC LCC/pound of payload are shown in Figures 74 and 75.

2.6.4.5 Oxidizer Pump-Out Capability - The reliability of a single-string pump-fed system is lower than the reliability of a pressure-fed system. The pump-fed system reliability can be improved through redundancy. The use of four pumps, each sized for 133 percent of the design flow rate, with common manifold and independent block valves, assures that the required oxidizer feed rate can be maintained if one pump fails. Pump-out capability has a minimal impact on LCC. LCC of the reference design increases by 0.35 percent ($40 million) and LCC/pound of payload increases 0.66 percent [$11/kilogram ($5/pound)]. Pump-out capability provides a predicted reliability equivalent to a pressure-fed system, but at a lower LCC.

2.6.5 Hybrid Model Optimizer Results

To complete the parametric trade studies, the hybrid booster model (HAVCD) was used to predict the optimum conditions for the hybrid booster for Mission I. The optimizer is a tool that can provide valuable insight into the

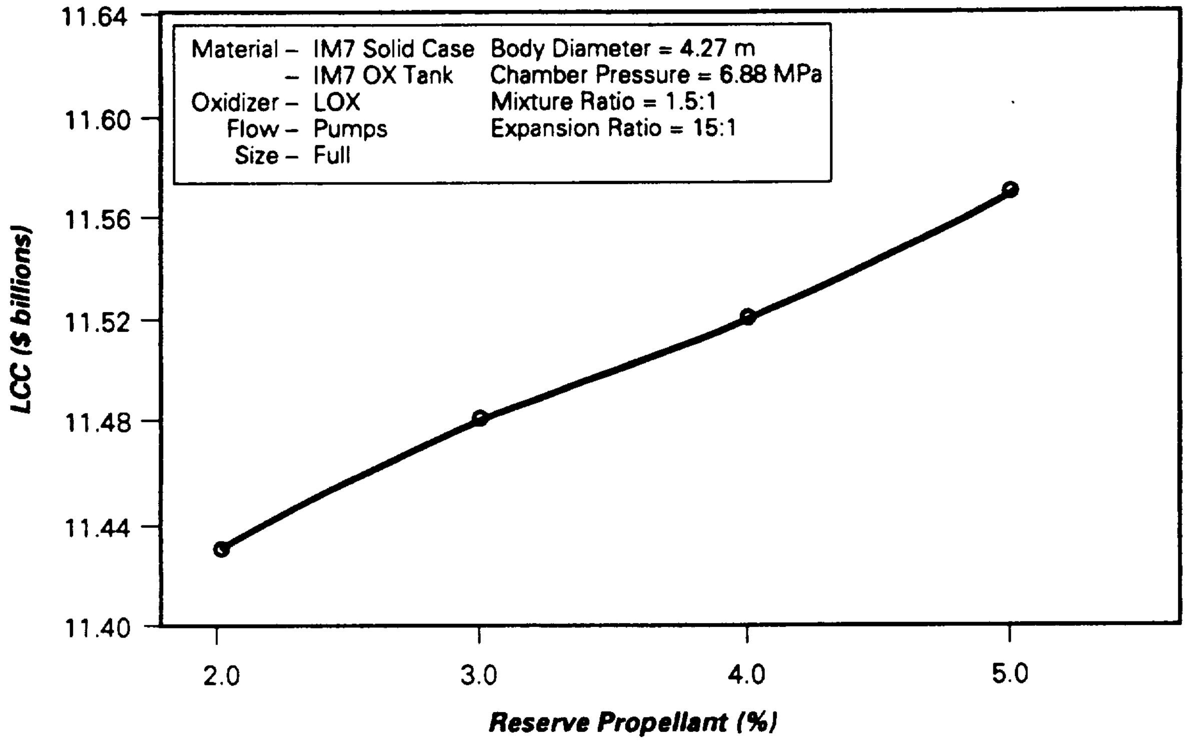

Figure 70. Life cycle cost vs reserve propellant.

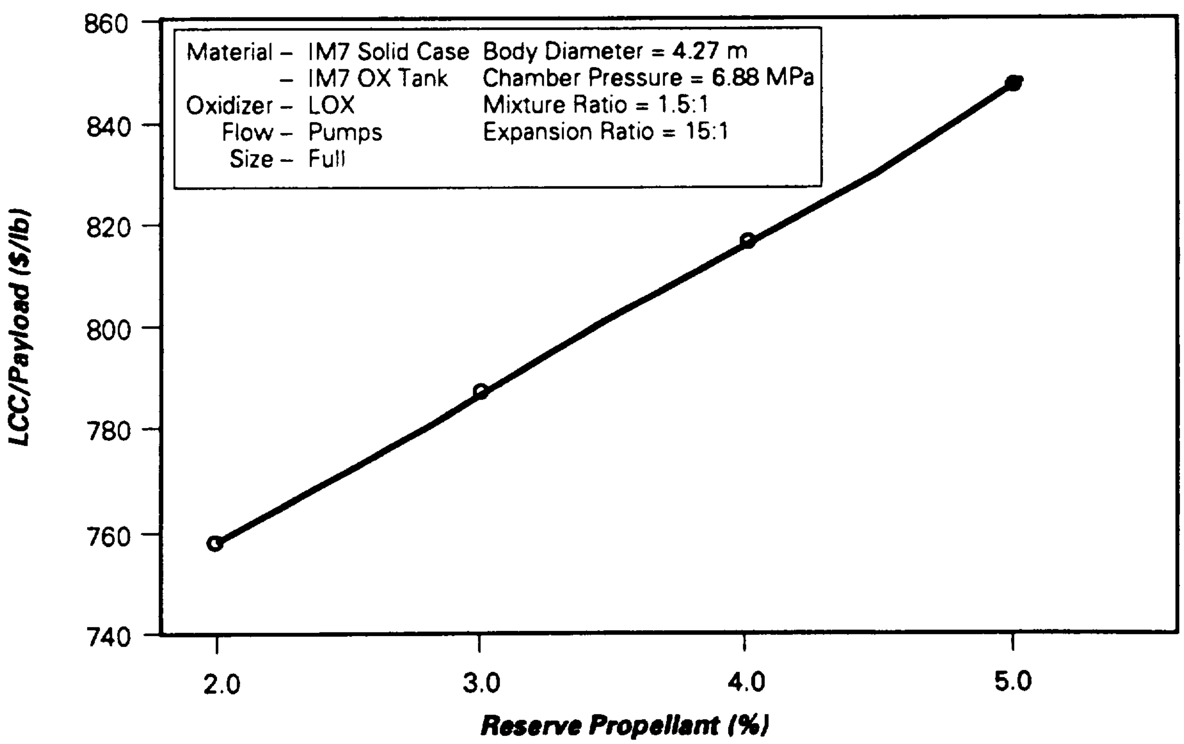

Figure 71. Cost per pound of payload vs reserve propellant.

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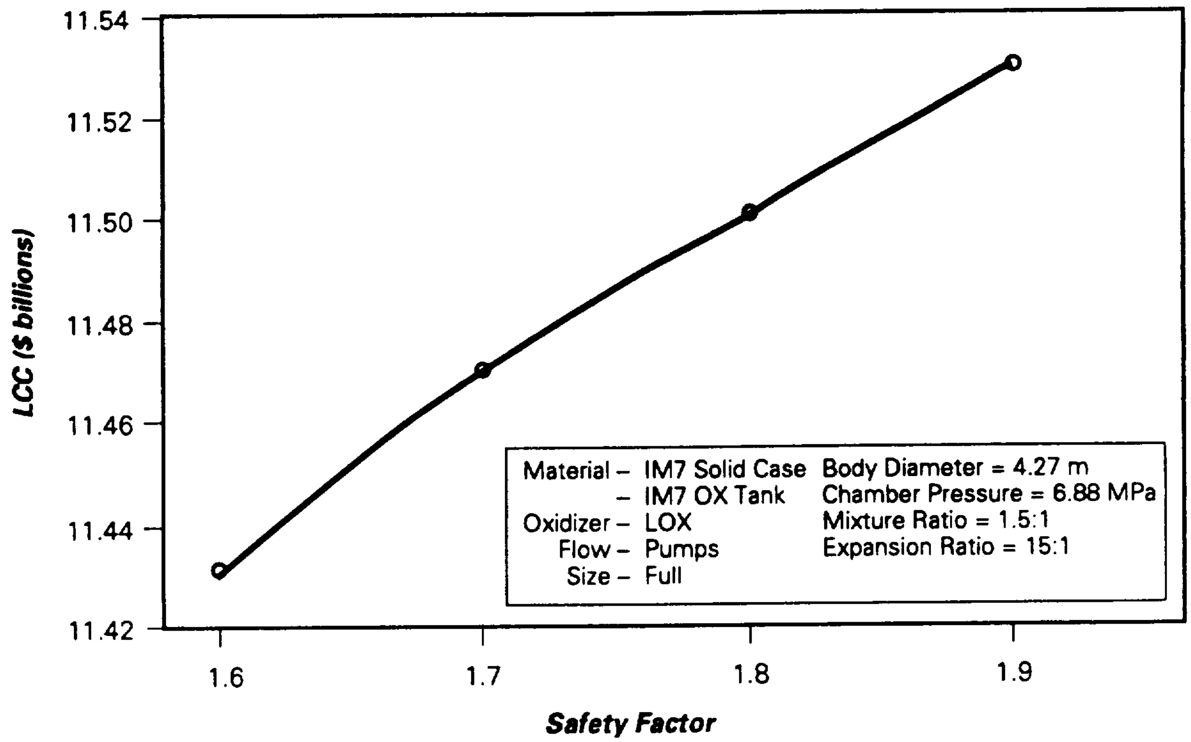

Figure 72. Life cycle cost vs safety factor.

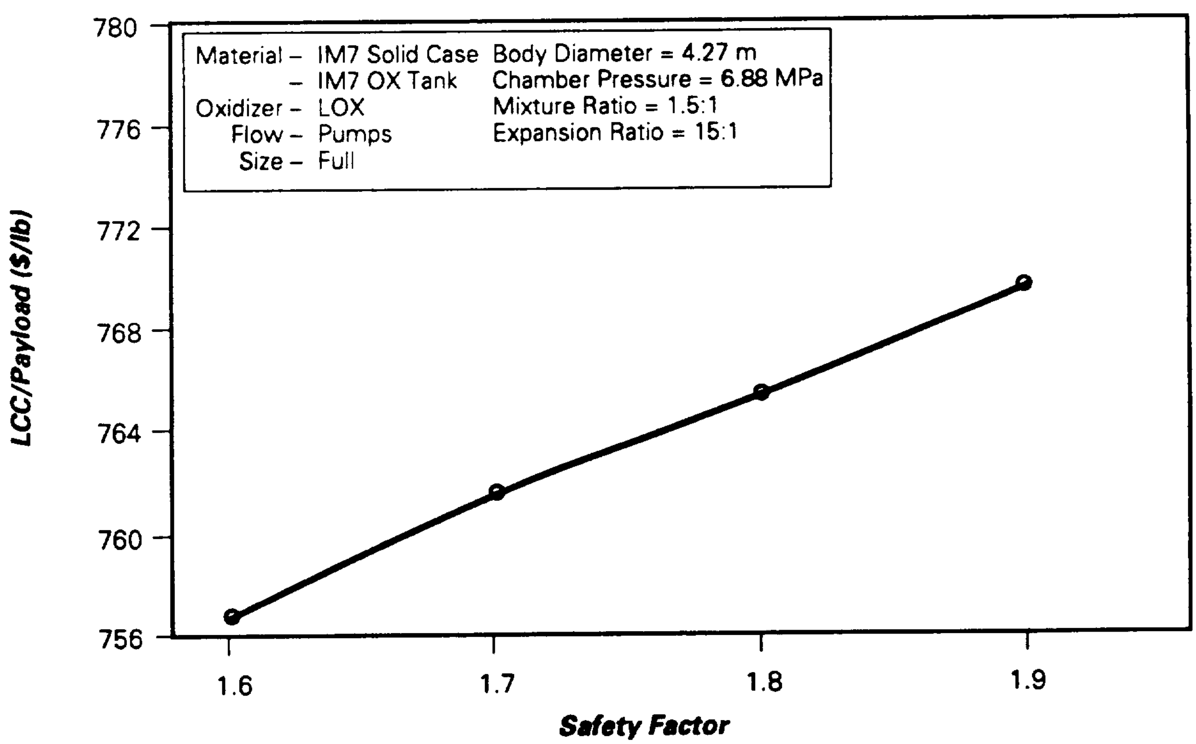

Figure 73. Cost per pound of payload vs safety factor.

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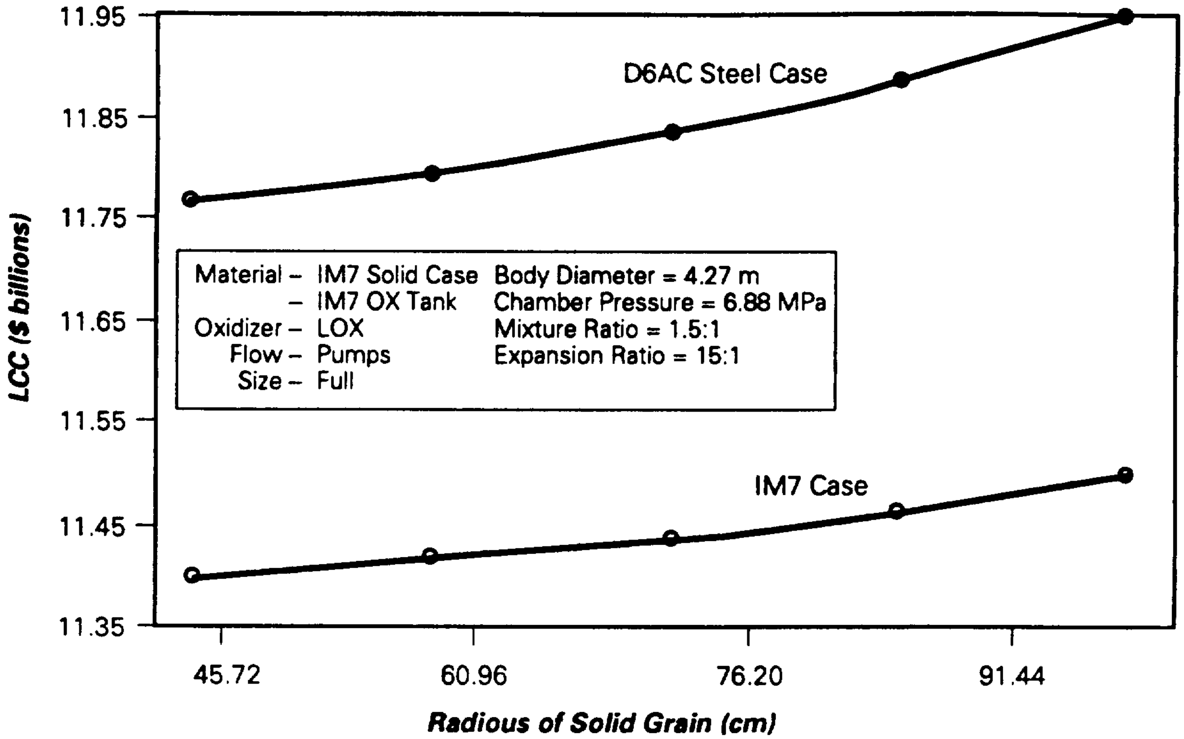

Figure 74. Life cycle cost vs grain radius.

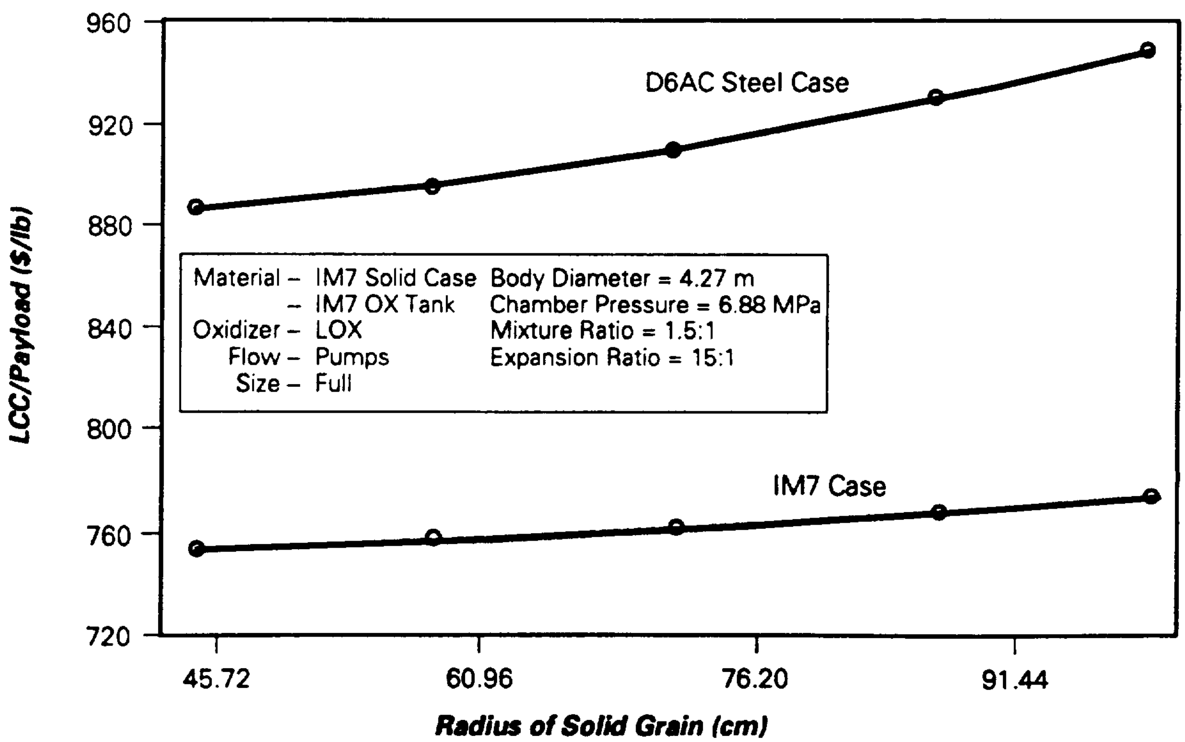

Figure 75. Cost per pound of payload vs safety factor.

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design of a booster system with a significant number of operating variables and materials choices such as the hybrid. Operating conditions were optimized for different results: minimum LCC/pound of payload, LCC/pound of payload, maximum payload, minimum LCC, minimum empty weight, and minimum gross lift-off weight (GLOW). Optimizer results are shown in Table 25.

The optimum conditions and results were consistent based on LCC, LCC/pound of payload, or payload. LCC/pound of payload results, when optimizing for minimum empty weight or minimum GLOW, were not consistent with the others. The difference between the optimum operating conditions and the reference conditions is in the selection of the chamber pressure; increasing chamber pressure from the reference 6.88 MPa (1,000 psi) to approximately 12.4 MPa (1,800 psi) results in lower LCC and improved performance.

The LCC of the reference booster varied 2.2 percent from the optimum. LCC/pound of payload for the reference booster was 5.4 percent higher than the optimum.

2.7 Reliability Analyses

2.7.1 Introduction

ARC performed a preliminary reliability analysis for the gas generator hybrid. The predicted reliabilities for the pressure-fed and pump-fed point designs are estimated to be 0.9985 and 0.9987, respectively. Only reliabilities related to the actual flight of the components were included; items such

Table 25. Hybrid Booster LCC Trade Studies Optimized Booster Design.

Optimized On	Mixture Ratio	Chamber Pressure (MPa)	Body Diameter (m)	Expansion Ratio	LCC ($ x 10^9)	Payload (kg)	$ Per kg Payload
$/Payload	1.496	12.8	3.9	18.8	11.207	47,491	1,581
Payload	1.487	12.4	3.0	17.5	11.480	48,126	1,588
LCC	1.600	12.8	4.8	22.5	11.180	46,992	1,584
Empty Wt	1.429	4.8	4.3	7.0	11.970	43,822	1,819
GLOW	1.600	7.3	3.7	25.0	11.390	43,577	1,740
*Reference Conditions	1.50	6.88	4.3	15.0	11.430	45,858	1,652

as prelaunch reliability and their effects on the probability of booster operation were not considered for this evaluation because of the limited data available at Boeing.

2.7.2 Reliability Block Diagram

Figure 76 presents the reliability block diagram for the hybrid booster system. The hybrid propulsion system, Figure 77, is presented as a seven-component system consisting of: (1) a solid fuel gas generator; (2) nozzle; (3) oxidizer feed system; (4) preburner; (5) turbopumps; (6) turbine drive system; and (7) pressurization system.

The block diagram is intended to imply operation of an independent series system requiring successful operation of each subsystem in the order depicted to obtain successful booster functioning. A series reliability math model is therefore used to arrive at the overall booster reliability and has the form:

$$R_T = R_1 \times R_2 \times R_3 \times \ldots \times R_r = \prod_{s=1}^{r} Rs \qquad (2)$$

Where: R_1 = predicted component reliability
R_T = system reliability

2.7.3 Reliability Estimation Procedures

One of the widely used distributions to describe "time to fail" for electrical and mechanical components and systems is the Weibull distribution:

$$R\ (t) = \exp\left[-\left(\frac{t-\gamma}{\delta}\right)^{\beta}\right] \qquad (3)$$

where: γ = location parameter
δ = scale parameter
β = shape parameter
t = mission time

When assuming $\delta = 0$ and $\beta = 1$, the above equation reduces to an exponential distribution with $\lambda = \frac{1}{\delta}$:

$$R\ (t) = \exp\left[-\lambda t\right] \qquad (4)$$

where: λ = failure rate

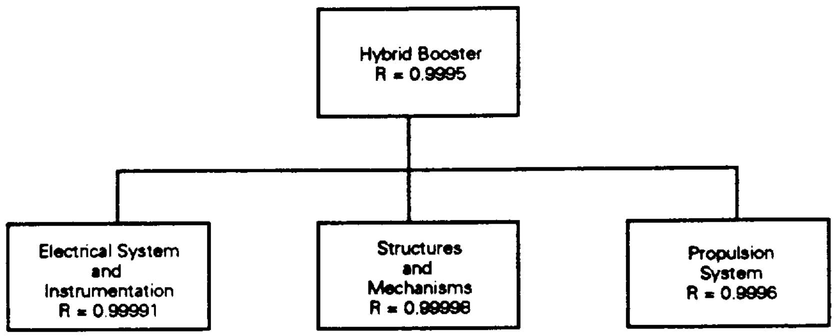

Figure 76. *Hybrid booster reliability block diagram.*

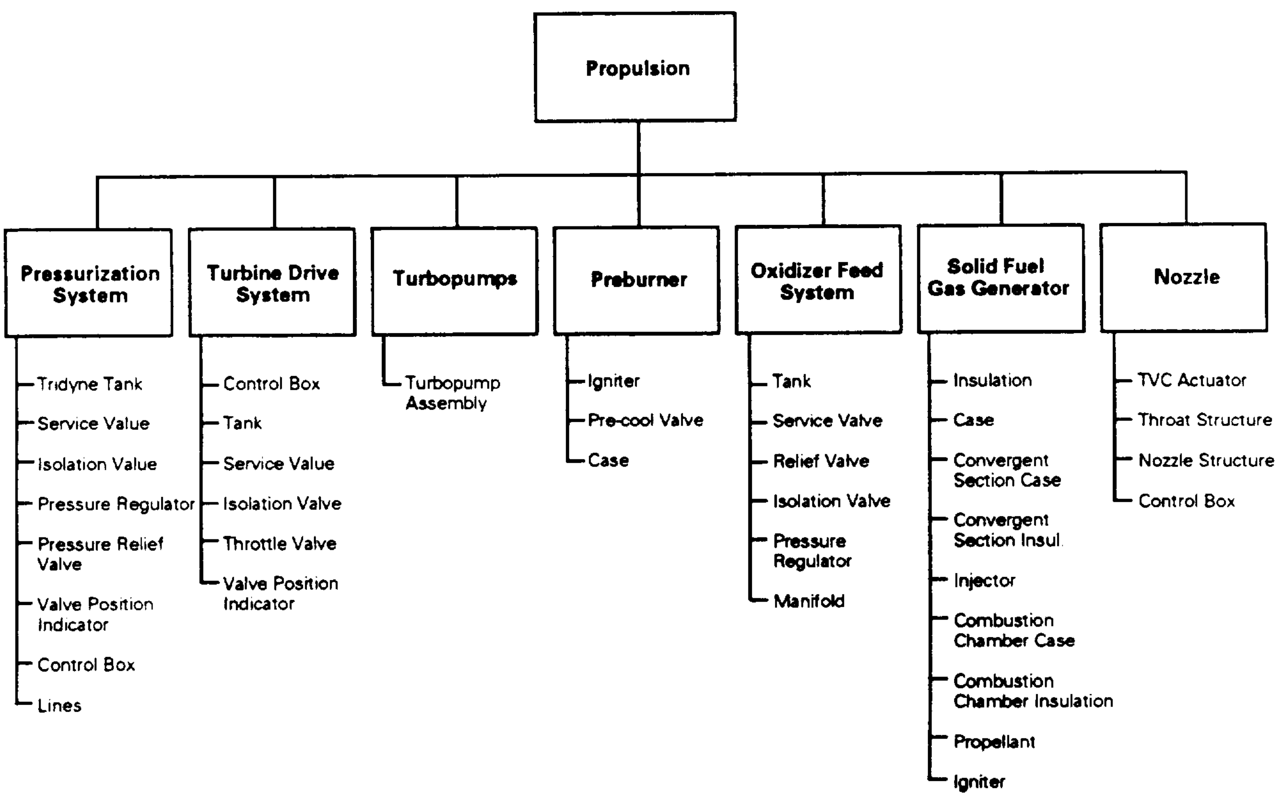

Figure 77. *Hybrid propulsion system.*

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The reliability of the structural components is based upon the probability that the strength of the structural material exceeds the applied stress. Mathematically, this is expressed as:

$$R = P\ (R > S) \tag{5}$$

$$= \int_{-\infty}^{\infty} f_s\ (S)\ [\int_{-\infty}^{\infty} f_r\ (R)\ dR]\ dS \tag{6}$$

$$= \int_{-\infty}^{\infty} f_r\ (R)\ [\int_{-\infty}^{\infty} f_s\ (S)\ dS]\ dR \tag{7}$$

where: f_r (R) = PDF of strength
f_s (S) = PDF of stress
PDF = probability density function

For normal distributions of strength and stress, the reliability of the structural components is calculated using the equation:

$$R = \Phi\ [\frac{\bar{R} - \bar{S}}{(\sigma_R^2 + \sigma_S^2)^{\frac{1}{2}}}] \tag{8}$$

where: $\bar{R}$ = mean value of the material strength
$\bar{S}$ = mean value of the material stress
σ_R = standard deviation of the material strength
σ_S = standard deviation of the material stress

For non-normal distributions of strength and stress, the reliability of the structural components is calculated by evaluating the integrals for these other distributions. When there are only two random variables involved, a computer program called POFAIL is used to evaluate the integrals for other distributions.[15] When there are more than three random variables, an approximation method called Mean Value First Order Second Moment (MVFOSM) method has

15. Ang, A. H. S., Wilson, Tang H., "Decision Risks and Reliability," Probability Concepts in Engineering Planning and Design, Vol. V, 1984.

been used.[16] A computer program for MVFOSM has been written and utilized for the Hybrid Propulsion Technology Program.

Prior to beginning the reliability analysis, an estimate of failure rates was obtained from a variety of data sources, reliability handbooks, engine analysis reports, and engineering estimates by reliability engineers. Table 26 is a compilation of component failure rates and sources for each line item.

Once the component reliabilities were predicted, the values were given to Boeing for input into their RELIB computer subroutine data files. This subroutine, part of the HAVCD program, calculated the subsystem reliability, and finally, the booster reliability. Predicted reliabilities for the pressure-fed and pump-fed designs were 0.9985 and 0.9987, respectively. This was lower than the reliability goal of 0.9995 established for the booster, but was the result of low historical data for the following components: (1) gas generator case; (2) combustor case; (3) nozzle; and (4) TVC. These specific items are emphasized for design improvement and validation during the Phase II activities.

2.7.4 Failure Modes and Effects Analysis

To identify potential impact of each failure on mission success, a preliminary failure mode and effects analysis (FMEA) has been performed for the pump-fed design. The major ground rule observed in the analysis is the single failure analysis; i.e., each failure is considered to be the only failure in the system. However, when critical failure modes are identified, the effects of a simultaneous failure mode which might worsen the situation are also investigated.

Another ground rule observed in the analysis is at the assembly level. The parts are considered to be assemblies of failure-free components as a result of having undergone receiving inspection and being dispositioned as acceptable. The FMEA is presented in Table 27.

16. Ang, A. H. S., Cornell, C. A., "Reliability Bases of Structural Safety and Design," Journal of Structural Division, ASCE, Vol. 100, Sept. 1974.

Table 26. Hybrid Component Predicted Failure Rates.

Item	Failure Rate*	Source
ELECTRICAL SYSTEM AND INSTRUMENTATION		
Avionics	20.0	9
Wiring	1.5	9
Batteries/Power Supply	169.0	9
Instrumentation	155.0	9
STRUCTURES AND MECHANISMS		
Nose Shell and TPS	45.0	6
Interstage	1.0	6
Aft Skirt	1.0	6
Attachment Struts	1.0	6
Separation System	0.0	7
PROPULSION		
Pressurization System		
Tridyne Tank	37.5	1
Service Valve	1.6	2
Isolation Valve	11.0	5
Pressure Regulator	55.3	5
Pressure Relief Valve	9.8	5
Valve Position Indicator	155.0	4
Control Box	20.0	9
Lines	5.0	6
Turbine Drive System		
Control Box	20.0	9
Tank	37.5	1
Service Valve	1.6	2
Isolation Valve	11.0	5
Throttle Valve	10.2	5
Valve Position Indicator	155.0	9
Turbopumps		
Turbopump Assembly	164.0	10
Preburner		
Igniter	74.0	2
Precool Valve	35.0	8
Case	1.0	6
Oxidizer Feed System		
Tank	37.5	1
Service Valve	1.6	2
Relief Valve	9.8	5
Isolation Valve	11.0	5
Pressure Regulator	55.3	5
Manifold	1.1	6

*Per 1.0×10^6 hours.

Table 26. Hybrid Component Predicted Failure Rates (Cont'd).

Item	Failure Rate*	Source
Solid Fuel Gas Generator		
Insulation	6.3	11
Case	134.0	11
Convergent Section Case	134.0	11
Convergent Section Insulation	6.3	11
Injector	45.0	11
Combustion Chamber Case	6.3	11
Combustion Chamber Insulation	6.3	11
Fuel	56.0	11
Igniter	85.0	11
Nozzle		
TVC Actuators	321.0	7
Throat Structure	248.5	11
Nozzle Structure	248.5	11
Control box	20.0	9

The following are the sources or assumptions used to assign failure rates:

1. Spacecraft Reliability Prediction, Boeing Aerospace, 1985, unpublished report based on analysis of a variety of systems.
2. NPRD-3, Non-Electric Parts Reliability Data, Reliability Analysis Center, RADC, Griffiss AFB, New York 21985.
3. Boeing Document D290-10404-1, Reliability & Maintainability Allocations, Assessments and Analysis Report - IUS System, CDRL #050A2, Boeing Company/Aerospace Division, Seattle WA 1979.
4. YVAE-80-005, Space System Effectiveness Requirements Document for Space Transportation System: Inertial Upper Stage (IUS), USAF/Space Division, 1981.
5. Engineering judgement for environment adjustment of data from item 2.
6. Assumed as based on high design margins of safety.
7. Calculated for data in item 3.
8. Engineeering judgement for environmental adjustment of data from item 4.
9. Assumed for components of undefined complexity.
10. Engineering judgement for environmental adjustment of data from Boeing Document D232-10627-1 AGM-86, Reliability and Maintainability Allocation Assessment and Analysis Report, 1980.
11. Based on a combination of data from CSD Titan SRMs and Thiokol SRM data.

*Per 1.0 x 10^6 hours

Table 27. Preliminary Failure Modes and Effects Analysis.

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Electrical System and Instrumentation	Avionics	Provide sequencing and control of all hybrid booster elements	Improper pressure, flowrates, thrust, vectoring signals	Vehicle shutdown, fire/explosion	Faulty wiring, improper power temperature, shock, moisture
	Wiring	Transmit electrical power and control signals	Interruption of power and/or signal	Avionics shutdown, loss of control, valves stuck, potential explosion	Structural damage or defect to batteries, environment degradation
	Batteries/Power Supply	Provide power for avionics, range safety, valves	Too much or too little power	Avionics shutdown - loss of control, valves stuck, potential explosion	Structural damage or defect to batteries, environment degradation
	Instrumentation	Provides data on valve position, pressures, temperatures, actuator positions for feedback and control	Incorrect readings	Overcompensation in control, performance loss	
Structures and Mechanisms	Nose Shell and TPS	Provide aerodynamic protection and drag minimization	Buckling, deformation	Secondary damage to impacts on booster or core vehicles	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Interstage	Structurally connect solid motor/gas generator case and oxidizer tank	Buckling, deformation	Booster structural failure	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Structures and Mechanisms (Cont'd)	Aft Skirt	Provide structural interface between solid motor case, launch pad, and TVC actuators	Buckling, deformation	Booster static structural failure, improper TVC if mount is deformed	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Attachment Struts	Provide structural attachment between hybrid booster and parallel core vehicle	Buckling, deformation	Vehicle structural failure, secondary damage due to booster/core impacts	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Separation Motors	Provide for physical separation of hybrid boosters and core vehicle after hybrid burnout	Failure to ignite, burst motor case	Incorrect separation, possibly impacting core vehicle	Cracked propellant, structural failure or rupture of casing
Propulsion Pressurization System	He tank	Store 10,000 psi gaseous Helium	Leakage or rupture	Possible explosive rupture; performance loss; loss of booster	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Service Valve	Fill and drain access to tank	Rupture or internal leakage of valve seat	Performance loss; loss of booster	Structural failure, crack or fracture of valve body or seal failure
	Isolation Value	Allow pressurant to reach pressure required	Stuck in wrong position	Inadequate flow of pressurant	Electrical failure, pyro charge failed to ignite

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Propulsion Pressurization System (Cont'd)	Pressure Regulator	Maintain proper pressure in flow of pressurant	Failure to open or close, leakage, contamination	Inadequate oxidizer flow due to low pressure, blow out pressure relief valve	Manufacturing defect, corrosion
	Pressure Relief Valve	Prevents overpressure of oxidizer system	Failure to open or close, seal leakage	Overpressure of oxidizer systems, possible tank rupture	Seal failure, structural failure, crack or fracture of valve body
	Control Box	Provides control activitation and servicing of pressurant flow	Does not provide correct control and sequencing to valves	Shutdown of pressurant control resulting in performance loss or oxidizer system overpressure and rupture	Electrical short, failure of electronics
	Lines	Transport flow of pressurant	Leakage or rupture	Performance loss; loss of booster	Structural failure, crack or fracture
Turbine Drive System	Control Box	Provides control of hydrocarbon (methane) flow	Does not provide correct control and sequencing to valves	Shutdown of hydrocarbon control resulting in performance loss	Electrical short, failure of electronics
	Tank	Store liquid hydrocarbon fuel for turbopumps or preburner	Leakage or rupture	Possible explosive rupture; performance loss; loss of booster	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Service Valve	Fill and drain access to tank	Rupture or internal leakage of valve seat	Performance loss; loss of booster	Structural failure, crack or fracture of valve body or seal failure

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Propulsion Turbine Drive System (Cont'd)	Isolation Valve	Allows flow to system	Stuck in wrong position	Inadequate flow of fuel	Electrical short, inadequate pyro charge
	Throttle Valve	Modulate fuel flow to turbopump preburner	Valve stuck in wrong position	Inadequate combustion in preburner, insufficient inlet gas to turbine inlet, performance loss	Insufficient lubricant, electromechanical failure of solenoid, crack or fracture of valve body
	Throttle Valve Position Indicator	Monitor valve position for controller	Inaccurate reading	Incorrect flow of of hydrocarbon resulting in performance loss	Electrical short
Turbopumps	Turbopump Assembly	Supply oxidizer to injector	Failure to rotate properly and pump oxidizer as required, internal leakage, external leakage	Thrust shutdown, fragmentation damage to adjacent components; performance loss; fire/explosion; loss of booster	Turbine rotor/blade, impeller or shaft fracture or fragment; bearing freeze-up or fracture; rotating parts rub housing; seal failure; manifold/housing crack or fracture
Preburner	Igniter	Provides spark ignition to start preburner	Fails to provide initiation energy to start combustion process	No gas produced, pumps don't operate, booster fails to start	Spark insufficient, structural failure, crack or fracture of igniter
	Case	Contains preburning gas generation reaction	Leakage or rupture	Fire or explosion, insufficient gas generation	Structural failure, crack or fracture

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Propulsion Oxidizer Feed System (Cont'd)	Tank	Store liquid oxidizer	Leakage or rupture	Possible explosive rupture; performance loss; loss of booster	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Service Valve	Fill and drain access to tank	Rupture or internal leakage of valve seat	Performance loss	Structural failure, crack or fracture of valve body or seal failure
	Relief Valve	Prevents overpressure of oxidizer system	Failure to open or close, seal leakage	Overpressure of oxidizer systems, possible tank rupture	Seal failure, structural failure, crack or fracture of valve body
	Isolation Valve	Allows oxidizer into system	Stuck in wrong position	Inadequate flow of pressurant	Electrical short, inadequate pyro charge
	Manifold	Transport oxidizer from 4 lines to circumferential injectors	Leakage or rupture	Possible explosive rupture; performance loss; loss of booster	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
Solid Fuel Gas Generator	Insulation	Provide thermal protection to surrounding structure	Burnthrough, debonding from case	Structural burnthrough, possible rupture or leakage	Inadequate bondline to case
	Case	Contain propellant and sustain pressures	Leakage or rupture	Booster structural failure, loss of vehicle; performance loss due to pressure leak	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Propulsion Solid Fuel Gas Generator (Cont'd)	Convergent Section Case	Contain propellant and sustain pressures	Leakage or rupture	Possible explosive rupture; performance loss	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Convergent Section Insulator	Provide thermal protection to surrounding structure	Burnthrough, debonding from case	Structural burnthrough, possible rupture or leakage	Inadequate bondline to case
	Injector	Contains, distributes, and atomizes propellants for proper mixing to produce efficient, stable combustion	Internal leakage	Performance loss, detonation/explosion	Structural failure, crack or fracture of injector passages due to material or manufacturing defect
	Combustion Chamber Case	Contain pressure and combustion process	Leakage or rupture	Performance loss; possible fire/ explosion	Structural failure, crack or fracture due to material or laminate defect, or damage in handling/transport
	Combustion Chamber Insulation	Provide thermal protection to surrounding structure	Burnthrough, debonding from case	Structural burnthrough, possible rupture or leakage	Inadequate bondline to case
	Fuel	Generate fuel-rich gas	Cracks, improper thrust trace, overpressure/explosion	Performance loss, overpressure resulting in potential case rupture	Improperly mixed, feedstock impurities, quality control

Table 27. Preliminary Failure Modes and Effects Analysis (Cont'd).

Assembly	Component	Function	Failure Mode	Effects	Possible Causes
Propulsion Solid Fuel Gas Generator (Cont'd)	Igniter	Light solid propellant grain	Failure to ignite, explosion	Booster does not ignite, fire	Structural failure, incorrect combustables formulation, electrical squib failure
Nozzle	TVC Actuators	Redirect thrust reactor in pitch and yaw	Incorrect positioning, stuck in wrong position	Vehicle becomes uncontrollable	Fluid leakage, structural damage, crack or failure of actuator case or attachments
	Thrust Structure	Contain pressure and constrict exhaust flow	Failure to contain or properly direct hot gases	Reduced performance	Rupture of wall; Buckling due to thermal, vibration, or gimbal acceleration forces
	Nozzle Structure	Contain pressure and smoothly expand exhaust flow	Failure to contain or properly direct hot gases	Reduced performance	Rupture of wall; buckling due to thermal, vibration, or gimbal acceleration forces
	Control Box	Provides control of nozzle actuators	Does not provide correct control and sequencing to valves	Shutdown of hydraulic control resulting in performance loss	Electrical short, failure of electronics

2.8 Technology Identification

ARC selected the pump-fed gas generator hybrid as our baseline concept. It offers advantages in safety, reliability, cost, and performance over the existing shuttle transport system (STS) solid rocket booster. The Phase 1 point design offers the following:

- Calculated reliability of 0.998.
- Reduced number of critical parts; only one cryogen (LOX).
- $11.4 billion life cycle cost.
- Engine shutdown and throttling capability.
- Mission accomplished even with loss of one pump.
- On-pad abort.
- 13,608 kilograms (30,000 pounds) (46 percent) shuttle payload improvement over ASRM boosters.
- Growth capability.

The gas generator hybrid proposed by ARC has several major technologies that have to be developed to demonstrate the concept, and several minor technologies that offer improvements (cost, reliability) to existing technology. The major technologies are listed below and discussed in the following sections. The major and minor technologies are listed in Table 28 and include the rationale for selection.

Major Technology	Priority
• Gas Generator Fuel Development	1
• Injector Design	1
• Combustion Interaction	2
• Combustor/Nozzle (Regenerative or Ablative)	2
Minor Technology	**Priority**
• Turbopump Development	3
• Tridyne Expulsion System Development	4
• Thrust Vector Control Using LOX	5
• Systems Integration	5

Table 28. Key Technology Issues.

Major Technology	Requirement	Rationale	Source
Gas Generator Fuel Development	• Environmentally Clean • Extinguishment • Ejection Efficiency • Ignition	• <1% HCl required • Required for pad abort • Required for fuel utilization goal • Provides turbopump spool-up and has to be extinguished	ARC will develop from our propellant database.
Injector Design	• Combustion Stability • Film Cooling • Pressure Feedback • Oxidizer Feedback	• Unchoked system could produce POGO effects • Improves reliability by reducing erosion • Required for fuel flow rate control • Design cannot allow oxidizer feedback into the gas generator	ARC will develop the design, consultants will support ARC.
Combustion Interaction	• Optimize Efficiency • Scale-Up • GG/Combustor Interface	• Lowers life cycle cost • No historical database • Determines extingishment and thrust termination	ARC will utilize ducted rocket and airbreathing technology experience.
Combustion Chamber and Nozzle	• Film Cooling • Low Erosion • Reduced Failure Modes • TVC Interface	• Provides improved reliability • Required to meet thrust-time trace • Monolithic construction, minimizes joints • Design has to include structural loads, mechanical installation constraints	ARC will fabricate preforms, densify forms, and assist in full-scale process development.
Oxidizer Delivery and Storage	• High Reliability • Low Cost • Driven from Main GG • Operate with High Solids	• Turbopump incorporates foil bearings • Turbopump is scaled from commercial line • Offers simplicity • Requires development of reverse pitot	Allied Signal will provide the design and hardware. ARC will provide the gas composition and pressure boundaries.

Table 28. Key Technology Issues (Cont'd).

Major Technology	Requirement	Rationale	Source
TVC	• Slew Rate	• Design constraints unavailable	ARC will provide piping and oxidizer interface; Boeing will provide requirements; Allied Signal will specify the design.
	• Frequency Response	• System capabilities not demonstrated	
	• Max Vector Angle	• Cost and performance estimates only	
	• Energy Source		
Propulsion System Integration	• Combustion Control	• Control interface has no historical data available	Boeing will provide the direction.
	• Launch Constraints	• Core vehicle constraints are unknown	
	• Attachments	• Attachment requirements are unknown	
	• Thrust Takeout Separation	• Booster separation and thrust termination effects on the core require investigation	
	• Guidance	• Active control tie into guidance is unknown	
	• Core Vehicle Integration		
Recovery System	• Turbomachinery Reuse	• Current database is insufficient to provide cost/reliability information	Boeing will provide the design. ARC will incorporate the design into the hardware. USBI will test the hardware.
	• Regenerative Cooled Components	• Regen.-cooled components offer some cost advantages	
	• Reuse	• Regen. components have higher nonrecurring cost which require reusable components	
	• Water Ingestion Seal System	• Water ingestion to protect pumps require demonstration	

2.8.1 Gas Generator Fuel Development

The fuel-rich propellants used in the hybrid booster should: (1) have burning rates of 0.76 to 1.27 centimeters/second (0.3 to 0.5 inches/second); (2) produce less than 1-percent hydrogen chloride emissions in the exhaust; (3) have high ejection efficiency; and (4) extinguish below 2.06 MPa (300 psia). These fuel-rich formulations are derived from both conventional propellants and fuel-rich formulations previously developed for air-breathing (ducted rocket and solid fuel ramjet) applications, but will need to be tailored and/or developed further to meet specific hybrid booster requirements.

Two promising gas generator formulations were evaluated under IR&D funding as discussed in Section 2.2. Both formulations were able to be extinguished, but their actual burning rates were too low for our baseline point designs. Both of these formulations will require tailoring to achieve the required burning rate. This tailoring must be performed experimentally to ensure that changes to improve one parameter (burning rate, for example) do not degrade another parameter (physical properties, for example). Further, following tailoring, the propellant/fuel must be fully characterized with regard to all of its properties including, but not limited to, reproducibility and reliability.

2.8.2 Injector Design

The injector in the gas generator hybrid is used to control the flow of the fuel-rich gas generator effluent, provide a location to inject the oxidizer, and to minimize uncontrolled feedback (instability) between the primary and secondary combustor. Because the gas generator operates unchoked when oxidizer is flowing, the pressure in the thrust chamber controls the burning rate of the gas generator and, therefore, its mass flow rate. In addition, the injector has to provide uniform mixing, film cooling of the combustor wall and damping of high-frequency oscillations. There are a number of critical development issues for the injector: (1) interaction of gas generator particulates (mixing, impingement, erosion); (2) gas generator/ injector interface temperature effects; (3) subsonic velocities/combustion feedback to produce thrust requirements; (4) oxidizer nucleate boiling; and (5) combustion instability. The development of the injector is critical to achieve high packing efficiency and high performance. An inefficient design will increase life cycle cost by lowering combustion efficiency, and reduce reliability because of increased combustor erosion. This has a high priority because of the historical problems associated with injector development.

2.8.3 Combustion Interaction

The gas generator effluent is important to the mixing and combustion processes in the thrust chamber. Incomplete mixing, nonuniform heat release, and short residence times have a direct impact on performance cost and reliability. The gas generator volume may have to be increased, excessive insulation added, combustor geometry reconfigured to incorporate flameholding or recirculation zones, and oxidizer delivery components increased in size to provide higher flow rates. Development is a high priority because it also impacts extinguishment and pad abort due to the feedback between the gas generator and combustor.

2.8.4 Combustor/Nozzle (Regenerative or Ablative)

The gas generator hybrid point design was evaluated using both a regeneratively cooled thrust chamber (combustor/nozzle) and an ablative (monolithic braided ablative) thrust chamber. The regeneratively cooled thrust chamber offers performance advantages by reducing component weight, reducing life cycle cost for a reusable system, and possibly improving reliability by reducing exhaust temperatures. The ablative thrust chamber offers improved reliability due to single-piece construction (no delaminations and simple design) and low life cycle cost due to inexpensive raw materials and automated processing. An ablative thrust chamber needs to be developed and/or demonstrated at the size and operating conditions for the hybrid since it is more cost effective for an expendable booster. The MBA approach will be investigated under the focused technology programs at MSFC for ALS boosters. This ALS program (Low-Cost, High-Reliability Cases, Insulation, and Nozzles for Large Solid Rocket Motors; NRA-89-MSFC-1) will complement the hybrid technology efforts, but because the exhaust environment for the hybrid is oxygen-rich, the development requirements will be different from ALS. Alternative fibers, variable component geometries, and case attachments will have to be developed.

2.8.5 Turbopump Development

ARC has selected foil-bearing turbopumps for the hybrid point design. Ball bearing LOX turbopumps have demonstrated poor durability and operating life was short and unpredictable in some programs. The poor reliability was primarily due to the premature failure of the ball bearings.[17,18,19,20,21] The life cycle cost and reliability objectives for the hybrid depend on the use of foil bearings. Foil bearings have accumulated approximately 510,000 hours of operation in small pump applications. This pump offers cost and reliability improvements compared to current ball bearing pumps, but a system sized for the hybrid requirements has not been developed or demonstrated.

2.8.6 Tridyne Expulsion System Development

The Tridyne system proposed for oxidizer expulsion was developed and demonstrated in subscale hardware by Aerojet but was never installed in an operating system. Tridyne consists of 0.91 moles helium, 0.06 moles hydrogen, and 0.03 moles of oxygen to form a nondetonable mixture that can be stored at high pressure. The energy is released by passing the mixture through a platinum catalyst bed. Gas temperatures are controlled by varying the reactant mixture. This Tridyne system offers cost and reliability improvements over cold gas or solid gas generator systems because of the lower volume requirements for the high-pressure helium and the reduced number of components.

2.8.7 Thrust Vector Control

Vectoring of the nozzle is common practice for solid rocket boosters. This method of thrust vector control adds weight and cost to the design. Fluid injection TVC was baselined in the point design because it raised the calculated predicted reliability from 0.987 to 0.995. An FITVC system using

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17. Gass, F. D., Alcock, J. F., and Flickinger, S. A., "Space Shuttle Main Engine - Alternate Turbopump Development Health Monitoring Program," AIAA-88-3411, 24th Joint Propulsion Conference, July 1988.

18. Hale, J. R., and Wood, B. Y., "Operational Life Improvement of SSME High-Pressure Turbopumps," paper presented at 36th International Astronautical Federation, Stockholm, Sweden, October 1985.

19. Childs, D. W., and Moyer, D. S., "Vibration Characteristics of the HPOTP of the SSME," paper presented at the 29th International Gas Turbine Conference, June 1984.

20. Merrimar, T. L., and Kannel, J. W., "Evaluation of EHD Film Thickness for Cryogenic Fluids," ASLE Preprint 85-AM-1F-1.

21. Duframe, D. D., and Kannel, J. W., "Evaluation of Shuttle Turbopump Bearings," NASA Contract Report CR-15096, November 1978.

LOX offers a simple design with low life cycle cost. If the system requirements defined during Phase 2 permit, a system with three degrees of deflection will be investigated as a means of improving the booster reliability and cost.

2.9 Acquisition Plan

2.9.1 Introduction

The Phase 2 Hybrid Propulsion Technology Program efforts are planned as a two-part, 33-month experimental and analytical study for the design, development, and investigation of critical components for the key technology issues affecting the gas generator hybrid with a pump-fed oxidizer delivery system. This propulsion system was selected because it offered the highest reliability and lowest life cycle cost in Phase 1 trade studies.

Part A, which will last 23 months, will consist of component development, fabrication, and demonstration; the goal of Part A will be to develop individual critical hybrid components consistent with the safety, reliability, and cost considerations determined in Phase 1 Technology Identification (see Section 2.0 of this report). Part B, which will last 10 months, will consist of component interactions, performance assessment, and system scale-up. This part will demonstrate interactions critical to achieving the safety, reliability, and cost goals for the booster system. It will also provide an assessment of the development risks that remain but are beyond the scope of Phase 2. A program schedule is presented in Figure 78. It shows major tasks to be performed in each phase and includes milestones. The four principal program elements are:

- Point Design Updates
- Part A Component Development, Fabrication, and Development
  - Propulsion System Development
    - Fuel Development and Characterization
    - Oxidizer Studies
    - Material Screening
    - Combustor/Nozzle Studies
    - TVC Design
  - Injector Design
    - Injector Studies

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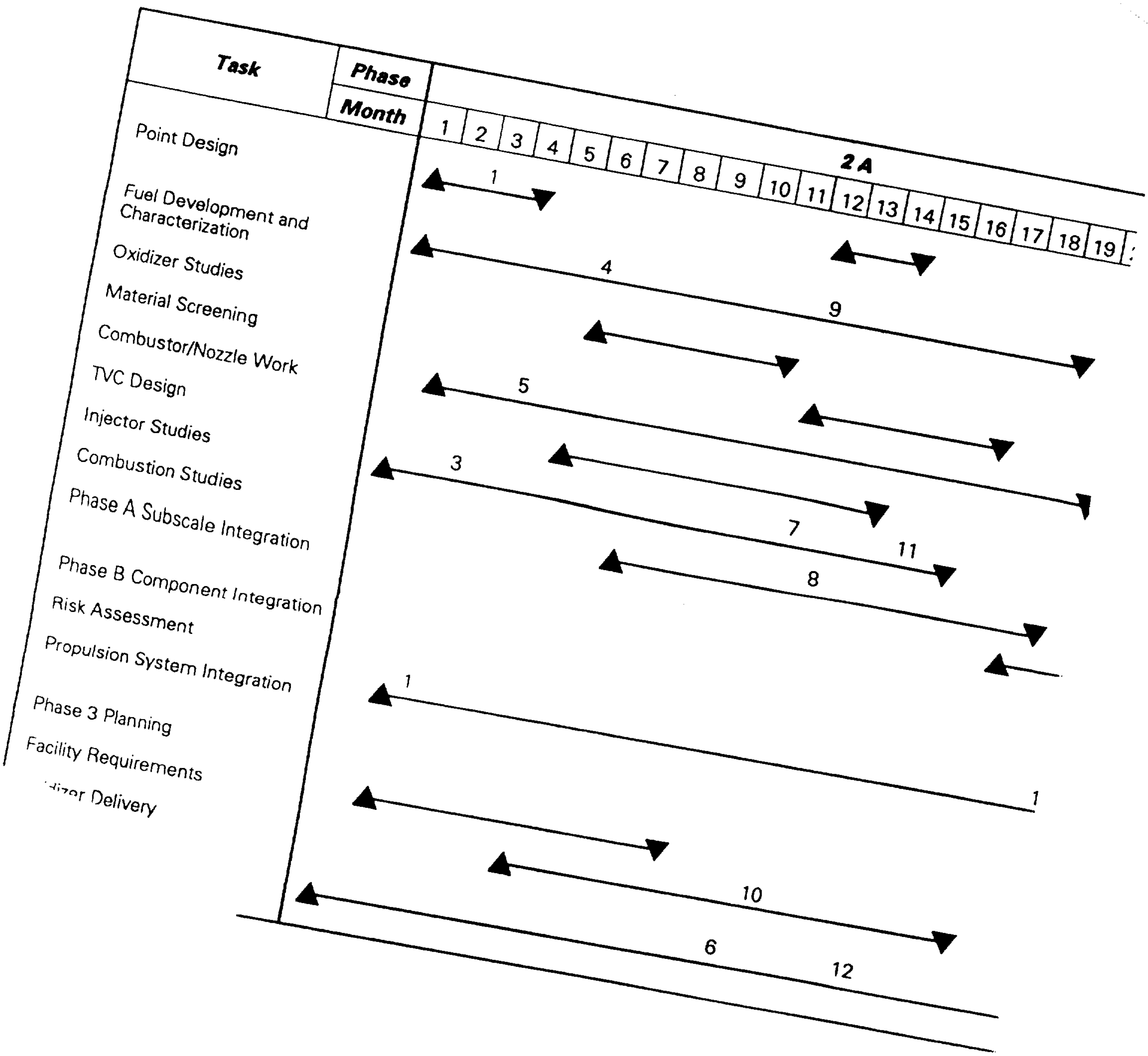

Programmatics

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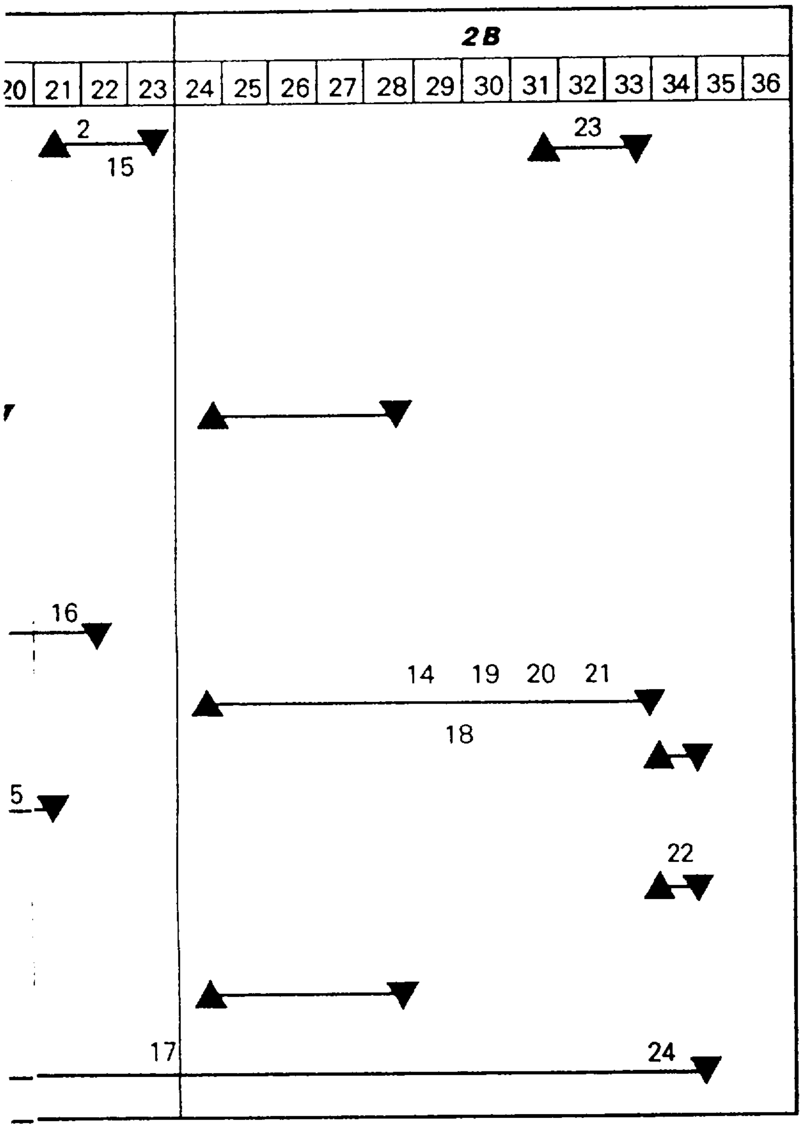

Phase II Milestones

1. Hardware integration with system contractor; detail point design.
   - LOX tank and auxiliaries
   - Injector design
   - Igniter design
   - Gas generator
   - TVC
   - Recovery system
   - Combustion chamber/nozzle design
2. Update design to incorporate fuel, combustion, and injector studies.
3. Fabricate subscale injectors.
4. Complete preliminary fuel development studies to measure efficiencies.
5. Fabricate subscale combustor nozzles.
6. Complete review with NASA of test plans.
7. Complete subscale injector tests.
8. Integrate subscale injector with gas generator and combustor/nozzle.
9. Finalize gas generator design.
10. Finalize oxidizer system design.
11. Complete injector design.
12. Interim review with MSFC to present results and future plans.
13. Complete changes as a result of MSFC review.
14. Complete fabrication of l00-k thrust motor hardware.
15. Integrate SE&I input into the point design.
16. Complete component testing.
17. Complete development of full-size oxidizer turbopump.
18. Complete assembly of l00-k motor, oxidizer delivery system and stand.
19. Demonstrate thrust termination.
20. Demonstrate gas generator extinguishment.
21. Demonstrate performance.
22. Complete hybrid system manufacturing plan.
23. Complete hybrid design.
24. Formal review with MSFC to present Phase II results, documentation, and Phase III program plan.

Figure 78. Program schedule.

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- Gas Generator/Combustion Chamber Interaction
  - Combustion Studies
  - Subscale Demonstration
- Oxidizer Delivery System
  - Oxidizer Delivery System Development

- Part B Component Interactions, Performance Assessment and System Scale-up
  - Component Integration
  - Risk Assessment
  - Phase 3 Planning
- Programmatics
  - Propulsion System Integration
  - Facility Requirements

These elements will be investigated in parallel efforts. The technical interaction between the experimental efforts will be emphasized. Interaction between the gas generator, injector, and combustion chamber will be tested as soon as practical. The testing of functional interaction between an active oxidizer system and the propulsion system will occur in Part B.

A program logic flow is presented in Figure 79. Direct and frequent MSFC involvement via formal and informal reviews is planned at all critical decision points.

Initially, the point design developed in Phase I will be updated. Second, fuel and oxidizer experimental investigations will be undertaken. Exploratory tests will be performed to identify areas requiring further definition. These tasks will be followed by more detailed characterization and definition of the injector plate and method and location of injecting LOX. Components will be investigated separately and then integrated with the gas generator to demonstrate capability at a subscale level, [88,964N (20,000 pounds) of thrust, 127 centimeters (50 inches) hardware].

Part B will integrate the key components into an overall system at a nominal 444,822N (100,000 pounds) of thrust, 190.5 centimeters (75 inches) hardware. Verification testing of integrated motors will be conducted to assess system performance. Ballistic and reliability analyses will be conducted for each test. Atlantic Research Corporation will incorporate a probabilistic reliability approach to verify the number of integrated tests

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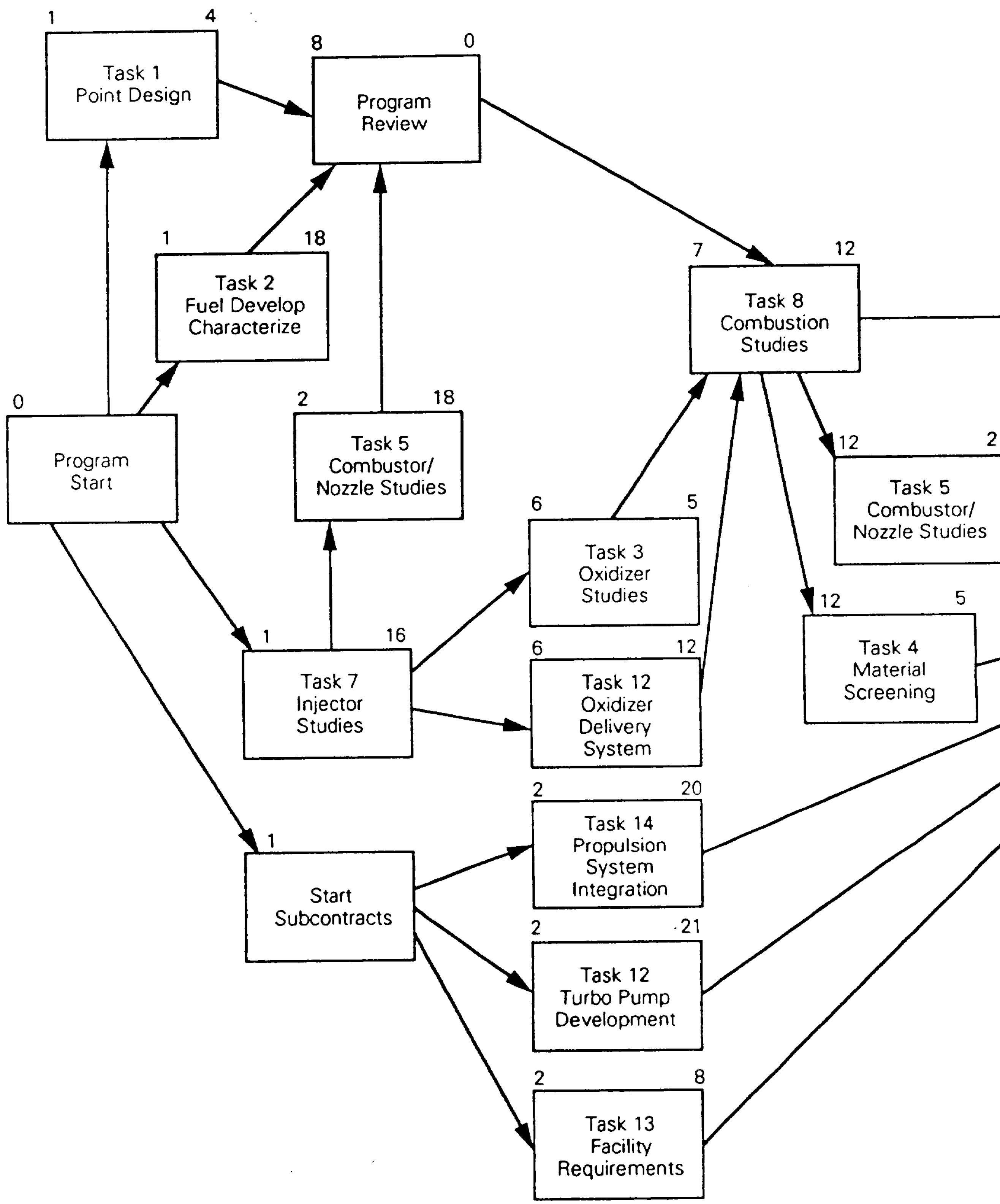

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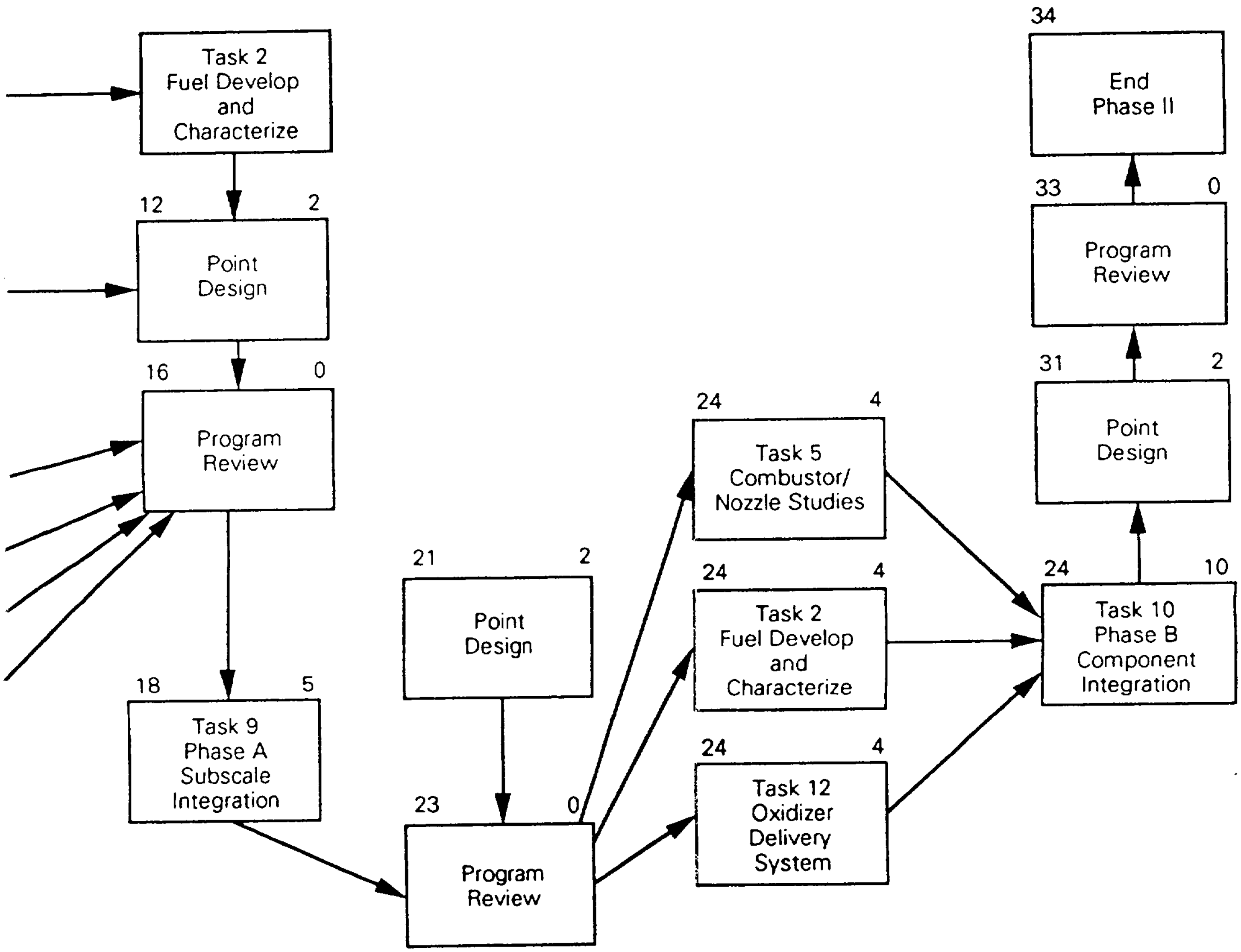

Figure 79. Program logic flow.

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required to demonstrate reliability. The test results will also be used to verify life cycle cost equations and results. Life cycle cost and reliability will be calculated as an integral part of the point design activities. The results will be presented to MSFC at each formal review.

Details on the work to be performed are discussed in the following sections. Table 29 summarizes each task that will be completed in Phase II.

2.9.2 Program Tasks

2.9.2.1 Point Design Updates (Task 1)

A POINT DESIGN WILL BE DEVELOPED FROM THE PHASE 1 RESULTS AND UPDATED AS COMPONENT DESIGN AND DEVELOPMENT MATURES AND AN OPERATING SYSTEM IS SELECTED.

A point design will be developed for the selected configuration. The point design will include system geometry, components, and materials of construction; weight breakdown; performance, cost, and reliability estimates;

Table 29. Phase II Task Summary.

Task	Title
1	Point Design Updates
2	Fuel Development and Characterization
3	Oxidizer Studies
4	Material Screening
5	Combustor/Nozzle Studies
6	TVC Design
7	Injector Studies
8	Combustion Studies
9	Subscale Demonstration
10	Oxidizer Delivery System Development
11	Component Integration
12	Risk Assessment
13	Phase 3 Planning
14	Propulsion System Integration
15	Facility Requirements

structural and thermal analysis; and specifications. The following design parameters will be evaluated relative to their impact on system parameters.

- Operating pressure
- Length-to-diameter ratio
- Oxidizer-to-fuel mixture ratio
- Structural requirements
- TVC requirements
- Start-up; shut-down; extinguishment requirements
- Expendable; reusable requirements

Operating pressure will be optimized by analyses to determine the weight/pressure/reliability/life cycle cost sensitivity of each system component. The individual sensitivities will be subjected to variational computation to maximize reliability and safety at its lowest attendant cost and weight.

Length-to-diameter trades will be made using the Boeing trajectory model, NTOP, to evaluate aerodynamic loading on the system to minimize cost and achieve the performance goals.

Oxidizer-to-fuel mixture ratio will be analytically evaluated to determine performance sensitivity during transients, impacts on turbopump design and operation, and booster size.

Booster structural stability will be analyzed to verify internal loading, stiffness, propellant grain, bond system, nozzle design and attachments, and thrust transfer to the case.

TVC design will be finalized based on MSFC requirements, trajectory analyses, and projected booster thrust mismatch. The requirements will define FITVC can be utilized.

Combustion modeling will be analytically optimized to determine the grain design, igniter, and burning rate exponent to meet the start-up and shut-down requirements. Gas generator extinguishment will be modeled to meet the on-pad abort requirement and flight thrust termination.

The design will be optimized by analyses to determine the cost sensitivity of a reusable system with an updated mission model. This will be evaluated using the updated design parameters against the expendable system.

The point design effort will be initiated immediately following contract award. This design will be updated three different times during the program and presented to MSFC. Life cycle cost and reliability calculations and updates will be an integral portion of the point design activities.

2.9.2.2 Fuel Development and Characterization (Task 2)

FUEL DEVELOPMENT AND PROPERTIES CHARACTERIZATION ARE ESSENTIAL TO FINAL CONFIGURATION DESIGN

In the gas generator hybrid, a fuel-rich propellant is burned in a primary combustor (gas generator). The fuel-rich exhaust is directed into a secondary combustor (thrust chamber) where it reacts with liquid oxidizer and expands through a nozzle to produce thrust.

ARC's baseline gas generator incorporates a fuel-rich propellant grain that has been formulated with equal molar amounts of AP and sodium nitrate to produce less than 1-percent HCl in the exhaust. The gas generator is required to extinguish below 2.06 MPa (300 psia), have high ejection efficiency, and a burning rate between 0.76 to 1.27 centimeters/second (0.3 to 0.5 inches/second).

The fuel grains to be used in the gas generator hybrid will be developed and evaluated in this task. The initial grains will be formulated utilizing the ingredients specified in Phase I.

Our primary approach is to develop a non-metallized fuel that meets the performance requirements, but does not create injector erosion and deposition problems. This formulation will be used to develop the injector configuration. To provide future growth potential, metallized fuels will be investigated as a secondary approach because they offer higher performance. Their impact on life cycle cost and reliability will determine if the development should continue.

The gas generator formulations for this program are derived from fuels that have been formulated and characterized for different applications, Table 30. The two variants selected are ARCADENE 399 and ARCADENE 246. ARCADENE 399 is a fuel-rich gas generator propellant developed for the Fixed Flow Ducted Rocket Program (Contract Number F33615-77-C-2057). This formulation gave good performance in full-scale ducted rocket flightweight hardware (DRPTV) in wind tunnel tests at AEDC. ARCADENE 246 is a conventional gas

Table 30. Gas Generator Fuels.

	Fuel Rich	Conventional
Binder %		
HTPB	22-26	21-30
CTPB	22-26	
PBAN		21-30
Solid Oxidizer %	27-37	60-70
Fuel %		
Polystyrene	30-45	
Poly (α-Methylstyrene)	34-40	
Poly (Methyl Methacrylate)	34-40	
σ_p %/°C	0.16 - 0.34	0.16 - 0.22
r_{1000} cm/sec	0.47 - 2.62	0.1 - 4.54
π_k %/C Motors	0.32 - 0.58	0.22 - 0.39
Catalyst %	0 - 3	0 - 3

generator propellant based on PBAN binder. ARCADENE 246 was developed for pressurization of the HARDROCK Silo Lid Door Opening Actuator and the MX Buried Trench Weapon System. This gas generator propellant provided a good history of reproducible burning rates and ejection efficiencies.

Beginning with the existing database, the formulations will be modified stepwise, changing one component at a time to evaluate the effects of each change/substitution required to achieve a suitable propellant for the hybrid application, see Table 31. The planned changes are listed as the first eight formulations in the table. These first eight changes consist of (1) alternative binders for evaluation of their impact on ballistic and ejection/residue properties, and (2) oxidizer modifications required to achieve a "clean" propellant (one which yields an exhaust free of all HCl).

Each grain formulation to be screened consists of one or more fuels; a binder which also serves as a fuel; an oxidizer necessary for primary combustion of the solid grain and subsequent ejection and expulsion of the fuel-rich species into the secondary combustion chamber; and catalysts necessary to modify the primary combustion ballistics such as burning rate. The preferred

Table 31. Compositions to be Screened.

Fixed Level Formulations for Reference to Database

Binder	Fuel	Additive	Oxidizer	
CTPB (HC-434)	PS	Fe_2O_3, CF_x/Al	AP	Baseline
HTPB (R-45M)	PS	Fe_2O_3, CF_x/Al	AP	Ref to 399 Database
HTPB (R-45 HT)	PS	Fe_2O_3, CF_x/Al	AP	Ref to 399 Database
PBAN	PS	Fe_2O_3, CF_x/Al	AP	Ref to 246 Database
PBAN	PS	Fe_2O_3, CF_x/Al	AP + $NaNO_3$	Clean Variant
HTPB (R-45M)	PS	Fe_2O_3, CF_x/Al	AP + $NaNO_3$	Clean Variant
HTPB (R-45M)	PS	Fe_2O_3, CF_x/Al	AN	Clean Variant

Formulation Variations For Screening[1]

		Additives (Catalysts)[2]	
Binder	Fuel	Fe_2O_3	Oxidizer[3]
HTPB (R-45M)	PS	X	AP + $NaNO_3$
HTPB (R-45HT)	PS	X	AP + $NaNO_3$
PBAN	PS	X	AP + $NaNO_3$
HTPB[4]	PS + Mg	X	AP
HTPB[4]	PS	X	AP + $NaNO_3$
HTPB[4]	PS + Al	X	AP + $NaNO_3$
HTPB[4]	Mg	X	AP
HTPB[4]	Mg		AN
HTPB[4]	Al	X	AP + $NaNO_3$

1. CF_x/Al will be evaluated at 0, 2, and 5 percent in selected candidate formulations for effect on secondary combustion.
2. Additive levels (not in combination): Fe_2O_3 = 0, 0.5, 1.0, 2.0 percent.
3. Oxidizer levels: 25, 30, 35, 40 percent.
4. R-45M or R-45HT based on previous results.

binder is HTPB (R-45HT or R-45M). Fuels include polystyrene (PS), Al powder, and Mg powder - the last of which also functions as a chlorine scavenger. The burning rate catalyst is Fe_2O_3, it is required to achieve adequate burning rate and ejection properties. CF_x/Al is also a catalyst, but has very little if any effect on primary combustion. Since it functions as a secondary combustion catalyst, it will be evaluated at low levels in selected formulations to determine if it enhances secondary combustion.

A limited evaluation of an ARCADENE 399 formulation was completed under corporate IR&D during 1989. This formulation consisted of 25 percent HTPB binder including 3 percent plasticizer; 34 percent polystyrene; 21.5 percent AP; 15.5 percent $NaNO_3$, 2 percent iron oxide, and 2 percent CF_x/Al. Pint mixes of the formulation were made and cast into cartons. Samples of the fuel were cut from the cartons and tested in a strand burner at six pressures [from 1.38 to 13.76 MPa (200 to 2,000 psia)]. The strands had a burning rate of 0.38 centimeters/second (0.15 inches/second) at a chamber pressure of 6.88 MPa (1,000 psia). Further, they would not burn below 3.44 MPa (500 psia).

A limited evaluation of an ARCADENE 246 formulation was also completed under corporate IR&D funding during 1989. This formulation consisted of 20.3 percent AP, 14.7 percent $NaNO_3$, 65 percent PBAN. The strands had a burning rate less than 0.25 centimeters/second (0.10 inches/second) at a chamber pressure of 6.88 MPa. Further, the strands would not burn below 3.44 MPa.

Both formulations will require burning rate tailoring to meet the requirements; however, both fuels can meet the extinguishment requirements.

Characterization of Fuel Properties - The initial step in characterizing each fuel formulation consists of the making a small mix and evaluating processing, ejection, residue type and amount, strand burning rates over a wide pressure range, and rapid pressure ($P_{d\ell}$) extinguishment. Formulations which show promise and have acceptable screening test results will be further characterized for ballistic and combustion properties including temperature sensitivity of burning rate, motor performance and ejection/expulsion efficiency, ignition and extinguishment properties, and mechanical properties. The effluent from the fuel generator will also be characterized for temperature and composition. Combustion characterization, extinguishment, and tensile testing will be conducted in parallel to quickly assess and select the most promising fuel formulation candidates.

Initial Screening - For the initial screening, a small mix will be made in the one-pint Baker Perkins mixer. If end-of-mix viscosity and processing characteristics are acceptable, the mix will be cast and cured. After cure, ambient ejection properties (in air), residue characteristics (at pressure under nitrogen), $P_{d\ell}$ extinguishment characteristics, and strand burning rates in duplicate at seven pressures [from 1.38 to 13.76 MPa (200 to 2,000 psi)] will be determined. Promising formulations will be further characterized from larger (1-gallon) mixes. Sensitivity tests [impact, friction, electrostatic discharge (ESD), and DSC] will be conducted on formulations containing new ingredients or new combinations of ingredients to establish potential hazard level.

Combustion and Ballistic Characterization - Formulations which have acceptable processing characteristics, and for which ejection, residue, and burning rate properties are deemed adequate, will be subjected to further testing for combustion properties. Nominal 4.5-kilogram (10-pound) grains in 15.2 centimeter (6 inch) diameter 6C4-11.2 Rohm and Haas hardware will be used to determine motor performance including C* efficiency, burning rate, and motor expulsion or ejection efficiency. An eroding nozzle throat will be used in these firings, and these data will be used in our ballistic computer routines to determine burning rate and pressure exponent over the pressure range of the firing.

Selected candidate formulations will also be cast into 7.6 and 22.9 centimeter (3 and 9 inch) diameter cartridges to produce center-perforated grains 3 to 11 kilograms in weight for later testing in heavywall hardware. The grain configurations tested will be designed to produce the higher mass-flow rates required to verify and scale the results from the 4.5 kilogram motor firings. A total of 52 7.6-centimeter grains and 21 22.9-centimeter grains will be cast for Task 4 testing.

Extinguishment - In parallel with the combustion and ballistic characterization studies, the effects of compositional variations on extinguishment boundaries will be established. The 4.5 kilogram Rohm and Haas hardware with a regressive grain design will be used to verify the $P_{d\ell}$ screening test results. The nozzle throats will be sized to generate an initial pressure level at which the fuel burns well, with subsequent decrease in pressure with time due to the regressive surface area, until the grain extinguishes. The pressure decay rate will not be sufficient to determine dp/dt extinguishment, but the results shall be correlated to $P_{d\ell}$ measurements. Confirmation tests will be repeated later in 22.9-centimeter hardware on selected candidates.

Ignition - Fuel-rich, gas generator propellants exhibit more marginal combustion characteristics than conventional propellants due to their oxidizer deficiency. The fuels tend to be more difficult and slower to ignite. A relatively long-acting pyrogen will probably be required for this system.

As part of the subscale testing, ARC will define the igniter characteristics required (flow rate, duration, product composition). The results will be used as inputs for calibration of a modified version of the Caveny-Kuo[20] ignition model to predict the requirements for larger gas generators. Confirmation tests of the 15.2-centimeter motor results will be made in 22.9-centimeter hardware to fine-tune the model for the full-scale definition.

Physical Property Testing - JANNAF Class C tensile tests (triplicate specimens, one strain rate, three temperatures) will be conducted on selected formulations. Based on these results, tailoring will be performed to improve and optimize tensile properties. Final candidates will be more extensively characterized (triplicate specimens, four strain rates, seven temperatures). Additional characterization of final candidates will include use of the RMS-4 for dynamic mechanical properties and gel time, and the Haake viscometer for rheological properties. Glass transition temperature, coefficient of thermal expansion, and DSC and TGA thermal profiles will also be determined on final candidates.

Bondline properties between promising fuel formulations and candidate liner and insulation materials will also be evaluated using bond-in-tension, double-lap shear, and peel boat specimens.

Effluent Characterization - The effluent from the gas generator will be characterized to provide information on (1) temperature; (2) gas composition; and (3) nature and size of condensed species.

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20. A. Pertz, L. H. Caveny, K. K. Kuo, M. Summerfield, "The Starting Transient of Solid Propellant Rocket Motors with High Internal Gas Velocities," NASA Grant NGL 31-001-109, Aerospace and Mechanical Sciences Report No. 1100, Princeton, April 13, 1989.

For characterization of the effluent, we will use isokinetic sampling of an unchoked stream (produced by firing into a pressurized tank with controlled venting) at several gas generator pressures. Gaseous products will be analyzed by standard laboratory techniques. Gas temperatures will be measured using embedded thermocouples and radiometer measurements.

Collected particulates will be sized, and the fractions will be chemically analyzed to determine composition. The results will be used to define the injector requirements.

2.9.2.3 Oxidizer Studies (Task 3)

OXIDIZER STUDIES DEFINE THE HEAT TRANSFER COEFFICIENTS FOR THE DESIGN OF A REGENERATIVELY COOLED COMBUSTOR AND NOZZLE.

The point design developed in Phase 1 includes the option of using a regeneratively cooled thruster (combustor/nozzle) with LOX to improve life cycle cost. The design is complicated because of the oxidizer throttling required to meet the prescribed regressive thrust trace. At the lower oxidizer flow rates, film boiling of the LOX may occur.

Oxidizer studies will generate the necessary data to determine if LOX can be used as the cooling fluid for the regeneratively cooled combustor and nozzle. This will be performed if the regeneratively cooled nozzle is selected for development.

Heat Transfer Measurements - Benefits for the system may be achieved with the use of LOX as the coolant for a regenerative nozzle. The Phase 1 point design is based on chamber pressure variations from 8.95 MPa (1,300 psia) to 4.65 MPa (675 psia) to meet the required thrust-time trace. At these pressures, the LOX would still be liquid; however, at slightly lower pressure, the LOX will begin to film boil. Heat transfer coefficients must be determined experimentally to determine if it is still possible to cool the combustor and nozzle.

A flow reactor will be designed, fabricated, and instrumented with thermocouples and pressure transducers. LOX will be flowed at various rates through a heated furnace to simulate the combustor temperatures. The temperature of the LOX will be measured at several flow rates and pressures to determine the thermal coefficients during boiling.

2.9.2.4 Material Screening (Task 4)

CRITICAL MATERIALS FOR COMPONENT TECHNOLOGY DEMONSTRATION WILL BE SCREENED.

Before integration tests are performed, the ability of the insulation, combustor and nozzle, and composites to function in the gas generator/combustor environment must be verified. The critical environments within the gas generator and combustor range from strongly oxidizing to strongly reducing. The oxidizer tank materials experience cryogenic temperatures that can result in embrittlement of the epoxy or polyimide resin system.

Screening of nozzle and oxidizer tank composites will consist of preparing test specimens of the systems considered. Nozzle specimens will be manufactured in an 20.3 centimeter (8 inch) square mold. Two PAN fibers (Hercules AS4 and Amoco T650-35) and two quartz fibers (J. P. Stevens Astroquartz and FMI High Purity Quartz) will be investigated using three different phenolic resin systems. Five duplicate specimens of each material system will be tested for thermal erosion, tensile, compression, impact and shear. Thermal erosion will be tested by subjecting the samples to the hybrid motor exhaust.

Oxidizer tank specimens will be manufactured into 30.5 centimeter (12 inch) square sheets. Three intermediate-modulus fibers (Amoco T650/42, Apollo 43-750, Hercules IM-7) will be investigated using epoxy and polyimide resins. The sheets will be cut into 15.2-centimeter (6-inch) squares. Five duplicate specimens of each material system will be tested for tensile, compression, impact, shear and chemical stability at ambient (298K) and 78K.

Insulation materials that will be tested include ARCTIP (HTPB with glass microballoon fillers), and Kevlar-filled EPDM. Test specimens of the elastomer candidates will be installed at the exhaust end of the 15.2 centimeter insulation screening motor. Dimensions will be measured before and after the test.

Thermal properties of the materials must be established to verify analytical results. Heat capacity and thermal diffusivity measurements will be made, and the results will be incorporated into the point design.

2.9.2.5 Combustor/Nozzle Studies (Task 5)

A MONOLITHIC BRAIDED ABLATIVE THRUST CHAMBER WILL BE DEVELOPED AND DEMONSTRATED.

ARC's hybrid fuel booster incorporates a monolithic braided ablative (MBA) thrust chamber. The MBA is an integral combustion chamber, nozzle, and extension cone. It consists of a three-dimensional (3D) braided architecture in resin matrix. This one-piece design requires no secondary structures, insulators, or complex assembly of flame-surface ablative components. The MBA thrust chamber achieves high reliability by eliminating failure modes due to joints and leak paths, secondary bonds, and delamination/ply-lifting associated with conventional, two-dimensionally laminated ablative components. The design of the combustor/nozzle for the hybrid incorporates quartz fiber in a phenolic resin selected to minimize the effects of the oxidizing environment.

Development of the MBA thrust chamber will proceed in a stepwise manner through 3D-braided quartz-phenolic material properties testing, reliability development and design/process validation via seven subscale engine test bed firings. Thrust chamber design and analysis methodology, manufacturing processes, and product evaluation techniques will be developed and matured concurrently during the course of this program. The full-scale MBA design, manufacturing process, and evaluation techniques will be refined at specific points in the program to reflect increased understanding of the thrust chamber and booster requirements. Verification/demonstration of the MBA thrust chamber will occur in an integrated subscale firing at the end of Task II.

Material Properties Analysis and Testing

ARC will conduct design, analysis, and testing activities to establish an initial database for the quartz-phenolic MBA material properties. MBA materials physical, mechanical, thermal and erosion properties will be determined to support subscale thrust chamber design and refinement of the full-scale point design. The following tasks will be performed:

- Micromechanics modelling
- Test plan definition
- Physical, mechanical and thermal properties testing and evaluation
- Subscale erosion testing

A micromechanics model will be developed to describe the various braided fiber architectures available. Using this model in conjunction with published properties for the selected quartz fiber and phenolic resin, MBA mechanical properties will be predicted. These properties will guide the selection of the most appropriate braided architecture.

Physical, thermal, and mechanical testing of the MBA material will allow creation of a preliminary material properties database, validation of the micromechanics model prediction, and assessment of the effects of process variables on material properties. A test matrix is presented in Table 32.

A 22.9 centimeter (9 inch) diameter hybrid test motor will be used to conduct laboratory screening of an axial series of cylindrical specimens from the process parameter variation study. Measured erosion data will be input to the aerothermal analysis models for correlation with predictions based on measured thermal properties.

Subscale Component Testing and Evaluation - Four subscale MBA thrust chambers will be fabricated in an iterative design-fabrication-evaluation sequence as part of a reliability development test series. Each successive subscale thrust chamber design will be refined based on the preceding motor test firing evaluations.

Six additional subscale MBA thrust chambers will then be fabricated as part of a design/process validation test series. The fabrication procedures will be frozen according to the subscale thrust chamber process specification so that overall repeatability of the process and performance can be evaluated.

Table 32. MBA Material Characterization Preliminary Test Matrix.

Tests	294K	2200K	3033K
Hoop Compression	3	3	3
Meridional Compression	3	3	3
Hoop Tension	3	3	3
Meridional Tension	3	3	3
Axial Shear	3	3	3
Radial Shear	3	3	3
Hoop Thermal Expansion	3	3	3
Axial Thermal Expansion	3	3	3
Radial Thermal Expansion	3	3	3
Meridional Conductivity	3	3	3
Radial Thermal Conductivity	3	3	3
Specific Heat	3	3	3

All ten thrust chambers will be instrumented with thermocouples and strain gages during the test firing. Thrust, pressure, strain gage, and thermocouple data, along with post-test hardware, will be analyzed following each test. Pre- and post-test computed tomography (CT) inspection will exhibit surface recession and char depth profiles for each test article. Dissection of each article, as appropriate, will aid in verification of CT evaluations and yield signs of anomalous performance or incipient failure if any exist. Updating of the full-scale design will occur during the subscale development phase as illustrated in Figure 80.

2.9.2.6 Thrust Vector Control Design (Task 6)

A TVC DESIGN WILL BE DEVELOPED USING LOX AS THE INJECTANT AND INCORPORATED INTO THE POINT DESIGN.

Several designs for a fluid injection thrust vector control system (FITVC) were investigated by Allied Signal in Phase I, Technology Identification. They determined that LOX was the most feasible because it provided a fairly simple, reliable design. Definition of the total duty requirements, and thus propellant usage, is crucial to making a final feasibility decision.

In this task, Allied Signal, under subcontract to ARC, will perform design studies of an FITVC system. The final design selected will provide the optimum combination of weight, development risk, complexity, and cost. Changes in the system pressure, number of control thrusters, and redundancy will be studied and evaluated on the basis of cost, reliability, and complexity.

Allied Signal will assemble the hardware, fabricate thrusters, and develop the electronic controller for the system. A prototype will be tested on the 100,000 pound thrust test motor in Task II to verify the design.

2.9.2.7 Injector Studies (Task 7)

INJECTOR PRESSURE DROP AND SPRAY PATTERN MUST BE OPTIMIZED TO PROVIDE FILM COOLING AND REQUIRED MIXING.

The gas generator hybrid point design includes a multi-port injector that separates the gas generator from the secondary combustion chamber (thrust chamber). Fuel-rich combustion products pass into the combustor via injector ports. Flow through the ports is subsonic at normal operating pressures

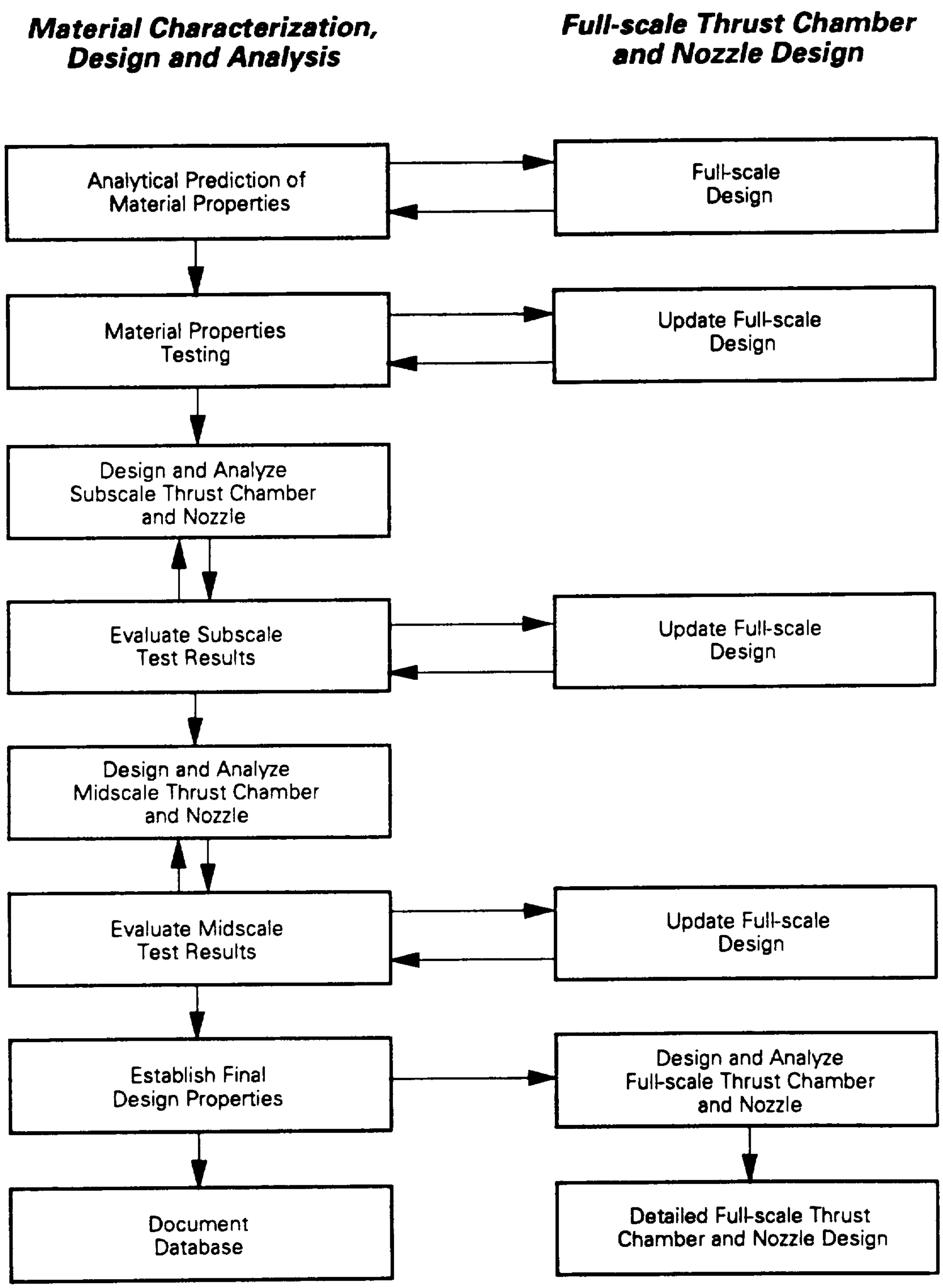


Figure 80. Full-scale nozzle design activity interacts with material characterization, design, technology efforts.

resulting in an unchoked injector; pressure changes occurring in the combustor are communicated to the gas generator. The fuel flow rate is controlled and modulated by adjusting the chamber pressure. This is accomplished by changing the LOX flow rate.

This task involves the evaluation of candidate plate injector designs. The tests will be conducted at ARC, where a separate facility will be set up for storage and flow control to conduct simulated cold flow studies using liquid nitrogen and liquid oxygen. This oxidizer facility will be integrated with our standard static test facility so simulated exhaust gases with entrained particulates can be used in flow studies.

A total of 75 injector tests are planned. We will initially evaluate the injection variables shown in Table 33 individually and in combination with water and liquid nitrogen to minimize test cost. In addition, since the combustion effluent from the gas generator will contain from 30- to 40-percent carbon particulates, we will also evaluate the variables using a hot, simulated gas (compressed air) that has been entrained with carbon using a metered injection system. The final tests will be conducted using liquid oxygen to verify the results.

A pressure-fed oxidizer delivery system will be used to minimize system fluctuations. Data from the tests will include tank pressure and temperature, oxidizer flow rate, high-speed movies, still photography, and pitot traverses to measure stagnant mixing zones. As part of this task, ARC will utilize combustion consultants to assist in the development of our injector study

Table 33. Injector Variables.

Variable	Number of Tests Per Series
Design	
Oxidizer/Fuel Flow Area Ratio	8
Swirl (Inlet Angle)	6
Impingement (Impact Angle)	6
Shape	
Circular Pattern (Orifice Pattern)	6
Diamond Pattern	6
Size	
Diameter (Orifice Diameter)	6
Length (Orifice Slot Length)	4
Angle of Injection	6

matrix. The consultants will provide expertise with acoustic cavities, baffles, injector posts, and 3-dimensional flowfield modeling.

Injector plate modules (zone of 1-fuel and 1-oxidizer injector) will be tested and verified in the 22.9 centimeter (9 inch) diameter hardware tests. We will run cold-flow studies and then verify the results in the 22.9-centimeter hardware (approximately 27 tests). This iteration will produce an injector for the 127 centimeter (50 inch) diameter subscale and 190.5-centimeter (75-inch) component integration tests.

2.9.2.8 Combustion Studies (Task 8)

CHARACTERIZATION OF THE GAS GENERATOR PRODUCTS AND THEIR INTERACTION WITH THE OXIDIZER IN THE COMBUSTOR IS ESSENTIAL TO THE DESIGN OF THE INJECTOR AND OPTIMIZATION OF COMBUSTION EFFICIENCY.

The nature of the gas generator effluent is important to the mixing and combustion processes in the secondary combustor. Jets of effluent and oxidizer must mix completely (down to a molecular scale) and burn for full utilization of the thermodynamic potential of the fuel and oxidizer. It is expected that combustion of the heavily laden particulate fuel species will require good mixing and sufficient residence time for relatively slow particle combustion (limited by microdiffusion of oxidizing species to the particle surface). In addition, in some cases it is important that the particles be exposed to hot environments for longer periods for ignition which requires controlled recirculation. The flow patterns will be such as to avoid nonuniform heat release patterns.

Combustor Modeling - ARC will use existing three-dimensional computational fluid dynamic (CFD) codes (offshoots of the TEACH code) to develop combustor geometries to be tested.[21,22,23] As we test, the results will be

21. S. P. Vanka, J. L. Krazinski, A. S. Nejad, "Efficient Computational Tool for Ramjet Combustor Research," AIAA 26th Aerospace Sciences Meeting, Reno, Nevada, January 1988.

22. S. P. Vanka, "Computations of Turbulent Recirculating Flows with Fully Coupled Solution of Momentum and Continuity Equations," Report, Wright Patterson Air Force Base, ANL-83-74.

23. D. G. Lilley, D. L. Rhode, "Computer Code for Swirling Turbulent Axisymmetric Recirculating Flows in Practical Isothermal Combustor Geometries," NASA Contract Report 3442.

fed back into the code to improve the modeling capability. The updated codes will provide scaling information for our larger test configurations, and will be used to predict our full-scale results.

Experimental Test Section - ARC will modify our existing 7.6 and 22.9 centimeter test motors to be compatible with the gas generator, see Figure 81. The test motors are flanged heavywall construction that can be assembled into different configurations. The hardware will function as follows:

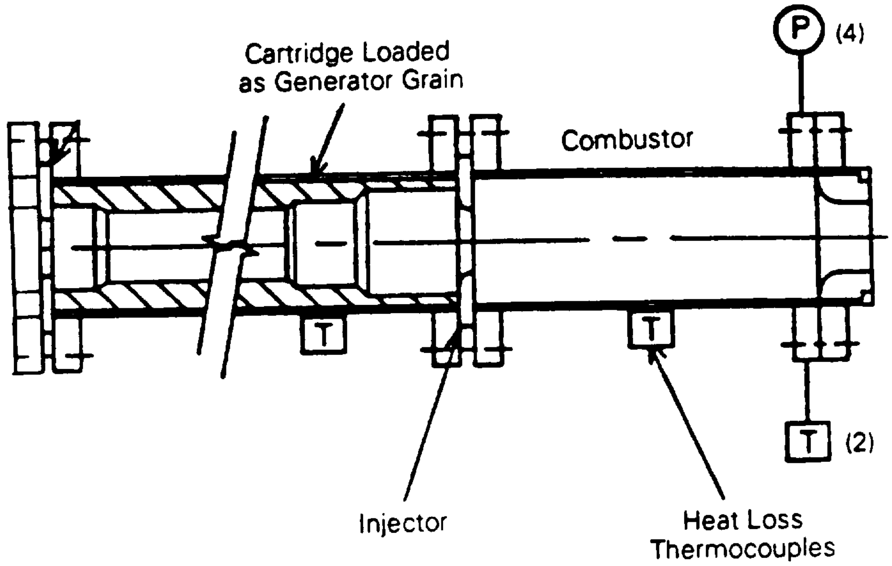
- A gas generator will be assembled into a long spool piece with the igniter mounted in the head flange.
- The exhaust gases will be directed into an insulated test section and through a flow control orifice.
- Oxidizer will be injected into the hot combustion gases.
- Combustion products will exit the cavity through a nozzle insert.
- Cavity pressure will be monitored.
- Oxidizer flow rate will be monitored using a mass flow meter.

Test Matrix - Exploratory testing will be used to assess the characteristics of the candidate fuels. We have outlined seven test series, Table 34, with the 7.6-centimeter hardware and five test series with the 22.9-centimeter hardware to verify performance and extinguishment. Gas generator development and propellant grain manufacture will be performed in parallel in Task 2.

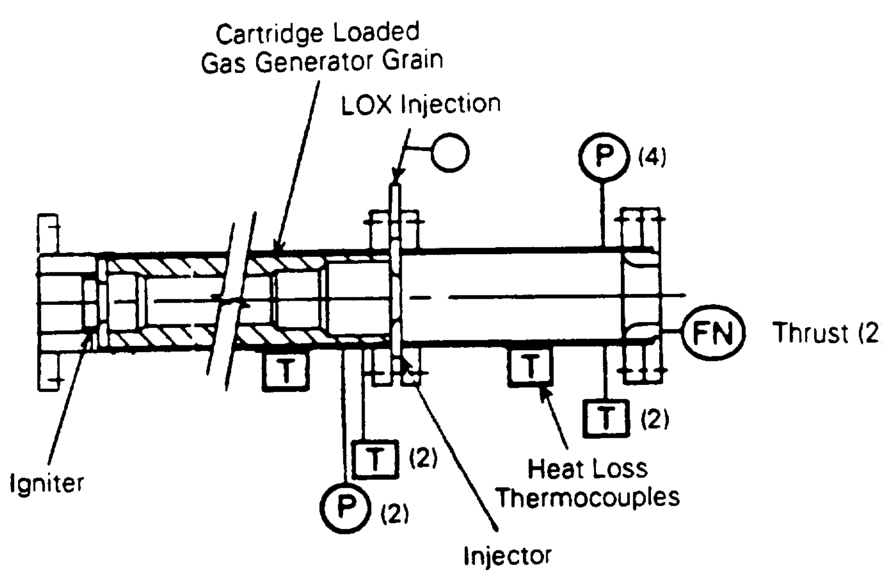
The first series of tests will establish a relationship between the 7.6-centimeter tests and prior IR&D tests. It will also validate the test hardware, data acquisition, and operating procedures. Series 2 and 3 will provide data on alternative fuels and demonstrate the effects of additives, and different fuels on combustion behavior.

Series 4 will evaluate combustion extinguishment as the oxidizer flow rate is shut off. This series will incorporate the optimum fuel formulations developed in Task 2. Series 5 will evaluate combustor geometry with the fuels used in Series 4. Series 6 will evaluate the injector designs developed in Task 5.

Series 7 will complete the 7.6-centimeter hardware tests. The series will integrate the optimum fuel formulation from Series 1, 2, and 3 with the injector and combustor from Series 5 and 6.



Typical 7.6 cm (3 in) Test Set-up



Typical 22.9 cm (9 in) Test Set-up

Figure 81. Test motor hardware.

Table 34. 7.6- and 22.9-Centimeter Diameter Test Matrix.

Series	Objective	Tests
1	Relate IR&D activities. Validate hardware, instrumentation. Measure combustion efficiency.	6
2	Relate combustion to fuel type. Relate combustion to additive content.	10
3	Relate combustion to solid oxidizer content. Relate combustion to oxidizer particle size.	10
4	Relate fuel formulation to extinguishment.	6
5	Relate stay time to combustion efficiency.	4
6	Measure combustion efficiency versus injector pressure drop. Measure combustion efficiency versus oxidizer spray pattern.	6
7	Measure combustion efficiency, thrust at fixed O/F ratio.	6
8	Duplicate Series 7 with 22.9-centimeter hardware. Validate 7.6-centimeter hardware results.	3
9	Measure combustion efficiency, thrust versus gas generator grain design.	
10	Verify gas generator extinguishment.	6
11	Measure efficiency and thrust at three different O/F ratios; run duplicate tests.	6
12	Measure efficiency and thrust at programmed O/F ratio to verify repeatability.	6

An additional five-test series with 22.9-centimeter diameter hardware will be performed at the conclusion of the 7.6-centimeter diameter tests. The first series (No. 8) will verify scalability of the results of the 7.6-centimeter tests. The remainder of the 22.9-centimeter diameter tests will provide scaleup data for the combustion modeling, verify extinguishment of the gas generator with termination of oxidizers flow, and verification of oxidizer-to-fuel ratio by measuring the resultant motor thrust. The data will be used to update the point design.

The final testing in the 22.9-centimeter hardware will involve repeatability/stability of the combustion process. We will use high-frequency-response pressure instrumentation to identify combustion instability between the gas generator and combustor. Pulser testing will be investigated to define stability margins. Since natural frequency concerns are dependent on scale, short-duration full-scale tests may be required to ensure that the system has an adequate safety margin.

2.9.2.9 Subscale Demonstration (Task 9)

ARC WILL UTILIZE A 127 CENTIMETER DIAMETER, 88,964N-THRUST SUBSCALE TEST MOTOR TO EVALUATE THE GAS GENERATOR/COMBUSTOR PERFORMANCE.

The objectives of the 127 centimeter (50 inch) subscale tests are to:

- Establish baseline performance.
- Evaluate the injector.
- Demonstrate combustion stability.
- Demonstrate extinguishment.

A detailed test plan will be written at the start of the task. The plan will identify the tests to be run and their objectives, facilities, procedures, updated schedule and data acquisition, and analysis procedures.

A 50-inch diameter gas generator is required for the 88,964N (20,000 pound)-thrust subscale demonstration tests to provide sufficient mass flow rates and thrust to verify scaling. The demonstration design will utilize a cartridge-loaded heavywall steel motor case flanged to provide geometry flexibility, Figure 82. ARC engineers will perform detailed structural and thermal

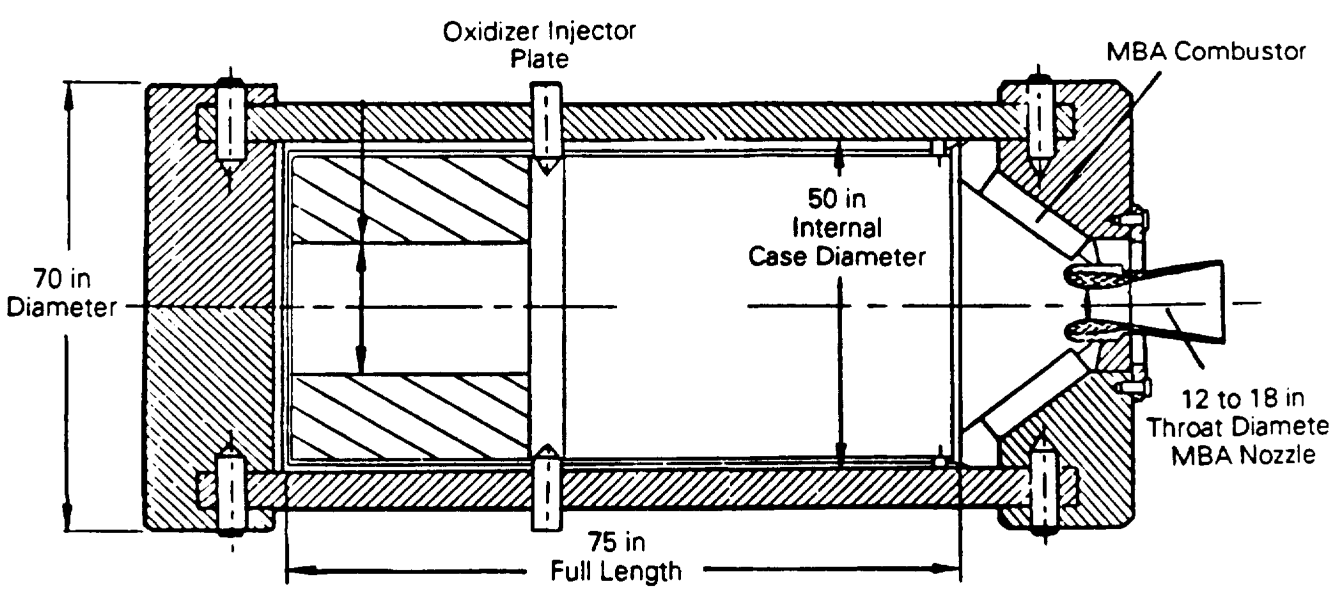


Figure 82. 50 in heavywall subscale demonstrator motor.

analyses, and use computer aided design and manufacturing (CADAM) software to document the hardware drawings for component fabrication. The fuel grains for the gas generator will be cast and cured in 102 centimeter (40 inch) phenolic sleeves. Each loaded sleeve will be inspected using x-ray and ultrasonic NDE prior to being insulated and loaded into the hardware for testing.

A cartridge-loaded MBA combustor/nozzle insert will be braided, inspected, and densified at ARC. The fiber selected for the preforms will be specified in Task 7.

The motor will be mounted horizontally for testing. Oxidizer will be supplied to the motor using a pad-mounted delivery system. Measurements will include axial thrust chamber pressures, oxidizer flow rate, inlet oxidizer pressure and temperature, and gas generator case strain measurements.

The matrix of planned tests with the subscale integration test motors is summarized in Table 35. The tests will demonstrate the scaleup of the gas generator, injector, and combustor/nozzle, all of which are critical components for Part B, Component Integration tests. Initially, only the gas generator will be tested to verify its performance. The injector will be replaced with a regressive nozzle. The fuel mass flow rate will be measured and compared to the required rates. The first series of subscale demonstration tests will evaluate the injector assembly. These six tests will evaluate the two most promising injectors from Task 5.

Series 2 evaluates the change in performance (pressure, regression rate, thrust) as a function of variable oxidizer flow rate. Series 3 will demonstrate gas generator extinguishment. In this series, on-pad abort and thrust termination will be simulated with oxidizer flow control. Series 4 will demonstrate the required L* (residence time) to provide the required fuel utilization. If necessary, additional tests will be added if secondary mixing

Table 35. Subscale Demonstration Test Matrix.

Series	Variable	Tests
0	Gas Generator Only	2
1	Injector	6
2	Oxidizer Flow Rate	4
3	Extinguishment	3
4	Combustor Geometry	4

is required for particulate combustion. A summary of the oxidizer and fuel flow rates for the 127 centimeter (50 inch) diameter test motor are summarized in Table 36.

2.9.2.10 Oxidizer Delivery System Development (Task 10)

ARC/LIQUID PROPULSION AND ALLIED-SIGNAL WILL PROVIDE AN OXIDIZER TANK, OXIDIZER TURBOPUMP, AND FEED SYSTEM DESIGN TO SUPPORT THE COMPONENT INTEGRATION AND POINT DESIGN TASKS.

A turbopump oxidizer system was selected for the Phase 1 point design. The turbopumps are powered by the gas generator and are required to operate over a wide throttling range. Since the gas generator exhaust contains solid particulates, an inertial filter arrangement incorporating a reverse pitot is used to provide clean fluid. The efficiency of the pumps is maximized by supplying a constant head pressure to the pump inlets. This head pressure is developed by reacting Tridyne, a mixture of helium, oxygen, and hydrogen to produce a 667°K (1,200°R) expulsion gas to pressurize the LOX tank.

We will perform design studies for the integration and specification of the oxidizer delivery system and controls to support the overall point design and component development studies. The delivery system design will incorporate the combustion results from Tasks 2 and 4.

We will also evaluate the Tridyne helium delivery system for the expulsion of LOX. The evaluation will include the selection of catalysts; optimum ratio of hydrogen, oxygen, and helium to provide the required temperatures; and the fabrication of the helium storage tank.

Allied-Signal, under subcontract to ARC, will perform design studies of the turbopump to support the overall design effort. The turbopump will be developed, built and tested within the first 23 months. The remaining 10 months in Phase 2 will be used to procure the long-lead hardware and finalize the design required for Phase 3.

A turbopump design will be developed based on the final definition of the duty cycle. Stress, aero and thermal analyses, bearing design, critical speed and rotor dynamic response calculations and a material study will be performed. Transient analyses will also be performed to verify the component performance. The design will be documented using computer aided design techniques.

Table 36. Subscale Demonstration Test Summary, Task 9.

Test Series	Test Objective	Test Conditions	Test Duration	Variables Measured
0 - Full Duration GG Test (2 Tests)	Test GG Parameters Verify Performance Full Duration Test	Monitor Flow Rate	50 sec	GG Chamber Pressure GG Mass Flow Rate
1 - Full Up Motor Test (6 Tests)	Injector Evaluation	Programmed Oxidizer and Fuel Flows 88,964 N Thrust	30 sec	GG Pressure and Flow Rate LOX Total Flow Rate LOX Flow Rate to Combustion Chamber Total Thrust Pressure Drop Across Chamber Injector Chamber Pressure
2 - Full Up Motor Test With Cutoff (4 Tests)	Evaluate Change in Performance with Oxidizer Flow Change	Programmed Oxidizer and Fuel Flows 88,964 N Thrust	40 sec	
3 - Full Up Motor Test (3 Tests)	Demonstrate GG Extinguishment with Oxidizer Flow Control	Programmed Oxidizer and Fuel Flows 88,964 N Thrust	50 sec	
4 - Full Up Motor Test Full Duration (4 Tests)	Combustor Geometry Variation to Determine Characteristic Length	Programmed Oxidizer and Fuel Flows 88,964 N Thrust	50 sec	

Note: After each test disassemble hardware to check for discrepancies.

Allied-Signal will perform development testing of the inducer (a specially designed axial flow impeller) in water. Since the inducer is highly loaded, careful development is crucial to the pump's reliability. In addition, the foil bearing will be tested for performance evaluation. Bearing stiffness, load capacity, running torque, damping and stability will be recorded.

Initially, air will be used as the test fluid. The bearing cavity pressure will be raised to match the Reynolds number in the LOX turbopump. The test speed will also be increased to account for the higher viscosity of oxygen relative to air. Liquid nitrogen will then be used to simulate the incompressibility of the LOX.

Once the component tests have been completed and evaluated, the design will be updated for manufacture. An 11-month fabrication cycle is planned. Once the pump is assembled and inspected, Allied-Signal will conduct a full-load test. Hot air will be used to drive the turbine, and liquid nitrogen will be used in the pump. The test will measure pump pressure rise, leakage, balance piston, and bearing operation. A full-speed test will be accomplished as a subset of the full-load test. The unit will be tested at 26-percent overspeed to verify mechanical integrity and demonstrate acceptable vibration levels and shaft motion.

The final testing will be performed jointly by ARC and Allied-Signal. A fully assembled pump will be shipped to ARC and tested to evaluate turbine, inducer, and pump performance in the design fluids.

2.9.2.11 <u>Component Integration (Task 11)</u>

ARC WILL INTEGRATE THE MBA COMBUSTOR/NOZZLE, FITVC, AND OXIDIZER TURBOPUMP IN A 444,822N (100,000 POUND), 190.5 CENTIMETER (75 INCH) THRUST MOTOR.

The objective of this integration testing is to verify the predicted performance of the hybrid motor that includes all of the propulsion and TVC components (including turbopumps) developed for the gas generator hybrid. Specific performance parameters include specific impulse, thrust termination, conformance to TVC duty cycle and thrust profile, stability, and extinguishment.

A detailed test plan will be written before component integration takes place. The test plan will update the matrix of tests to be performed, the duty cycle, the data acquisition and instrumentation, test procedures, data reduction methods, and reporting format. The plan will be submitted to MSFC for review and approval.

Detailed designs and test support systems will be established concurrent with the test plan. This effort will be documented with a complete drawing package.

Motor hardware will be of heavywall flanged construction which utilizes cartridge-loaded gas generator fuel grains. The motors have been configured to minimize hardware risks. The motor hardware case thickness, insulation, combustor/nozzle and injector plate material safety factors have been increased to 1.8. The gas generator case will be a steel heavywall construction with two flanged openings. The forward and aft closures will have additional ports for test instrumentation. The gas generator will be cast from four 300-pound mixes. The hardware will be scaled from the design used in the 88,964N (20,000 pound) thrust demonstration motor. A 190.5 centimeter (75 inch) diameter gas generator is required to produce the 444,822N (100,000 pounds) of thrust planned for this task.

The injector used in the subscale demonstration will be scaled for the 190.5-centimeter gas generator. The injector design will be verified by bench tests with liquid nitrogen. Pressure drop versus oxidizer flow rate will be measured and compared to the predicted results.

The test matrix for this effort, shown in Table 37, is outlined as follows:

- Series 0 - Gas generator operation only with no oxidizer flow (2 tests, 75-second duration).
- Series 1 - Gas generator operation with programmed oxidizer flow rate scheduled, no TVC (2 tests, 35-second duration).
- Series 2 - Gas generator operation at maximum operating pressure with reduced TVC duty cycle (2 tests, 50-second duration).
- Series 3 - Gas generator with programmed oxidizer flow rate and normal TVC duty cycle, but terminate oxidizer after 75 seconds (2 tests).

Table 37. ***Component integration test summary, Task 11.***

<table>
<tr><th>Test Series</th><th>Test Objective</th><th>Test Conditions</th><th>Test Duration</th><th>Variables Measured</th></tr>
<tr><td>0 -
Full Duration GG Test
(2 Tests)</td><td>Test GG Parameters for Full Duration Test Firing</td><td>Monitor Flow Rate</td><td>75 sec</td><td>GG Chamber Pressure
GG Mass Flow Rate</td></tr>
<tr><td>1 -
Full Up Motor Test With Cutoff
(2 Tests)</td><td>Maintain GG PMBT
No TVC Duty Cycle</td><td>Programmed Flow Rates for Oxidizer and Fuel
444,822 N Thrust</td><td>35 sec</td><td rowspan="4">GG Pressure and Flow Rate

LOX Total Flow Rate

LOX Flow Rate to Combustion Chamber

LOX Flow Rate to TVC Injectors

Chamber Pressure

Total Thrust

Pressure Drop Across Chamber Injector and TVC Injector</td></tr>
<tr><td>2 -
Full Up Motor Test With Cutoff
(2 Tests)</td><td>Gas Generator at Maximum Pressure
Reduced TVC Cycle</td><td>Constant Maximum Oxidizer and Fuel Flow Rates
444,822 N Thrust</td><td>50 sec</td></tr>
<tr><td>3 -
Full Up Motor Test Full Duration
(2 Tests)</td><td>Maintain GG PMBT
Normal TVC Duty Cycle
Terminate Oxidizer at 75 Seconds</td><td>Programmed Flow Rates for Oxidizer and Fuel
444,822 N Thrust</td><td>75 sec</td></tr>
<tr><td></td><td></td><td></td><td></td></tr>
</table>

Note: After each test disassemble hardware to check for discrepancies.

Thrust and characteristic exhaust velocity (C-star) efficiency measurements will be made. The gas generator and combustor/nozzle will be instrumented with thermocouples to define heat release distributions. Information gained from the 100,000-pound thrust tests will be used with the modeling results from Task 4 to permit the design of a 4,448,221N (1,000,000 pound) thrust motor. In addition, the tests will be defined to identify stability margins in various resonant frequency regimes.

2.9.2.12 Risk Assessment (Task 12)

ARC'S PLAN ALLOWS FOR IDENTIFICATION OF ITEMS REQUIRING ADDITIONAL DEVELOPMENT AND ASSESSMENT.

After completion of the component integration tests, an overall review of the status of the development effort will be made. Items requiring additional development will be identified, and the probability of success will be assessed. To the extent possible, recommendations for further development or new initiatives with improved reliability or cost data will be made to MSFC during the course of program and in the final summary report.

2.9.2.13 Phase 3 Planning (Task 13)

ARC'S PROGRAM ALLOWS FOR IDENTIFICATION AND UPDATE OF THE PHASE 3 ACTIVITIES TO INCLUDE THE TEST RESULTS FROM PHASE 2.

During the Phase 2 efforts, ARC will update our plans for the Phase 3 4,448,221N (1,000,000-pound) thrust demonstration. Our technical reports will include an update of facility requirements, instrumentation, data acquisition requirements, and documentation.

2.9.2.14 Propulsion System Integration (Task 14)

BOEING WILL INTEGRATE THE TECHNOLOGY RESULTS INTO A SYSTEM TO DETERMINE IF ALL DESIGN CRITERIA ARE BEING DEVELOPED.

Since the point design developed in Phase 1 was not referenced to any particular system, ARC selected the STS and ALS launch platforms to calculate some of the design requirements for the trades. This assumption permitted a preliminary evaluation and identified additional work required to estimate reliability and cost.

In this task, ARC will subcontract with the Boeing Aerospace Company to provide detailed assessments of the impact of each technology on the hybrid development. Boeing will determine overall cost and schedule risk associated with the integrated booster and identify critical areas requiring additional work. Boeing's early integration will result in development and verification cost savings.

2.9.2.15 Facility Requirements (Task 15)

ARC WILL IDENTIFY AND PLAN FOR THE PHASE 3 FACILITY REQUIREMENTS.

Early in the program, we will review the hybrid design and establish a facility plan and manufacturing plan for the demonstration components. Due to the size of the Phase 3 gas generator, our initial plan will focus on manufacturing the gas generator grains at our Camden, Arkansas facility with shipment by rail to MSFC where they will be assembled with the remaining components for testing.

The facility and manufacturing plan will identify the components to be fabricated, a vendors contract list, training requirements, the materials of construction, the specifications required, capital requirements, schedules, critical paths, milestones, transportation plans, assembly procedures, locations and requirements, permits, and government agencies to be contacted.

We will include in our plans the quality assurance and nondestructive evaluations that will be required for the critical components (gas generator, combustor/nozzle, oxidizer and helium storage tanks, turbopump, injector, combustion controller, and igniter).

ARC will submit our plans to MSFC for review and approval. During the program, the requirements will be updated to include the results of the program tasks.

2.10 Million-Pound Thrust Demonstration Plan

2.10.1 Introduction

The Phase 3 Hybrid Propulsion Technology Program will demonstrate scaleup of components developed in Phase 2 (Technology Acquisition) in a 4,448,221N (1,000,000 pound) thrust demonstration motor. It will also provide an assessment of the technology development risks that remain and need to be addressed in full-scale engineering development. This phase is planned as a 36-month effort, comprising five tasks.

The motor demonstrations will be conducted in the F-1 engine test stand at MSFC. The program schedule is presented in Figure 83. It shows the major tasks to be performed and includes milestones. The principal program tasks are:

Task	Title
1	Motor Design
2	Component Procurement and Verification
3	Motor Assembly and Shipment
4	Testing
5	Data Analysis and Documentation

ARC's million-pound thrust demonstration plan is structured to request direct and frequent MSFC involvement from the early planning and implementation through to the data analysis and documentation stages. MSFC will be involved in the decision making process at all critical decision points. Details on the work to be performed are discussed in the following sections.

2.10.2 Motor Design (Task 1)

THE PHASE 2 RESULTS AND PROGRAM RECOMMENDATIONS WILL BE REVIEWED AND INTEGRATED INTO THE PHASE 3 PROGRAM PLAN.

After completion of the six 444,822N (100,000 pound) thrust component integration tests in Phase 2, ARC will make recommendations to MSFC concerning component development. In this motor design task, ARC and its subcontractors, Boeing and Allied-Signal, will review the list of recommendations and perform a detailed analytical evaluation of the design and material selection impacts. We will prepare a detailed program plan to implement the changes and present this plan to MSFC for review and approval. Included in the program plan will be the following:

- A schedule and request to proceed with the procurement of long-leadtime hardware.
- A manufacturing plan listing facilities to be used, schedule, critical personnel, major milestones, and quality assurance plan and documentation efforts.
- Preliminary test plan for the F-1 stand which will include all of the milestones. The stand shall be ready for occupancy 24 months after

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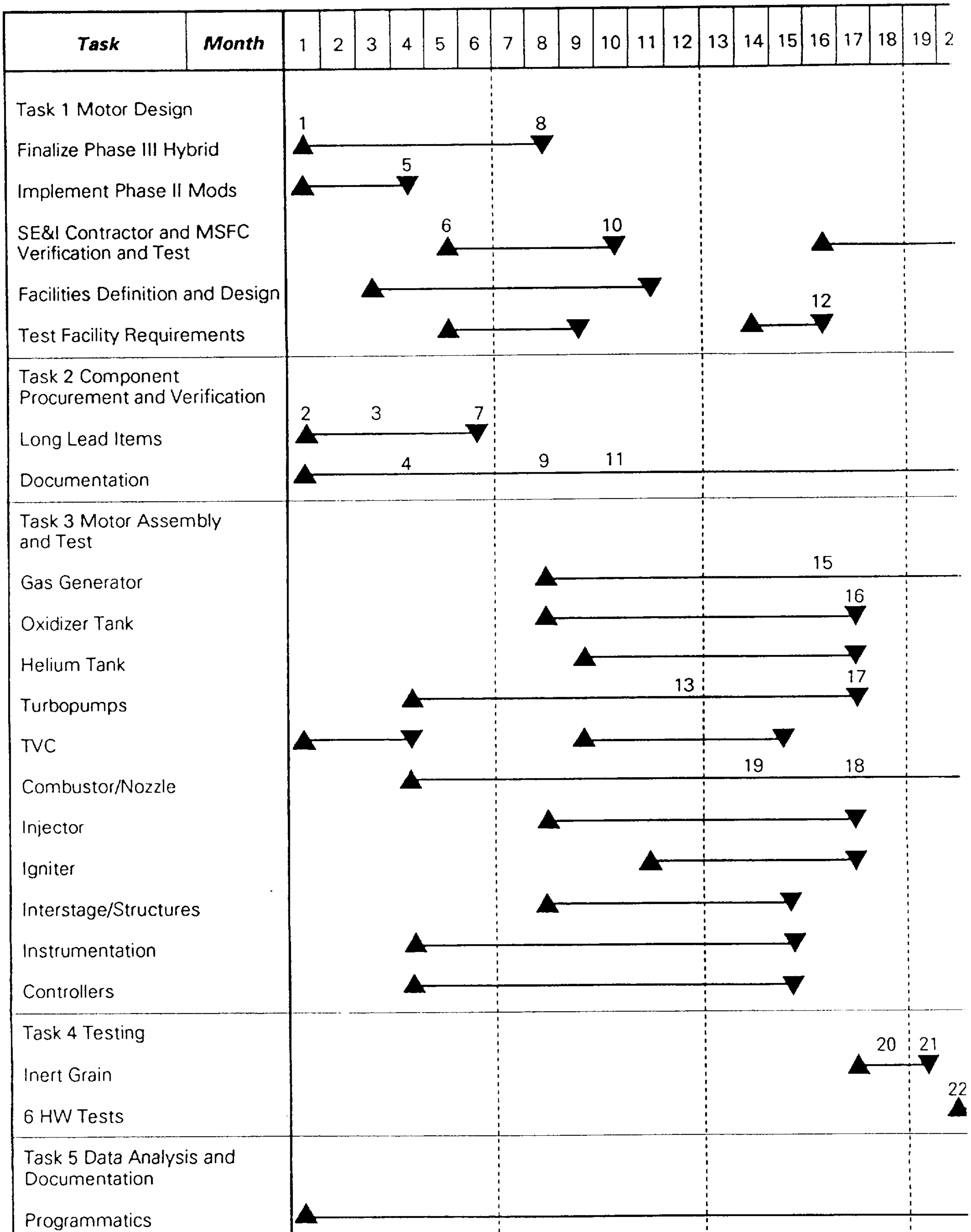

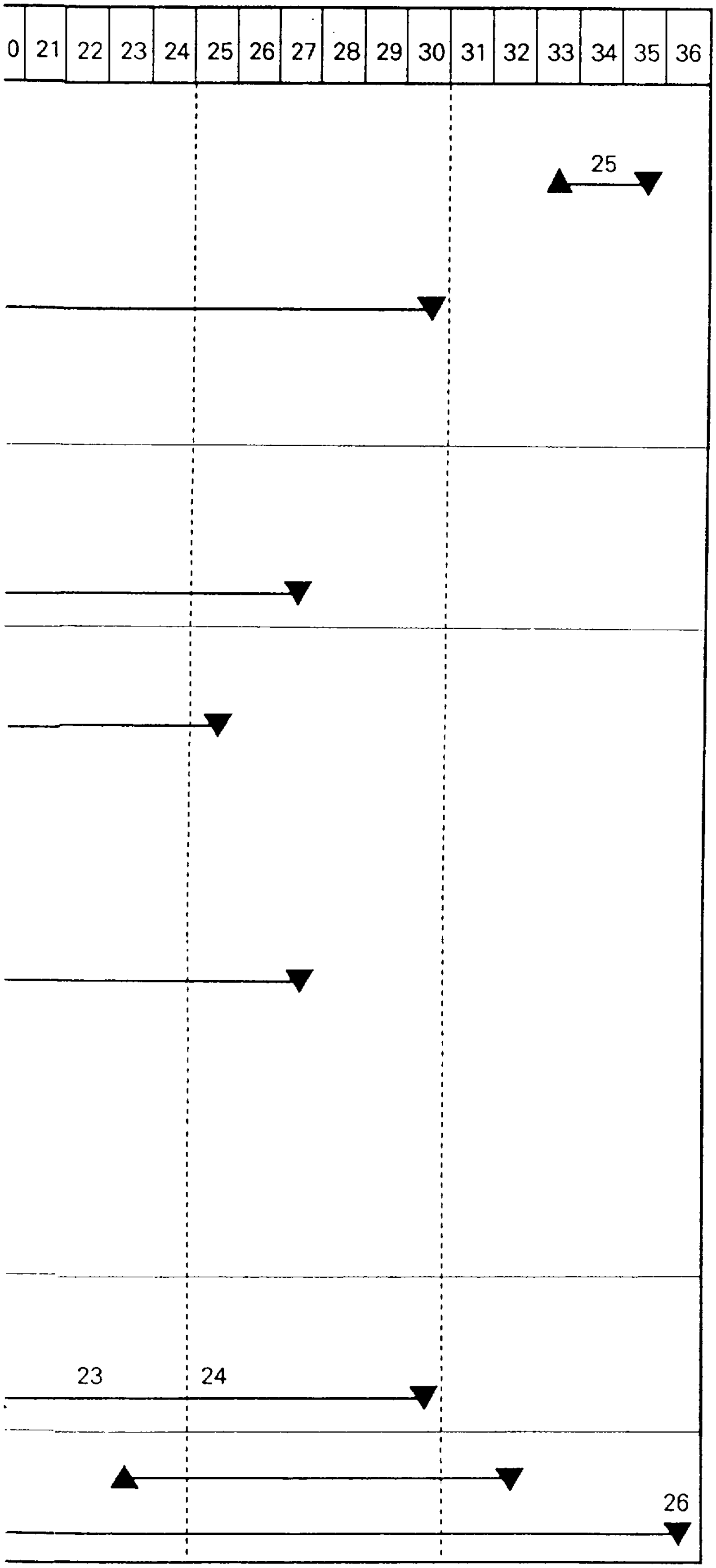

Phase III Milestones

1. Contractor/MSFC meeting to review hybrid design.
2. Obtain MSFC approval/funding authorization for long lead hardware.
3. Let contracts for long lead hardware.
4. Initiate transportation permits and specifications.
5. Complete design recommendations from Phase II.
6. Formal review with MSFC on production and schedule.
7. Review with MSFC the test schedule and manufacturing plan.
8. Complete process specifications, drawings, procedures for signoff.
9. Establish quality assurance procedures, specifications, and product recovery.
10. Establish a test team of ARC, Boeing and MSFC personnel.
11. Establish material review board; set procedures and schedule.
12. Investigate MSFC test sites; develop Level I plan.
13. "Green run" pumps at Allied Signal.
14. Complete fabrication of first three MBA nozzles.
15. Complete fabrication and inspection of heavywall hardware.
16. Receipt of oxidizer and helium tanks.
17. Receipt of turbopumps.
18. Trial fit all components.
19. Ship all components to MSFC.
20. Conduct simulated test with inert gas generator.
21. Finalize all test procedures, complete all safety reviews.
22. Finalize stand checkout.
23. Run gas generator tests.
24. Run hybrid motor tests.
25. Complete final design; prepare drawing package.
26. Submit final report.

Figure 83. Program schedule.

0290-HYBRID-RPT

authority to proceed. The plan will define transportation and handling effects, preliminary test procedures, instrumentation requirements and installation procedures, stand checkout, and data collection and analysis.

- An appraisal of any interface control drawing effects on the Level II documentation for the 4,448,821N (1,000,000 pound) and 15,568,776N (3,500,000 pound) thrust motors.
- A detailed definition of the tests to be run, their objectives, expected results, and criteria for success or failure.

This program plan will be reviewed and updated during the Phase 3 tasks and submitted to MSFC for review.

2.10.3 Component Procurement and Verification (Task 2)

SPECIFICATIONS WILL BE WRITTEN FOR EACH HYBRID COMPONENT REQUIRED IN THE DEMONSTRATION MOTORS. THE SPECIFICATIONS WILL BE DISTRIBUTED TO VENDORS FOR QUOTATION AND CAPABILITY VERIFICATION.

ARC, with support from Boeing, will develop mechanical, electrical, and performance specifications for the gas generator hybrid components. Included in the specifications will be packaging and shipment requirements. The specifications will be submitted to MSFC for review. The specifications will be distributed to vendors selected from a compilation of companies that have performed well on previous ARC and Boeing contracts. The ARC Procurement Department will verify the companies' ability to meet the design specifications and schedule for the major components by completing a site visit prior to contract award.

ARC and Boeing will implement the quality assurance plan and establish scheduled visits at each major vendor for inspection of hardware and documentation. ARC will implement a system for off-specification product recovery and compliance. A Material Review Board of experienced ARC and Boeing representatives will be established to review deviations and discrepancies reported in the quality inspectors site reports; MSFC representation on the board will be requested.

Each major manufactured component will be inspected at the vendor's plant prior to shipment. All of the components will be reinspected at the delivery point.

2.10.4 Motor Assembly and Shipment (Task 3)

ARC WILL FABRICATE HYBRID MOTORS, BOEING WILL FABRICATE THE OXIDIZER TANK AND COMBUSTION CONTROLS, AND ALLIED SIGNAL WILL FABRICATE THE OXIDIZER TURBOPUMPS. ALL OF THE COMPONENTS WILL BE TRIAL FIT AND THEN SHIPPED TO MSFC FOR TESTING.

This task's technical effort will result in the casting, curing, and inspection of eight 86,183-kilogram (190,000-pound) gas generators; fabrication and inspection of eight, monolithic braided ablative combustor/nozzles, and the assembly and checkout of the oxidizer delivery system.

ARC's approach for the demonstration tests is to fabricate an integrated system comprised of gas generator, combustor/nozzle, oxidizer tank, helium expulsion system, turbopumps, oxidizer lines, valves, controllers, and thrust vector control. The system will be installed in the F-1 stand and reused. Consumables (gas generator and combustor/nozzle) will be replaced after each test.

ARC will design a heavyweight, monolithic cartridge-loaded, 317.5 centimeter (125 inch) diameter carbon fiber/polyimide composite gas generator case with integral domes. A total of ten cases will be manufactured by an outside vendor and shipped to ARC. After the cases have passed inspection, they will be insulated and prepared for fuel loading. One of the cases will be loaded with an inert simulated fuel, and eight will be loaded with fuels for testing (two gas generator-only tests, six integrated hybrid tests); the tenth case will be a spare. Each fuel grain will be cured and then inspected using x-ray and ultrasonics. The inert gas generator will be shipped to MSFC for trial installation and checkout. The first two live gas generators will be used for scaleup proof of concept. The remaining gas generators will be cast in lots of two following a review of each preceding test or test series. For this plan, we have assumed a new casting facility would be set up in the Highland Industrial Park adjacent to our existing Camden Arkansas facility.

Design of the oxidizer delivery system will be directed by ARC/Liquid Propulsion. They will send representatives to our Virginia Propulsion Division facilities to direct the receipt, checkout, and assembly of components. Allied Signal, under subcontract to ARC, will provide primary and redundant oxidizer turbopumps. The pumps will be green run by Allied Signal at their facilities prior to shipment. The remaining components will be assembled into subsystems, passivated, and then sealed for shipment. The oxidizer tank fabrication and delivery will be directed by Boeing. They will inspect the tank prior to shipment, passivate the interior, and then transport it to the final assembly point at MSFC. The helium expulsion system will be manufactured by ARC. The composite tank will be fabricated by the Composites Group of Virginia Propulsion Division, and the remaining components will be assembled and integrated by Liquid Propulsion.

The MBA combustor/nozzle will be fabricated at our Virginia facilities. The nozzle preforms will be braided with quartz fibers in an automated cylindrical braider using a rubber mandrel corresponding to the nozzle internal dimensions. The preform will be densified by ARC using solvated phenolic resin. The preform will be cured at 350°F and then consolidated to accommodate shrinkage.

The hybrid motor components will be shipped to ARC's Arkansas Propulsion Division. ARC, Boeing, Allied Signal, and MSFC personnel will inspect and trial fit the components to finalize the assembly procedures. This integration will ensure that: (1) MSFC stand personnel are aware of the procedures and they are being implemented; (2) MSFC can make ARC aware of required deviations or discrepancies needed for the tests; and (3) if there are any problems they can be resolved prior to arrival at MSFC.

Once the components have been tested, they will be crated and shipped to MSFC. Upon arrival, they will be reinspected and then stored for testing.

2.10.5 Testing (Task 4)

ARC WILL CONDUCT TWO GAS-GENERATOR-ONLY TESTS AND SIX HYBRID MOTOR TESTS TO VALIDATE PROOF OF CONCEPT.

ARC will assist MSFC to conduct two gas generator tests in the F-1 stand. The gas generator case and combustor/nozzle will be fully instrumented for pressure, strain, and temperature. For these tests, a smaller nozzle throat will be used to produce the gas generator pressures expected during the hybrid tests. The two gas generator tests will validate the predicted fuel delivery rate for the full-up tests and will be used to check out data acquisition. The tests will be run for a full 135-second duration.

The data from the two gas generator tests will be analyzed, and the results will be used to check the grain design for the hybrid tests. If a different mass flow rate is required, the grain design will be modified and new casting tooling fabricated.

ARC will mix and cast the next two gas generators required for the first two test series, shown in Table 38. The gas generators will be inspected, packaged, and shipped to MSFC. The gas generators will be bolted to the oxidizer tank for testing. The first two tests will be run and the data analyzed.

We will manufacture the next two gas generators following review and analysis of Series 1 and 2. If changes are required, they will be incorporated prior to the next series of tests. The last two tests will be run to demonstrate repeatability. After each hybrid motor test, all of the hardware will be disassembled and inspected. If necessary, specific tests will be repeated to assure resolution of any problems. A summary of the tests follows:

<u>Test Series 0</u>: Two full-duration tests of the gas generator, no oxidizer injection. Run with reduced nozzle throat to produce design operating pressure.

<u>Test Series 1</u>: Full-duration test of the complete design. Maintain predicted thrust-time trace within the operating temperature limits. Run reduced TVC slew angle and duty cycle. Terminate oxidizer flow rate after 35 seconds.

<u>Test Series 2</u>: Full-duration test scheduled. Run a prescribed TVC duty cycle. Measure structural loads at the simulated vehicle attachment points. Terminate oxidizer flow rate after 70 seconds.

<u>Test Series 3</u>: Repeat Test Series 2 with cutoff at 105 seconds.

<u>Test Series 4</u>: Full-duration test with the maximum prescribed TVC duty cycle. Program cutoff at 135 seconds with TVC deflecting thrust in maximum degree position.

<u>Test Series 5</u>: Repeat Test Series 4 for statistical data (2 tests).

Table 38. *Large subscale motor test series.*

Test Series	**Test Objective**	**Test Conditions**	**Test Duration**	**Variables Measured**
0 - Full Duration GG Test (2 Tests)	Test GG Parameters for Full Duration Test Firing	Programmed Fuel Flow No Oxidizer Full Duration	135 sec	GG Chamber Pressure GG Mass Flow Rate
1 - Full Up Motor Test With Cutoff (1 Test)	Maintain GG PMBT Full Duration Attempt Shutdown Motor Record Vehicle Loads Run TVC Duty Cycle	Programmed Fuel and Oxidizer Flow 4,448,221 N Thrust	35 sec	GG Pressure and Flow Rate LOX Total Flow Rate LOX Flow Rate to Combustion Chamber LOX Flow Rate to TVC Injectors Chamber Pressure Total Thrust Pressure Drop Across Chamber Injector and TVC Injector If Cutoff then Measure Loads from Strain Gauges
2 - Full Up Motor Test With Cutoff (1 Test)	Maintain GG PMBT Full Duration Attempt Shutdown Motor Record Vehicle Loads Run TVC Duty Cycle	Programmed Fuel and Oxidizer Flow 4,448,221 N Thrust	70 sec	
3 - Full Up Motor Test With Cutoff (1 Test)	Maintain GG PMBT Full Duration Attempt Shutdown Motor Record Vehicle Loads Run TVC Duty Cycle	Programmed Fuel and Oxidizer Flow 4,448,221 N Thrust	105 sec	
4 - Full Up Motor Test Full Duration (1 Test)	Maintain GG PMBT Full Duration Test Run Maximum TVC Duty Cycle	Programmed Fuel and Oxidizer Flow 4,448,221 N Thrust	135 sec	
5 - Full Up Motor Test Full Duration (2 Tests)	Maintain GG PMBT Full Duration Test Run Maximum TVC Duty Cycle	Programmed Oxidizer and Fuel Flow 4,448,221 N Thrust	135 sec	

Note: After each test disassemble hardware to check for discrepancies.

For each test, the following will be measured: motor thrust, gas generator pressure, combustor pressure, oxidizer flow rate to the TVC injectors and main injector, total oxidizer consumed, injector pressure drop, motor case strain, combustor strain, and structural loads to the core vehicle. A summary of the instrumentation is shown in Table 39.

2.10.6 Data Analysis and Documentation (Task 5)

THE 4,448,221N (1,000,000 POUND) THRUST MOTOR DATA WILL BE ANALYZED, AND A STATISTICAL EVALUATION COMPLETED TO VERIFY OVERALL PERFORMANCE.

ARC will measure the mechanical and ballistic properties of each fuel mix used to cast the motors to determine mechanical property and performance repeatability. The variation in mix-to-mix properties will be established, and the impact on cost and reliability will be calculated.

The test data will be recorded on FM tape for playback at ARC. ARC's standard static firing data analysis software will be used to analyze the data. We will supply MSFC with the test data for independent evaluation of the tests. Complete test reports will be submitted to MSFC and will include a description of the test, test facility and equipment, instrumentation, and test data analysis. A statistical evaluation will be completed to verify: (1) reproducible gas generator operation; (2) hybrid motor ignition; (3) extinguishment within the specified limits; (4) TVC requirements and duty cycle; and (5) turbopump operation. In the event of a test anomaly or failure, ARC will deliver an oral and a written failure analysis report. Included in the report will be a corrective action report to assure MSFC that ARC has taken the appropriate action to minimize recurrence.

A logbook for all motors tested at MSFC will be prepared to provide history and traceability. The logs will include the results of the nondestructive evaluation test, manufacturing and inspection records, and event records or discrepancy reports.

To complete the Phase 3 activities, ARC will submit a final report that summarizes all technical activities accomplished. Included in the report will be an updated booster point design scaled to meet the performance requirements, a booster drawing package, and a detailed full-scale engineering development plan.

Table 39. Phase 3 Motor Instrumentation.

Channel	Description	Value	Channel	Description	Value
P_1	Gas Generator	2K	S_1	Case Forward Dome 0° Fiber	2%
P_2	Gas Generator	2K	S_2	Case Forward Tan 0° Fiber	2%
P_3	Combustor	2K	S_3	Case Mid Hoop 0°	2%
P_4	Combustor	2K	S_4	Case Mid Axial 0°	2%
P_5	Oxidizer Tank	1K	S_5	Case Mid Axial 180°	2%
P_6	Helium Tank	20K	S_6	Case Mid Hoop 180°	2%
P_7	Helium Tank	20K	S_7	Case Aft Tan 0° Fiber	2%
P_8	ΔP TVC Injector	.5K	S_8	Case Aft Dome 0° Fiber	2%
P_9	ΔP Main Injector	.5K	S_9	Combustor Axial 0°	2%
			S_{10}	Combustor Hoop 0°	2%
F_1	Forward Thrust	1000K	S_{11}	Nozzle Axial 0°	2%
F_2	Forward Thrust	1000K		Nozzle Hoop 0°	2%
F_3 A&B	TVC Test Side Forward	50K	S_{12} A&B	Interstage Skirt Axial 0°	2%
F_4 A&B	TVC Test Side Aft	50K	S_{13} A&B	Interstage Skirt Hoop 0°	2%
			S_{14} A&B	Oxidizer Tank Axial 0°	2%
T_1 A&B	Case Forward Dome 0°	500°F	S_{15} A&B	Oxidizer Tank Hoop 0°	2%
T_2 A&B	Case Forward Dome 180°	500°F	S_{16} A&B	Oxidizer Tank Axial 180°	2%
T_3 A&B	Case Aft 0°	500°F	S_{17} A&B	Oxidizer Tank Hoop 180°	2%
T_4 A&B	Case Aft 180°	500°F			
T_5 A&B	Nozzle Flange 45°	500°F			
T_6 A&B	Nozzle Cone (1/2)	1500°F			
T_7 A&B	Nozzle Cone (3/4)	1500°F			
T_8 A&B	Nozzle Cone (Aft)	1500°F			

APPENDIX A

OXIDIZER FEED SYSTEM TRADE STUDIES

PRESSURE FED

PUMP FED

PRESSURE FED SYSTEMS

Introduction

The objective of this task was to investigate pressure-fed oxidizer feed systems for both the classical hybrid (HC) and gas generator hybrid (GG). Oxidizers to be evaluated for each hybrid approach were 95-percent hydrogen peroxide (H_2O_2) and liquid oxygen (LOX). The depth of this study was to be sufficient to make major feed system selections and was not intended to include any detail component designs.

A typical pressure-fed oxidizer feed system consists of an oxidizer tank and a means of pressurizing the oxidizer tank. Trade studies were conducted to enable the selection of the appropriate pressurization subsystem based upon the following criteria.

- Safety
- Cost
- Weight

Reliability and safety are of equal importance. Each feed system was designed so that a single point failure would not cause failure of the mission, the only exceptions being the oxidizer storage tank and the helium storage vessel. Since these components are benign in operation, they should have a 100-percent reliability if the design and fabrication processes are satisfactory; therefore redundancies are not required.

The thrust profile of the mission requires that the thrust be varied (throttled) over a fairly wide range (1.6:1). Thus, tank pressure must be varied to accomodate the range of oxidizer flow rates required to support the thrust profile. Table A-1 provides a summary of booster system requirements upon which the oxidizer system trade studies were based.

The oxidizer feed system, for all these designs, consists of four 20.3 centimeter (8-inch) liquid manifolds from the oxidizer tank to the injector valves. Normally closed explosive isolation valves (isolvalve) are located in each oxidizer line. A normally open isovalve is located at the exit to the oxidizer tank which can be actuated for emergency shut down. The classical

Table A-1. System Requirements

	HC/H_2O_2	GG/H_2O_2	HC/LOX	GG/LOX
Oxidizer Load, KG	467,382	431,810	362,167	304,876
Max Oxidizer Flow Rate, KG/sec	4,814	4,454	3,805	3,144
Max Chamber Pressure, MPa	7.48	7.48	7.48	7.48
Min Chamber Pressure, MPa	3.45	3.45	3.45	3.45

hybrid has the additional requirement of gasifying the oxidizer prior to injection into the solid motor combustion chamber.

Pressurization System Trades

Systems Using Hydrogen Peroxide

Ninety-five percent H_2O_2 has many favorable features as an oxidizer. It has a high density, is noncryogenic, and has a relatively high mixture ratio with the solid fuel constituents. It is also an energetic monopropellant which offers a number of potential advantages. However, is also decomposes at relatively low temperatures which can lead to safety problems, and is not in wide use today as an oxidizer.

Pressurization system options that were evaluated for this system are summarized in Table A-2, along with advantages and disadvantages for each approach. Some pressurization options were immediately screened out, such as warm gas (N_2H_4), due to the possibility of reaction of pressurant gas with the liquid H_2O_2 which could lead to a catastrophic uncontrolled reaction. The warm gas (H_2O_2) approach requires that the decomposed H_2O_2 be cooled to a temperature that precludes self-decomposition of the liquid H_2O_2 adding complexity and cost to the system.

A schematic diagram of a cold gas (helium) pressurization system is shown in Figure A-1. Helium is fed from a high pressure storage bottle through explosively-actuated isolation valves to pressure regulating valves and the oxidizer tank. A relief valve is present downstream of the regulators to preclude the overpressurization of the tanks. A fill/vent port is used to prepressurize the tank to normal operating pressure shortly before launch. This system is simple and has a high historical reliability, but results in heavy, large pressure bottles and heavy helium loads. Pressurization system weights as a function of helium storage pressure for both HC and GG systems are presented in Table A-3.

The schematic of a warmed helium system is shown in Figure A-2. The system operates in the same manner as the cold gas system except a solid propellant charge is fired to heat the gas remaining in the bottle at a point in the mission for more efficient expulsion. This approach reduces the weight and size of the system, but is more complex, less reliable, and may produce large solid particles complicating gas filtration to the regulators. Also,

Table A-2. Pressurization System Options (H_2O_2)

PRESSURIZATION SUBSYSTEM	ADVANTAGES	DISADVANTAGES
Cold Gas	Simple Low Cost High Reliability	Heavy Large Bottle Volume
Warmed Helium	Lower Weight	More Complex Solid Particle Filtration Less Reliable
Heated Helium	Low Weight Smaller Volume	More Complex (Heat Exchanger, GG)
Warm Gas (H_2O_2)	Pressurant Stored as Liquid	Complex Warm Gas must be Cooled
Warm Gas(N_2H_4)	Low Weight	Warm Gas may react with Ox Complex

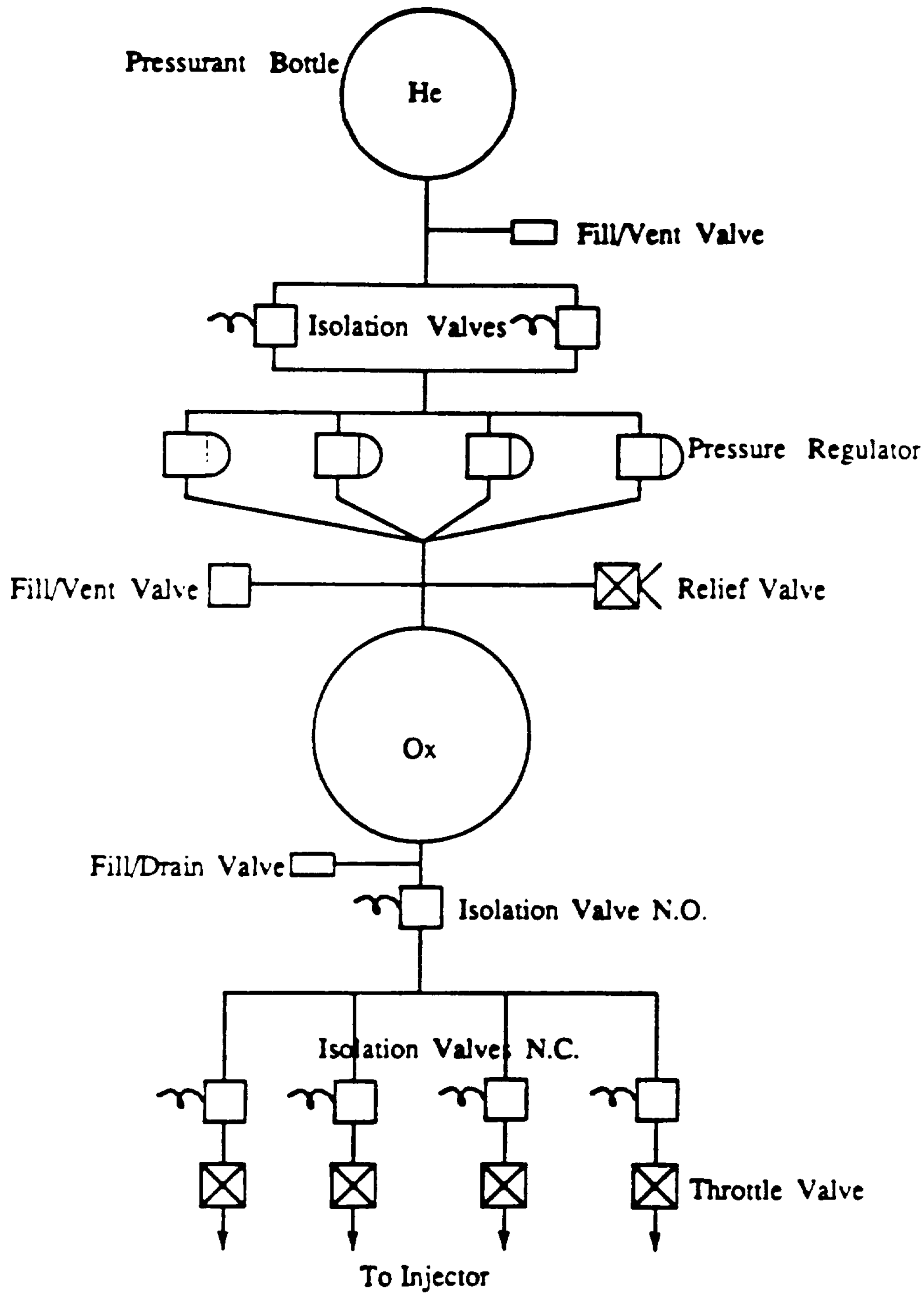

Figure A-1. Cold Gas Pressurization System Schematic.

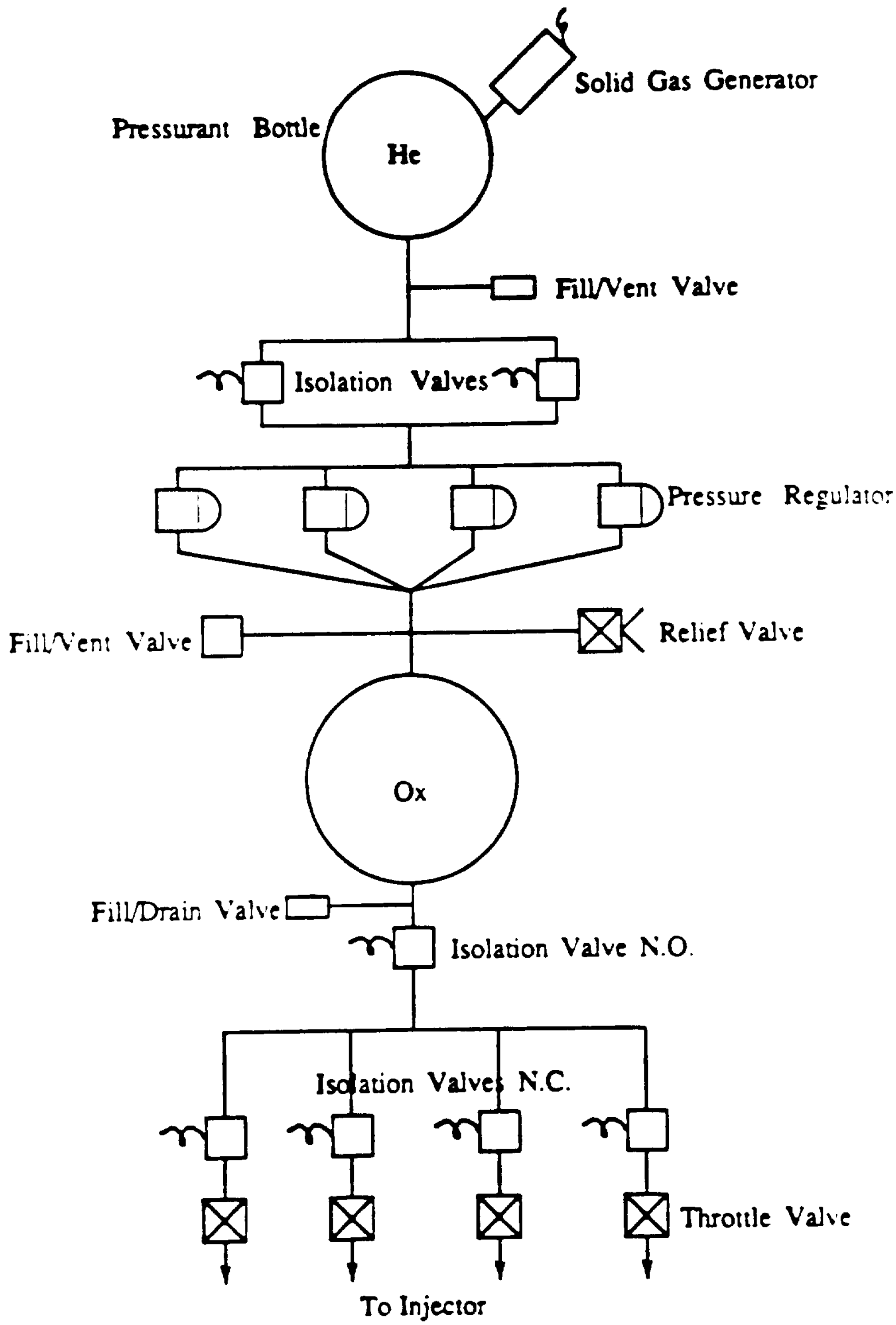

Figure A-2. Warm Gas Pressurization System Schematic.

Table A-3. Cold Gas (He) Regylated Pressurization

a. Gas Generator

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Bottle KG	Bottle Length CM
34.5	10,737	5,762	33,151	2,328
51.7	8,804	5,760	28,822	1,434
68.9	8,037	5,757	27,662	1,091
86.1	7,616	5,756	27,328	909

b. Classical/H_2O_2

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Bottle KG	Bottle Length CM
34.5	11,105	6,073	34,320	2,406
51.7	9,172	6,070	30,014	1,491
68.9	8,397	6,068	28,893	1,137
86.1	7,970	6,065	28,610	923

the accidental firing of a solid charge at the wrong time could cause over-pressurization of the helium storage bottle. This possibility could be countered with a relief valve, but would result in the loss of pressurant.

A heated helium system is shown schematically in Figure A-3. Cold helium is stored at high pressure and low temperature (167K) to minimize bottle size. Helium flows through explosively-actuated isolation valves to a shell-and-tube heat exchanger, where it travels through the tube side of a heat exchanger and is heated to 333K. The heated helium is fed to pressure regulators to pressurize the tank. The shell side of the heat exchanger uses decomposed H_2O_2 from the oxidizer tank to provide heat to the cold helium. Catalytic gas generators are used to decompose the H_2O_2. Decomposition products at approximately 1,111K are fed in countercurrent flow to the shell side of the heat exchanger. The use of heated helium results in a lighter and more compact system as compared to the cold gas system. The addition of more components slightly lowers the predicted reliability of this system.

Pressurization system weights as a function of helium storage pressure and temperature for GG and HC systems are presented in Tables A-4 and A-5. Heat exchanger dimensions as a function of storage pressure are presented in Table A-6. Oxidizer tank weight and dimensions for various cases are presented in Tables A-7 and A-8.

Systems Using LOX

Liquid oxygen (LOX) is a cryogenic oxidizer widely used in the industry today. The main problem in pressurizing LOX is that the pressurant is cooled upon contacting the LOX, thus increasing the amount of pressurant required.

Table A-9 summarizes pressurization system options which were evaluated for the designs using LOX. Advantages and disadvantages are also presented for these systems.

Some approaches were quickly screened out. Solids were not considered due to reactive combustion gases, solid particles, relatively high temperature of the pressurant gas and the problems associated with emergency shutdown. Warm gas (N_2H_4) was ruled out due to the relatively high temperature of its decomposition gases, the complexity of the system, and the reactivity of its pressurant gases.

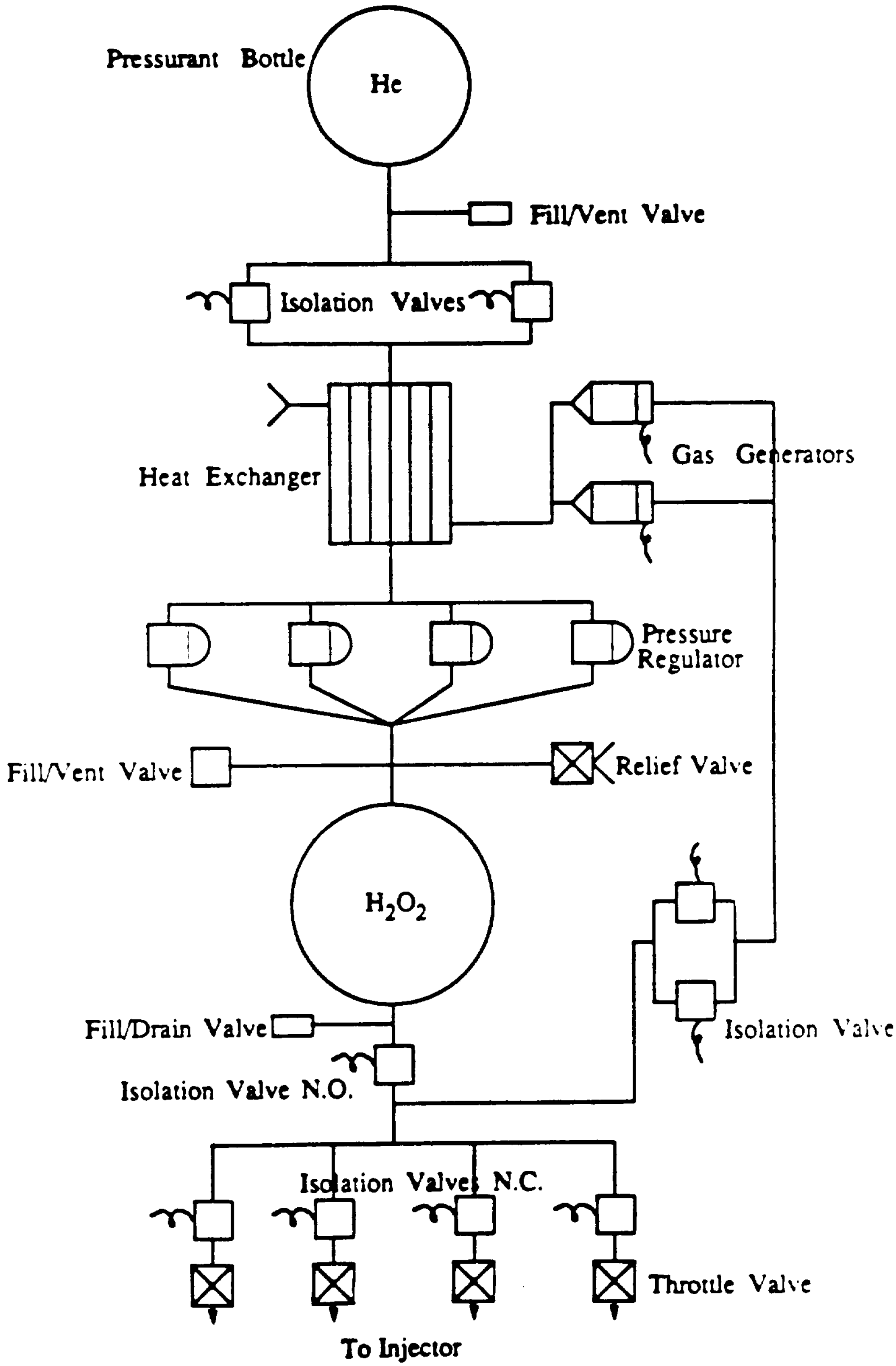

Figure A-3. <u>Cold Gas With Gas Generator/Heat Exchanger Pressurization System.</u>

Table A-4. Cold Gas (He) Regulated with GG/Heat Exchanger Pressurization (GG/H_2O_2)

P_{BO} MPa	$T_{He\ O}$ K	W_{He} Loaded KG	W_{He} Used KG	W_{He} Bottle KG	Bottle Length CM	$W_{H_2O_2}$ Decomposed KG	W_{HX} KG	W_{GG} KG
34.5	55.6	11,095	4,648	9,790	776	6,992	378	62
51.7	55.6	9,226	4,646	9,982	567	7,001	374	62
68.9	55.6	8,501	4,644	10,878	490	6,996	373	62
86.1	55.6	8,114	4,643	11,956	452	6,551	371	62
34.5	111	9,479	4,633	12,895	1,002	5,571	319	50
51.7	111	7,832	4,631	12,137	677	5,571	317	50
68.9	111	7,183	4,629	12,485	554	5,576	310	50
86.1	111	6,830	4,628	13,105	490	5,576	315	50
34.5	167	8,945	4,618	16,406	1,259	4,160	273	38
51.7	167	7,373	4,616	14,885	1,259	4,164	272	38
68.9	167	6,750	4,614	14,827	647	4,164	266	38
86.1	167	6,409	4,613	15,169	558	4,169	272	38

Table A-5. Cold Gas (He) Regulated with HC/Heat Exchanger Pressurization (HC/H_2O_2)

P_{BO} MPa	$T_{He\ O}$ K	W_{He} Loaded KG	W_{He} Used KG	W_{He} Bottle KG	Bottle Length CM	$W_{H_2O_2}$ Decomposed KG	W_{HX} KG	W_{GG} KG
34.5	55.6	11,274	4,863	9,955	789	7,281	378	62
51.7	55.6	9,456	4,860	10,233	580	7,281	374	62
68.9	55.6	8,743	4,858	11,640	503	7,285	373	62
86.1	55.6	8,278	4,856	12,316	464	7,289	371	62
34.5	111	9,667	4,847	13,136	1,021	5,796	319	50
51.7	111	8,057	4,844	12,491	695	5,796	317	50
68.9	111	7,414	4,843	12,887	570	5,805	310	50
86.1	111	7,063	4,841	13,548	505	5,801	315	50
34.5	167	9,136	4,832	16,771	1,285	4,325	273	38
51.7	167	7,596	4,829	15,346	839	4,330	272	38
68.9	167	6,979	4,827	15,333	568	4,330	266	38
86.1	167	6,638	4,826	15,712	575	4,334	272	38

Table A-6. Dimensions of Heat Exchanger and Gas Generator for H_2O_2 System

P_{BO} MPa	T_{HeO} K	L_{HX} CM	D_{HX} CM	L_{GG} CM	D_{GG} CM
34.5	55.6	229	39	11	52
34.5	111.1	192	39	11	46
34.5	166.7	162	39	11	40
51.7	55.6	226	39	11	52
51.7	111.1	189	39	11	46.5
51.7	166.7	161	39	11	40
68.9	55.6	224	39	11	52
68.9	111.1	184	39	11	46
68.9	166.7	156	39	11	40
86.1	55.6	221	39	11	52
86.1	111.1	186	39	11	46
86.1	166.7	158	39	11	40

Table A-7. Oxidizer System Parameters with Cold Gas Regulated Pressurization

System	W_{Tank} KG	L_{Tank} CM	P_{Tank} MPa
GG/H_2O_2	14,461	3,085	10.9
HC/H_2O_2	15,153	3,331	10.6

Table A-8. Oxidizer Tanks with Cold Gas Regulated GG/Heat Exchanger Pressurization

a. GG/H_2O_2 Hybrid

$T_{He\ O}$ K	W_{Tank} KG	L_{Tank} CM	$P_{Tank\ Max}$ MPa
55.6	14,412	3,145	10.9
111.1	14,365	3,135	10.9
166.7	14,319	3,125	10.9

b. HC/H_2O_2 Hybrid

$T_{He\ O}$ K	W_{Tank} KG	L_{Tank} CM	$P_{Tank\ Max}$ MPa
55.6	15,083	3,392	10.6
111.1	15,036	3,382	10.6
166.7	14,989	3,371	10.6

Table A-9. Pressurization System Options

PRESSURIZATION SUBSYSTEM	ADVANTAGES	DISADVANTAGES
Cold Gas Regulated	Simple Low Cost High Reliability	Heavy Large Bottle Volume
Warm Gas ($He/O_2/H_2$)	Simple Low Weight	More Complex (Catalytic Reactor)
Warm Gas (N_2H_4)	Low Weight	Warm Gas may react with ox Complex
Solid	Simple	Hot gases react with ox. Cannot be shut off. Hot gas must be released overboard.

Cold gas pressurization systems are shown schematically in Figures A-4 and A-5 for the GG and HC designs, respectively. The GG/LOX pressurization system is similar to the cold gas system for H_2O_2. The HC/LOX system requires a preburner before injection. A fuel tank containing propane (C_3H_8) is also pressurized by the helium. The fuel tank is sealed by isolation valves. Check valves in the line leading to the oxidizer tank prevent the backwash of gaseous oxygen into the pressurant lines in case the relief valve opens and closes. The cold gas system is simple and reliable, but results in a large and heavy system. Pressurization system weights as a function of helium storage pressure for HC and GG systems are presented in Table A-10.

Figures A-6 and A-7 show schematics of a warm gas (tridyne, $He/O_2/H_2$) pressurization system for the GG and HC designs, respectively. This system utilizes catalytic heating of a nondetonable gas mixture called tridyne, composed of both inert and reactive components in a single bottle, to provide a warm pressurization gas. The catalytic reactor promotes the reaction to heat the predominately inert mixture. The mixture consists of helium (He), oxygen (O_2) and hydrogen (H_2), with O_2 and H_2 proportioned stoichiometrically. This approach results in a much lighter and more compact pressurization system; it was selected as the baseline system for the LOX oxidizer system. Further, separate trade studies selected the gas generator hybrid as the baseline hybrid system.

Pressurization system weights as a function of helium storage pressure for the GG and HC systems are presented in Table A-11. Table A-12 shows the weight and dimensions of the oxidizer and fuel tanks (HC) using cold gas pressurization. Table A-13 shows the weight and dimensions of the oxidizer and fuel tanks (HC) for a warm gas regulated system.

Baseline System Description

Pressurization Subsystem - The baseline pressurization subsystem consists of Tridyne pressurant gas, a carbon fiber/epoxy resin wrapped bottle, a fill/vent valve, redundant pyrotechnically actuated isolation valves, four pressure regulators, a catalytic reactor and associated gas manifolds. A schematic of these components is shown in Figure A-8.

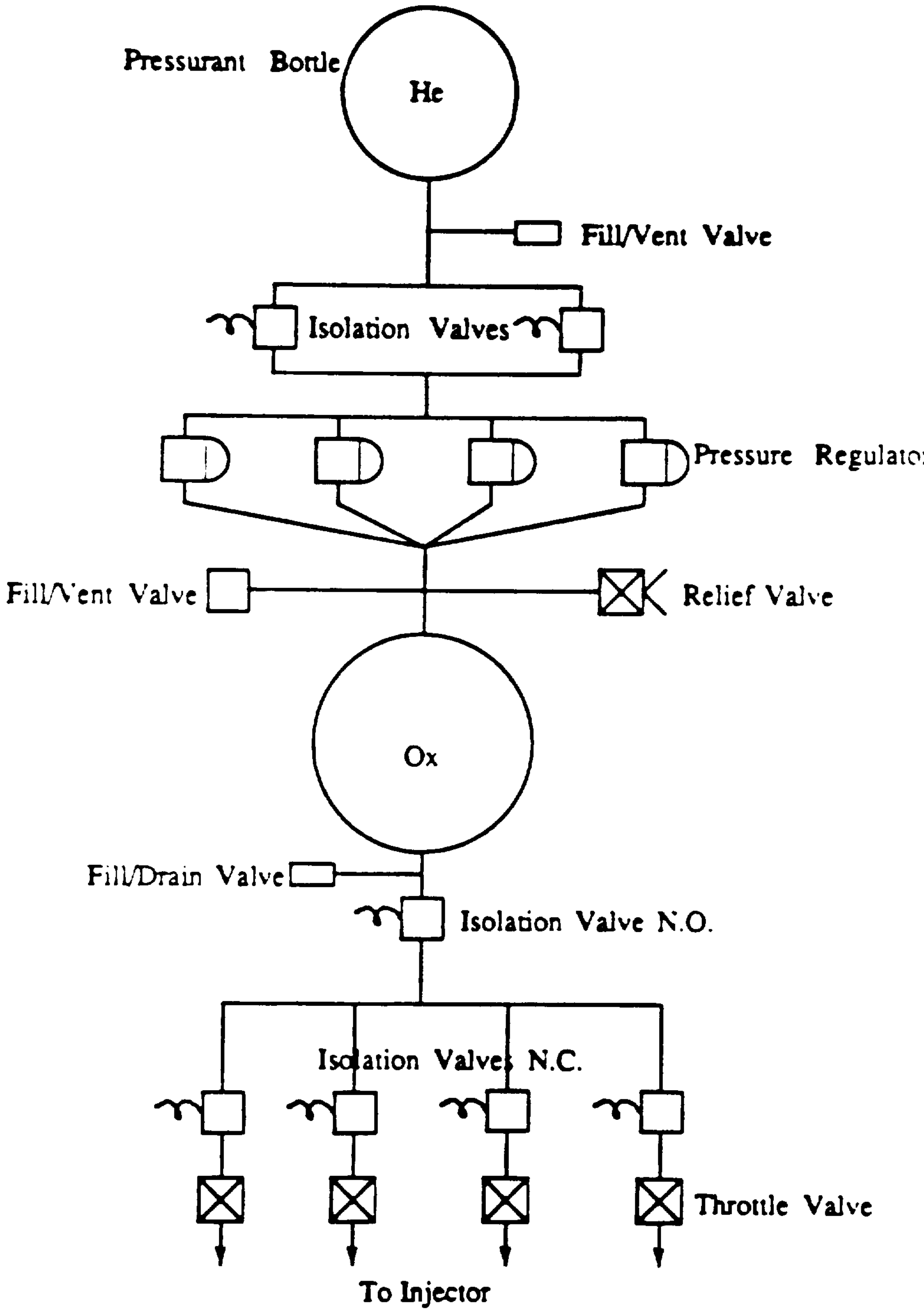

Figure A-4. Cold Gas Pressurization System Schematic for GG/LOX.

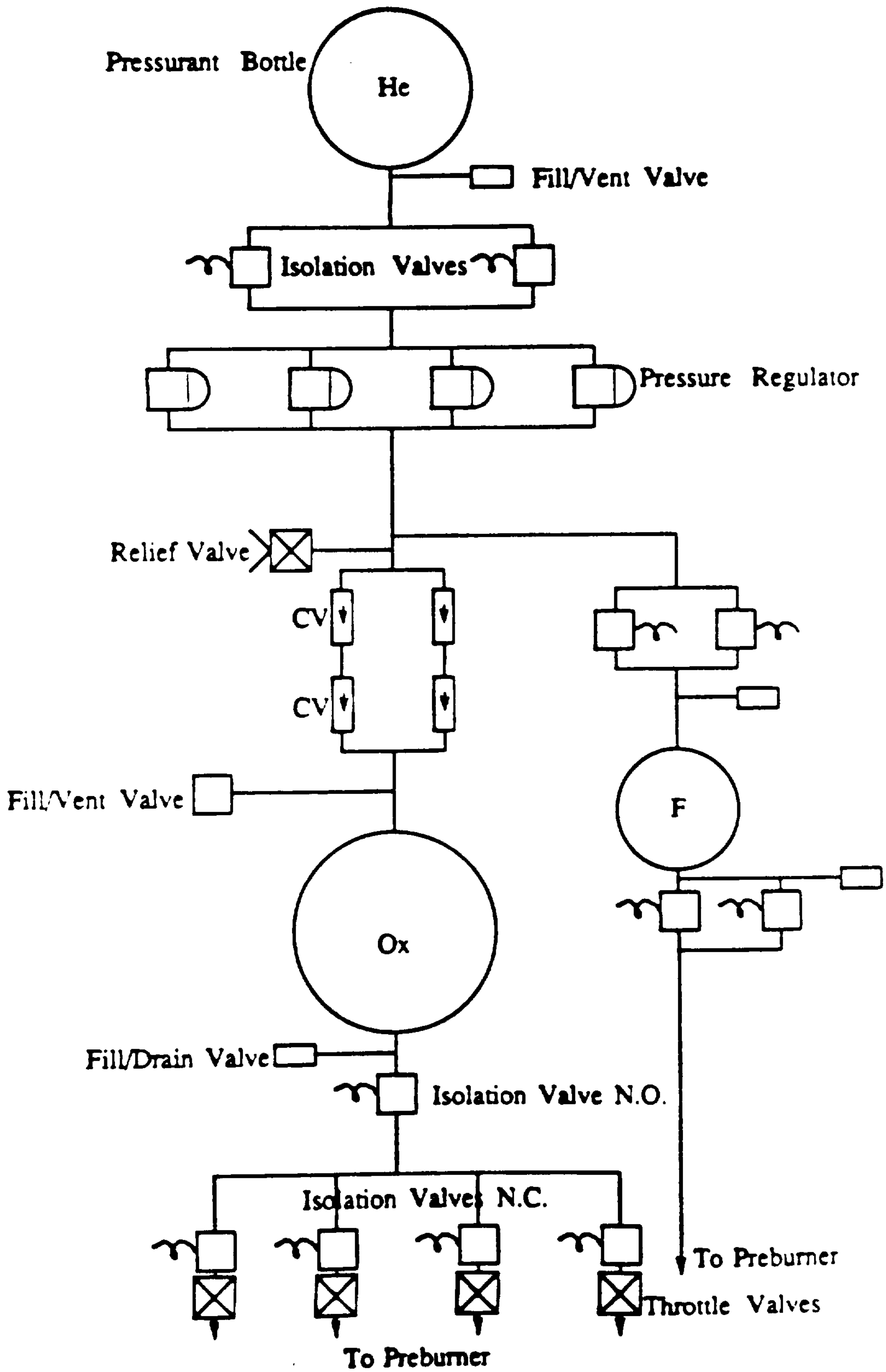

Figure A-5. Cold Gas Pressurization System Schematic for HC/LOX.

Table A-10. Cold Regulated Pressurization (Tridyne)

a. GG/LOX Hybrid

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Used KG	Bottle Length CM
34.5	9,901	5,221	25,550	1607
51.7	8,063	5,218	22,596	998
68.9	7,340	5,216	22,026	766
86.1	6,945	5,215	22,098	644

b. HC/LOX Hybrid

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Used KG	Bottle Length CM
34.5	12,587	6,561	32,191	2018
51.7	10,203	6,558	28,229	1239
68.9	9,272	6,555	27,410	945
86.1	8,765	6,554	27,395	789

Table A-11. Warm Gas Regulated Pressurization (Tridyne)

a. GG/LOX Hybrid

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Used KG	Bottle Length CM
34.5	5,056	2,527	10,546	728
51.7	4,002	2,519	9,224	457
68.9	3,601	2,513	9,001	358
86.1	3,385	2,508	9,058	306

b. HC/LOX Hybrid

P_{BO} MPa	W_{He} Loaded KG	W_{He} Used KG	W_{He} Used KG	Bottle Length CM
34.5	6,477	3,176	13,285	907
51.7	5,084	3,165	11,446	556
68.9	4,560	3,157	11,084	429
86.1	4,280	3,150	11,108	363

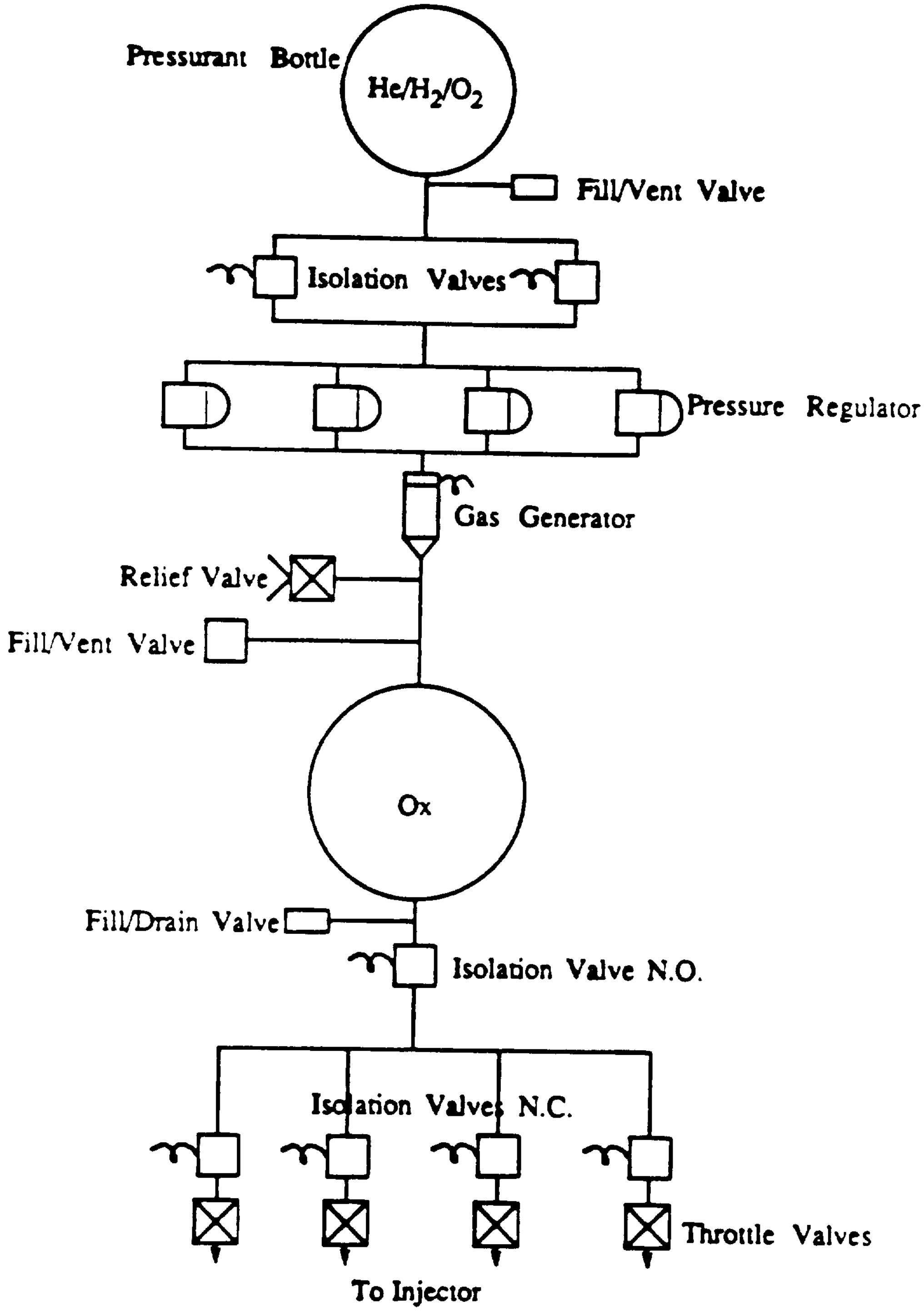

Figure A-6. Catalytic Warm Gas Pressurization System Schematic for GG/LOX.

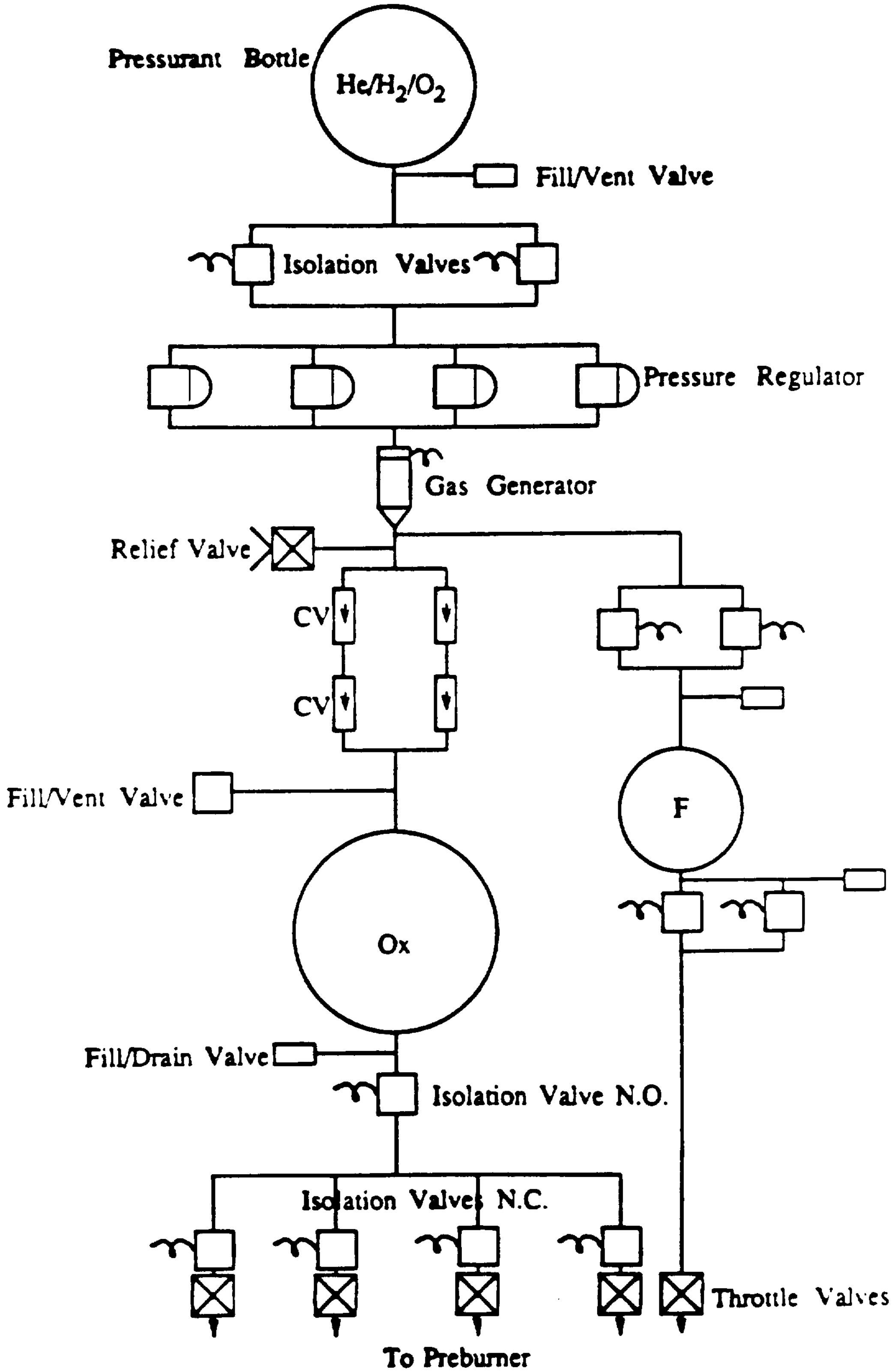

Figure A-7. Catalytic Warm Gas Pressurization System Schematic for HC/LOX.

Table A-12. Oxidizer and Fuel Tanks with Cold Gas Regulated Pressurization

System	$W_{Tank\ Ox}$ KG	$L_{Tank\ Ox}$ CM	$W_{Tank\ F}$ KG
GG/LOX	10,026	1,958	0
HC/LOX	12,128	2,309	679

Table A-13. Oxidizer and Fuel Tanks with Warm Gas Regulated Pressurization (He/H2/02)

System	$W_{Tank\ Ox}$ KG	$L_{Tank\ Ox}$ CM	$W_{Tank\ F}$ KG
GG/LOX	10,885	1,962	0
HC/LOX29,223	413	4,107	52

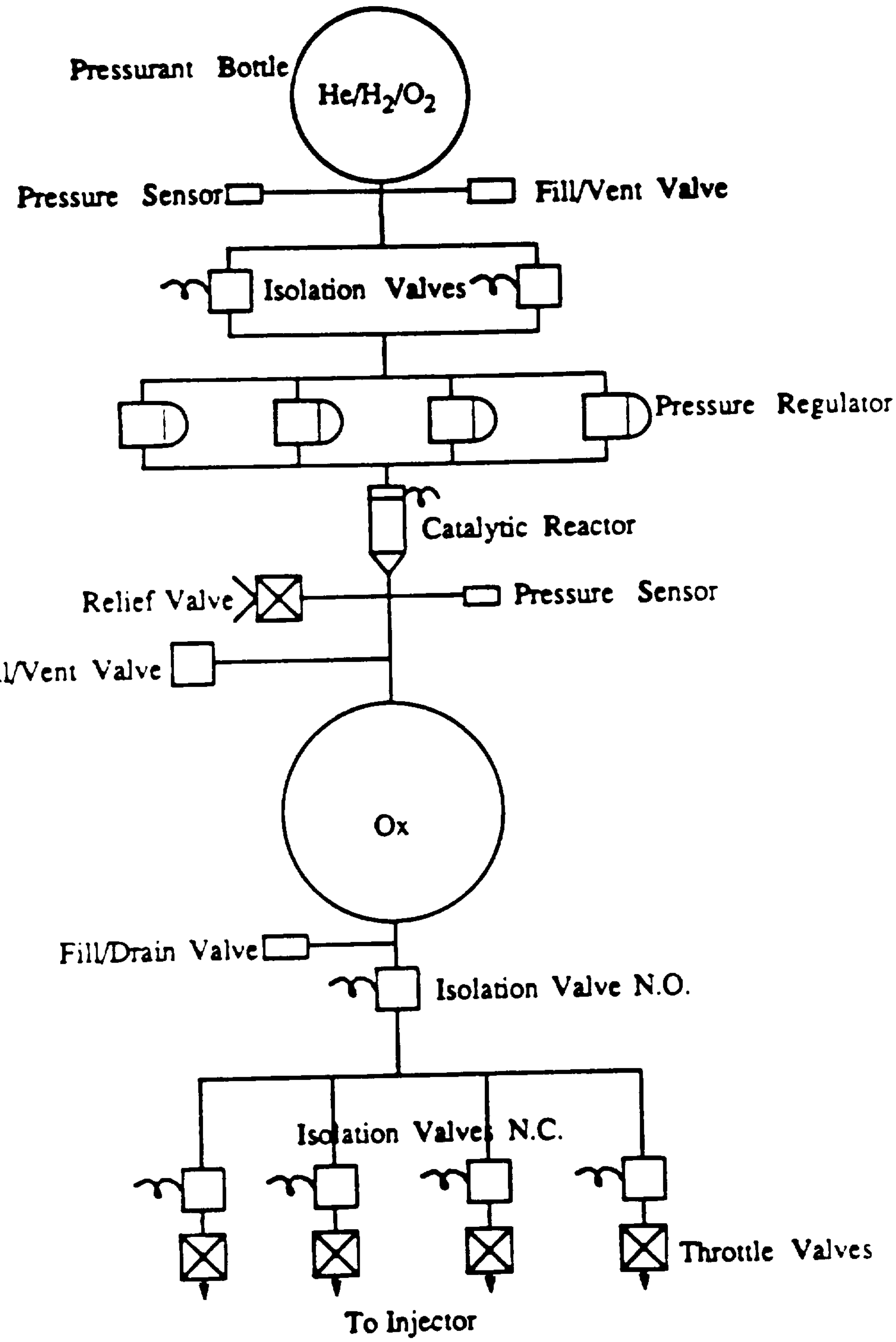

Figure A-8. Catalytic Warm Gas Pressurization System Schematic for GG/LOX.

The Tridyne pressurant consists of a small fraction of reactive gases, (H_2 and O_2), which are combined with an inert diluent (helium) to form a nondetonable mixture that can safely be stored at high pressure in a bottle. Energy release is accomplished by passing the mixture through a catalyst bed which combines the reactants and creates a hot gaseous mixture. Gas temperature is controlled by varying the reactant concentration.

The weight of Tridyne pressurant is dependent upon the volume to be pressurized, the storage pressure, the final blowdown pressure, the final gas temperature in the tank and the catalytic reaction temperature rise. A nominal 68.9 MPa (10,000 psia) was selected as a reasonable compromise between weight, bottle size, and safety concerns. As Figure A-9 shows, the weight savings of going above 68.9 MPa (10,000 psia) are minimal. The nominal bottle pressure of 68.9 MPa (10,000 psia) is well within the demonstrated design capability of composite-wrapped tanks.

The selected Tridyne molar composition of 0.91 He/0.06 H_2/0.03 O_2 corresponds to a theoretical reaction temperature of 983K (1,770°R) at an inlet temperature of 554K (997°R). The respective mass composition is 0.7711/0.2033/0.0256.

Tridyne is supplied at regulated pressure to the catalytic reactor where the oxygen and hydrogen are combined to convert the cold Tridyne to a heated mixture of helium and water vapor. The catalyst, designated DEOXO MFSA by Engelhard Industries, consists of platinum-group metals on the surface of aluminum oxide spheres contained in a cylindrical housing with drilled end plates. A 300 series stainless steel wire screen prevents the catalyst from obstructing or migrating through the holes of the plates. Injection orifices are used to evenly distribute the gas flow and prevent channeling within the catalyst bed. The outer shell of the reactor is also made of 300 series stainless steel. A maximum wall temperature of 900°F is expected for the mission.

The coldest temperature at the inlet to the catalytic reactor is 235K (424°R) which corresponds to expansion from 68.9 MPa to 14.4 MPa (10,000 psia to 2,084 psia) with a polytropic exponent of 1.15. This results in a drop of reaction temperature of approximately 61K (110°R). Using the empirical

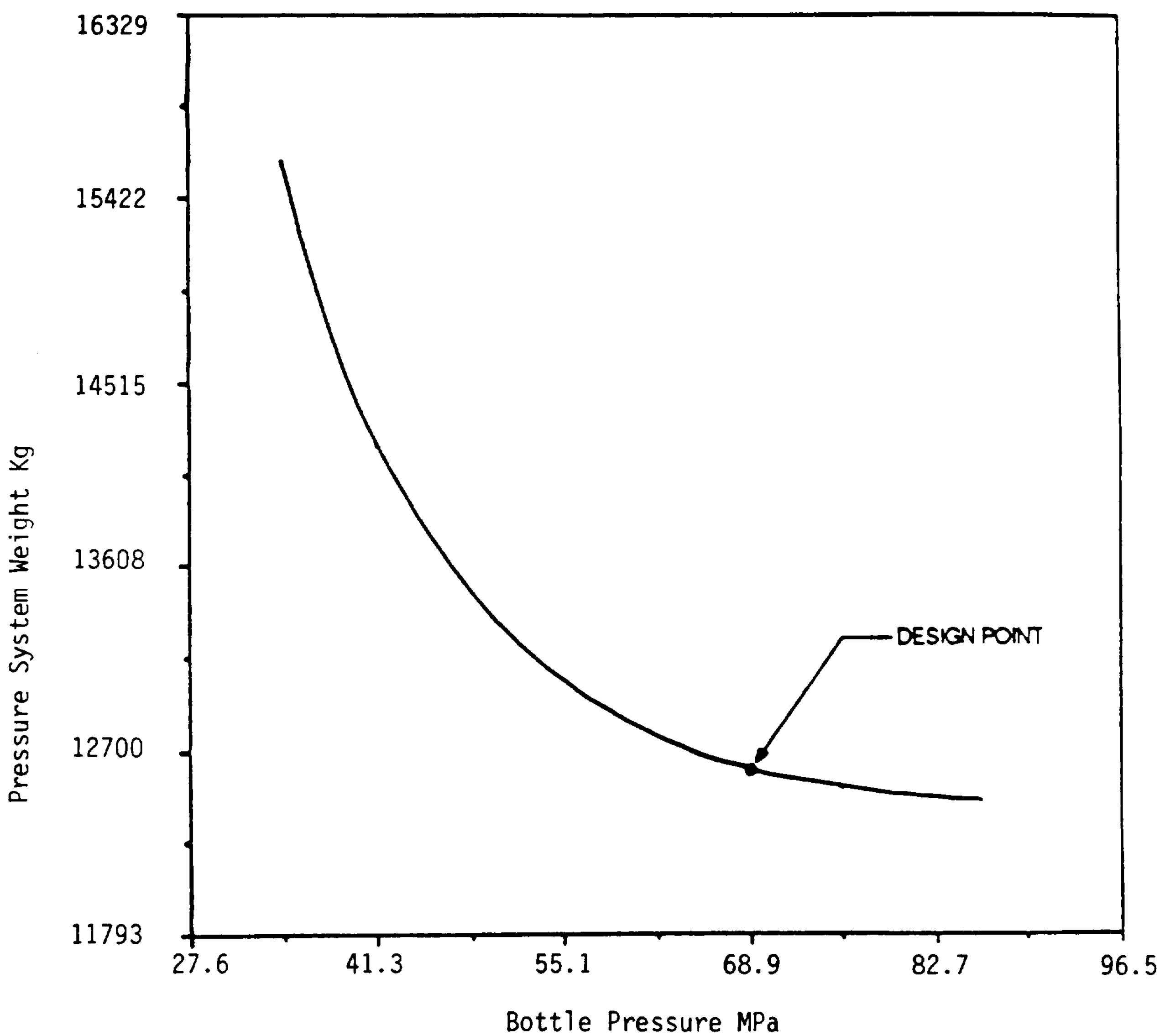

Figure 9. GG/LOX Hybrid System Weight vs Bottle Pressure.

Epstein equation, an equilibrium gas temperature of approximately 667K (1,200°R) in the oxidizer tank was determined.[1]

A fill/vent valve is located at the bottle outlet. A pressure sensor is also located at the outlet to the bottle to provide continuous monitoring of the storage pressure. Two redundant isolation valves provide containment of the pressurant gas during storage and ensure activation of the pressurization system upon signal. These valves are normally closed, explosively-actuated units.

Tank pressure control is provided by four pressure regulators. Each regulator is sized to provide one-third of the maximum expected flow rate. In the unlikely event a regulator should fail closed, the three remaining regulators can handle the maximum flow demand. A relief valve is located downstream of the regulators so that if a regulator should fail open, the tank will not be overpressurized. A pressure sensor downstream of the regulators monitors the tank pressure and will indicate before launch if a regulator has failed open. A regulator failure (open) during flight should not affect the mission because total flow demand from the engine does not decrease below one-third of maximum flow rate until tailoff. The regulators have a filter and slam suppressor built into the inlet. The slam suppressors prevent overheating of the valve seat during system enable.

Propellant Tank - The oxidizer tank is constructed of a filament wound IM-7 fiber impregnated with EPON 826 resin. A liner consisting of Teflon and insulation is located inside the tank. The total tank volume is $254 m^3$ (8,962 ft^3), plus a 3 percent ullage allowance. The case has been sized to handle a maximum tank pressure of 12.4 MPa (1,793 psia) during a no-flow condition. The burst safety factor is 1.6.

Liquid Feed System - The liquid feed system has been shown schematically in Figure A-8. The oxidizer will be fed to the liquid injector via four 20.3 centimeter (8-inch) diameter stainless steel feed lines. Each manifold will have a liquid throttling valve immediately upstream of the injector to

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1. Epstein, M., Georgius, H. K., and Anderson, R. E., "A Generalized Propellant Tank-Pressurization Analysis," Advances in Cryogenic Engineering, Volume 10B, Plenum Press, New York (1965), Page 290.

control the flow rate of the LOX. Throttling pintle-type valves, operated by hydraulic-mechanical actuators, are used. Upstream of this valve an explosively actuated isolation valve is located to provide double containment. When the oxidizer tank is filled, LOX will be bled down to the isolation valve.

The liquid feed system activation sequence is:

1. Fill/vent and fill/drain valves on the LOX tank are closed.

2. LOX feedline isolation valves are opened. This will fill the feedline to the throttle valve and will minimize the water hammer the throttle valve will see.

3. Gas feedline isolation valves are opened.

The gas feed lines will fill to the regulators. The regulator will flow full open until such time as the downstream regulated pressure is reached. The catalyst bed of the catalytic reactor will warm up. Tank pressure will be monitored to ensure that after regulated pressure is reached, no regulator has failed in an open condition resulting in a pressure rise and relief valve opening.

Pressure Schedule

The pressure schedule at the maximum chamber pressure condition is presented in Table A-14.

The acceleration head (due to long feed lines and large oxidizer tank) present at the throttling valve has not been considered in sizing the pressurization system. Figure A-10 shows how the acceleration head varies with vehicle acceleration levels and oxidizer use. Figure A-10 also shows how the feed line pressure drop varies with oxidizer flow rate. The net acceleration head available during the mission is shown in Figure A-11; a minimum of 0.2 MPa (35 psi) is available for pressurization, that the pressurization subsystem will not have to supply. This results in a decrease of subsystem weight of approximately 472kg (1,040 lbs).

System/Component Weights

The pressurization and oxidizer delivery subsystem weight breakdown is presented in Table A-15.

Table 14. Pressure Schedule

	Pressure (MPa)
Chamber Pressure (Max.)	7.5
Injector Drop	2.2
Valve Drop	0.3
Isovalve Drop	0.03
Manifold Drop	0.4
Tank Liquid Pressure	10.4
Max. Tank Pressure - No flow	12.4
Manifold Drop	0.04
Catalytic Reactor	0.99
Regulator Outlet, Nom.	11.4 + 8%
Regulator Inlet, Min.	14.2
Manifold Drop	0.1
Isolvalve Drop	0.03
Blowdown Pressure, Max.	14.4
Initial Pressure, Min.	68.9 @ 289K
Initial Pressure, Max.	73.9 @ 311K

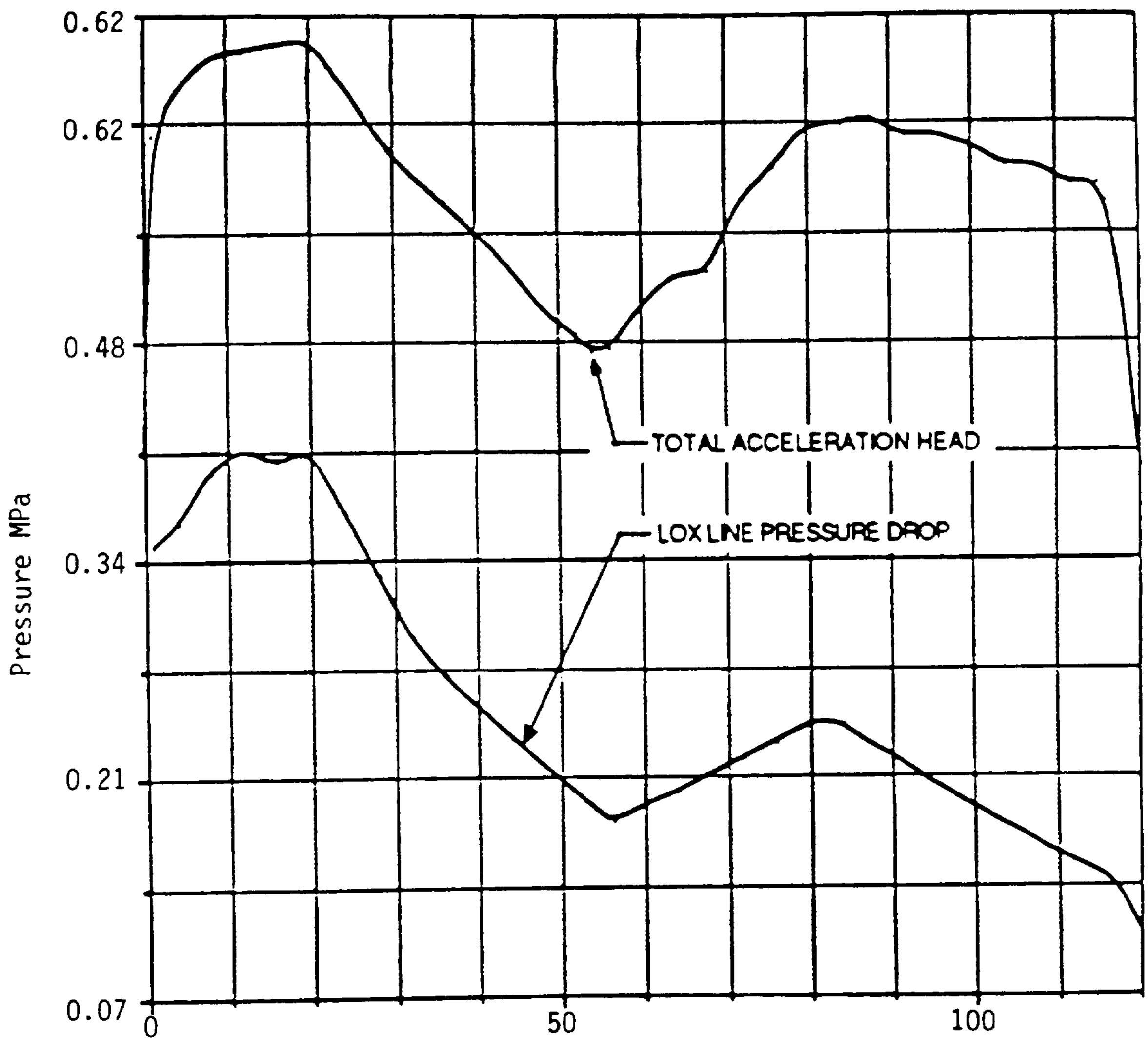

Figure A-10. GG/LOX Hybrid Acceleration Head and LOX Line Pressure Drop.

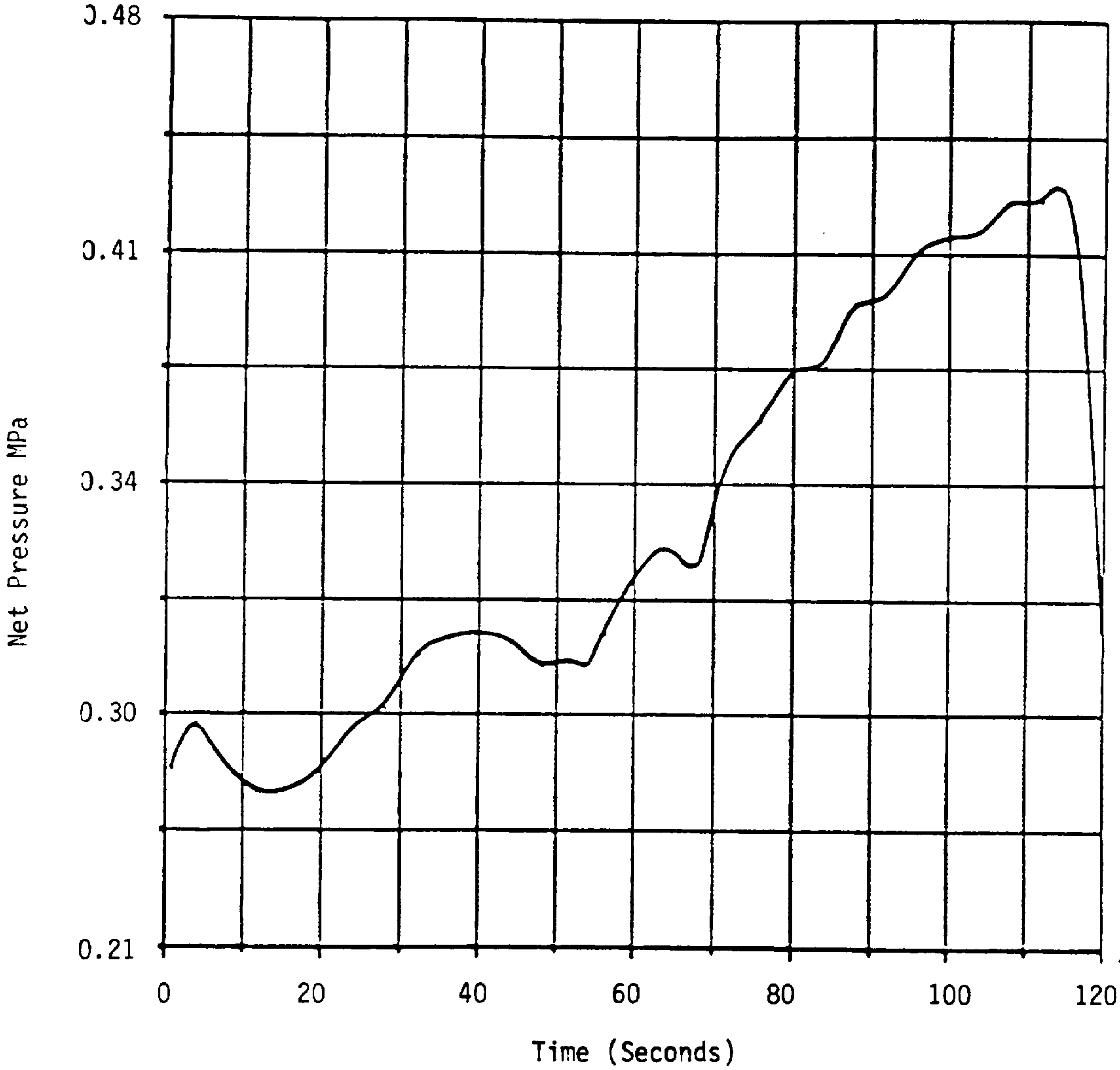

Figure A-11. GG/LOX Hybrid Net Pressure Head.

Table A-15. System Weight Breakdown

Pressurizing System	Weight (KG)
Bottle	9,001
Fill/Vent Valve	0.9
Isolation Valve (2)	9.1
Gas Manifold	7.3
Regulator (4)	83.5
Relief Valve	4.5
Catalytic Reactor	298.5
Tridyne Gas	3,600.6
	13,005.4

Oxidizer Delivery System	Weight (KG)
Tank	10,885.3
Fill/Vent Valve	0.9
Fill/Drain Valve	0.9
Isolation Valve (N.O.)	90.7
Liquid Manifold (4)	1,138.5
Isolation Valve (N.C.) (4)	90.7
Throttling Valve (4)	110.7
	12,317.8

PUMP FED SYSTEMS

Introduction

The main thrusts of the pump fed system study involved the following:

- Investigation of candidate pressurization systems to provide a relatively low tank pressure, reliably, safely and at minimum cost.
- Accomplish sufficient pump studies to determine that a pump could be designed to accommodate the required throttling and determine the most efficient way to throttle the flow.

Pressurization System

The pump fed system requires a pressurant to provide a small "head" pressure to the oxidizer tank. The basic types of pressurization systems considered were essentially the same as those investigated during the pressure fed system studies. A summary of these systems and their advantages and disadvantages are summarized in Table A-16. This table is applicable to both oxidizers. Some approaches were eliminated for safety concerns; This is particularly true for 95-percent H_2O_2. Although H_2O_2 was eventually eliminated as the oxidizer, the work accomplished is reported.

Pressurization Systems for H_2O_2

The pressurization systems shown in Table A-17 were investigated. All of the systems were conceptually designed so that a single point failure would not cause loss of mission. All of the pressurization system schemes were assessed to meet reliability and safety requirements. The actual selection process was made on a cost and weight basis.

The autogenous system was eventually rejected because of cost and complexity. Although the solid-grain-augmented cold gas system basically costs the same and is slightly lower weight than the stored gas system, stored gas was finally selected because of its larger historical data base and slightly higher reliability. In addition, the higher temperature gases are not as compatible as cold gas. In reality, these systems would probably be safe. The problem arises from the fact that when 95-percent H_2O_2 gets to about 250°F, auto thermal decomposition can result unless closely controlled and an overpressurization of the tank could occur. The feature that tends to make this concept safe, even though some of the hydrogen peroxide is

Table A-16. Pressurization System Concepts Summary

CONCEPT	ADVANTAGES	DISADVANTAGES
Stored Gas (Helium)	Simple, low cost, high reliability.	Heavy, large volume (particularly for cryogenic propellants).
Stored Gas (Solid Propellant Grain Augmented)	Lower Weight & Volume than Cold Gas	Good for cyrogenics, must be careful when pressurizing 95% H_2O_2.
Stored Gas (Tridyne)	Lower Weight & Volume than any of the stored gas systems.	Slightly more complex due to catalytic G.G. Care must be taken when using with 95% H_2O_2.
Stored Gas/Heat Exchanger	Lower weight & volume than stored cold gas.	Requires heat source, heat exchanger and associated controls.
Solid Propellant Grain	Simple, Lowest Volume, Low Cost	Variable ox flow makes this concept basically unworkable. Would need a very low temp grain for use with 95% H_2O_2.
Main Tank Injection	Pressurant stored as liquid, low weight and volume.	Thermal reaction not good for use with 95% N_2O_2. Complex controls required to prevent overpressurization caused by throttling.
Autogenous	No pressurant other than additional oxidizer required. Low volume and weight.	Dependent upon turbine outlet conditions which vary as a result of throttling. Controls could be complex. May not be good for pressurizing 95% H_2O_2.

decomposing, is that the tank ullage volume is constantly increasing due to oxidizer usage. The rate of ullage increase should be much greater than the rate of gas being produced by the decomposition of the hydrogen peroxide.

Stored Gas (Helium) - A schematic diagram showing the stored helium gas concept is presented in Figure A-12. It uses redundant components so a single point failure would not cause a failure of a mission. One example is the use of two regulators. During normal operation, only one regulator will be utilized. A health monitoring system will be used to detect a regulator failure whereby an isolation valve will lock out the malfunctioning regulator and open an isolation valve to allow the redundant regulator to come on line.

The amount of pressurant required is a function of oxidizer tank volume, oxidizer tank pressure, oxidizer temperature, and pressurant temperature. The largest portion of the pressurization system weight is from the quantity of helium and the helium bottle weight. The helium bottle weight is a function of the quantity of gas and the storage pressure and temperature. Reliability is a function of redundancy, number of components, and database of the components. Safety (ground and flight operation) is a function of components or processes that can malfunction in a worst case scenario.

Autogenous System - This system is shown schematically in Figure A-13. In this concept, the turbines which drive the pumps are designed to have an output pressure to accommodate the required tank pressure. The system is simplified since a separate gas source is not required. The oxidizer is used catalytically, in this case to drive the turbine, and as its own pressurant. A relief valve is used to prevent overpressurization of the oxidizer tank. The static head of oxidizer in the tank, as well as a solid charge to spin up the turbines, should be sufficient to start the system. A precaution must be taken to minimize the turbine outlet temperature to the predetermined maximum. Also, the design of the system must be sufficiently flexible to accommodate a range of turbine outlet conditions due to the pump throttling requirement.

Stored Gas (Solid Grain Added) - A schematic of the concept is shown in Figure A-14. It is identical to that of the cold gas concept except for the addition of a solid charge which will be used part way through the mission to provide additional gas to heat up the remaining helium. The advantage of this system is the gas storage vessel is smaller than the stored cold gas because

Table A-17. Comparison of Selected H202 Pressurization Concepts

SYSTEM CONCEPT	WEIGHT*	COST INDEX	REMARKS
Stored Gas (Helium)	2616	100	1st - Simple & low cost.
Autogenous (Turbine Discharge)	1860	115	High Complexity (cost).
Stored Gas (Solid Grain Augmented)	2559	105	A little more complex than plain stored gas.

* Weight in kilograms.

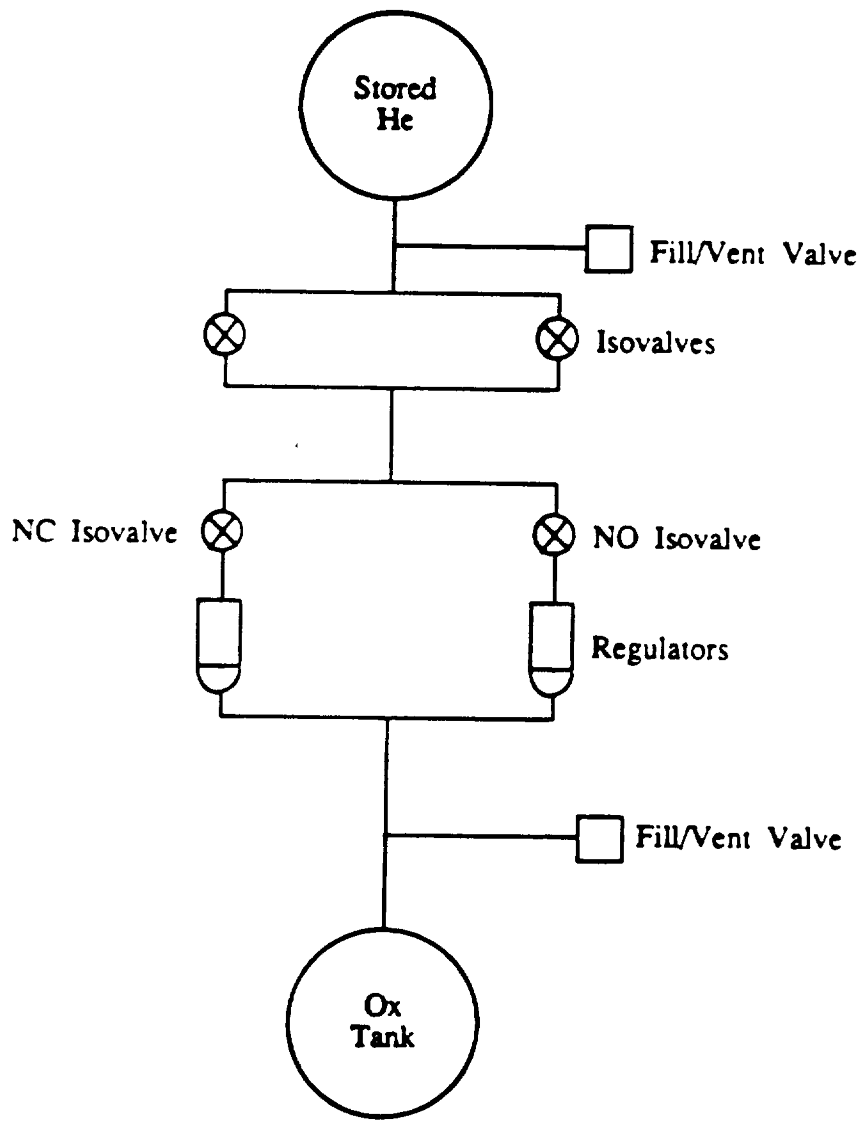

Figure A-12. Stored Helium Gas Pressurization System.

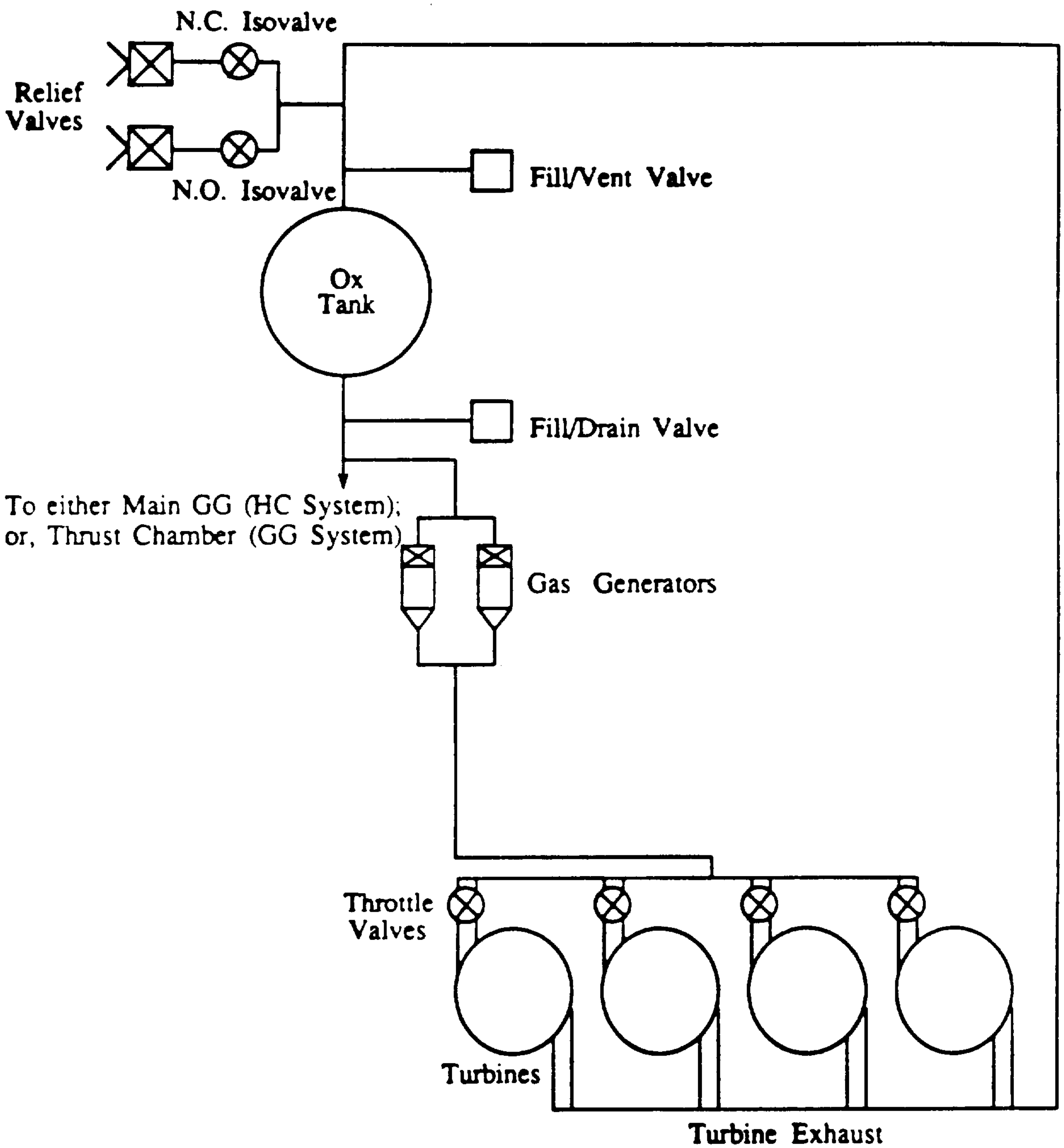

Figure A-13. Autogeneous Pressurization System.

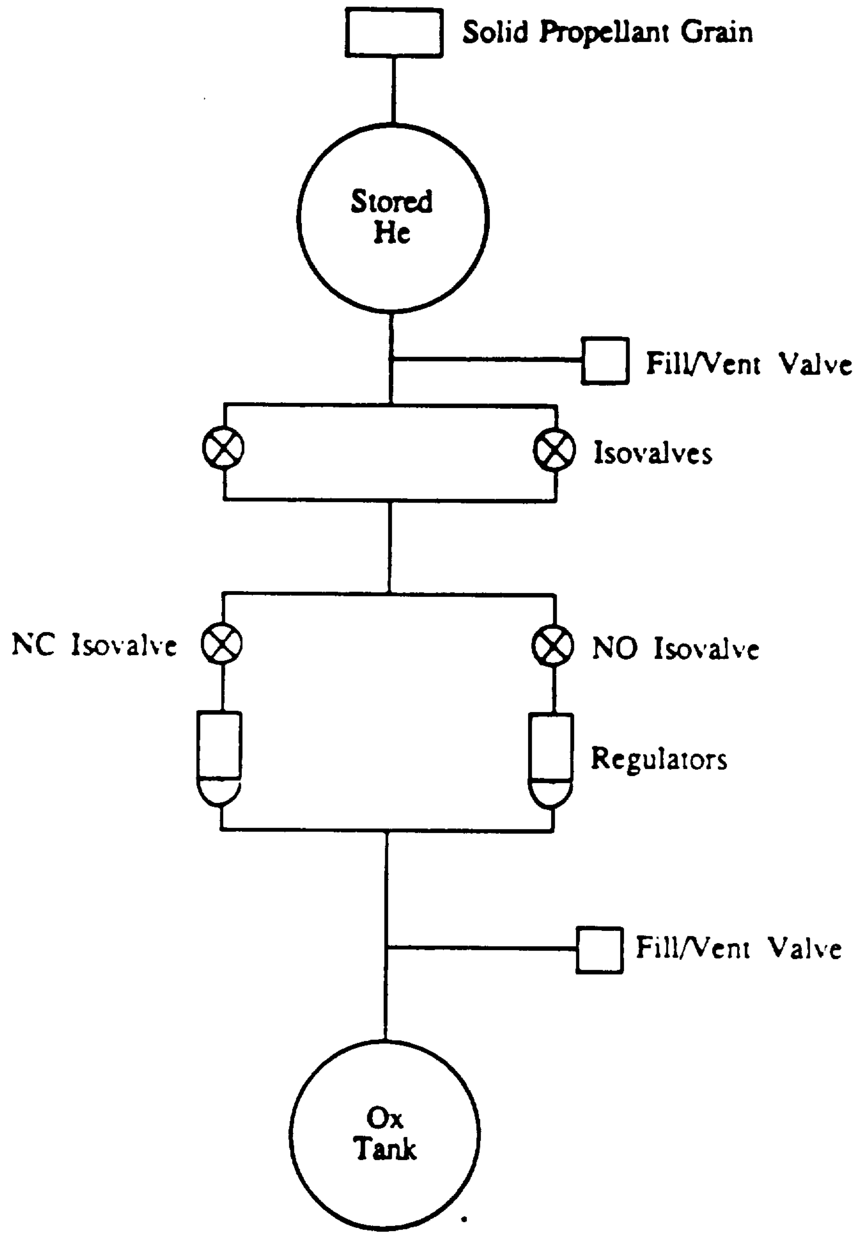

Figure A-14. Stored Gas (Solid Grain Augmented).

there is less gas to store. However, additional complexity is added because a solid propellant grain and ignitor and associated wiring is required. Accidental early ignition of the solid charge could result in overpressurization of the helium bottle. In addition, a filter is required prior to entry of the gas into the regulators because of debris from the solid. The temperature to which the remaining gas is heated can be easily controlled by the type of propellant grain utilized.

Pressurization Systems for LOX

Unlike hydrogen peroxide, LOX is not a monopropellant, therefore, more pressurization concepts are applicable than with hydrogen peroxide. LOX, however, is a cryogen, therefore, any pressurant coming into the tank will be cooled; thus more pressurant is required to maintain a given tank pressure. Systems selected for study are discussed below.

Stored Gas (Helium) - This is the same pressurization as shown in Figure A-12 except that the oxidizer tank contains LOX instead of H_2O_2. The results of the analysis are summarized in Table A-18.

Stored Gas (Solid Grain Augmented) - This system is identical to that discussed previously in Figure A-14. The results are presented in Table A-18.

Autogenous System - This system concept is presented in Figure A-15. There are two possible sources to power the turbine. The first taps combustion gases (shown in schematic) from the fuel-rich solid gas generator, and the second uses separate gas generators to drive the turbines. The latter uses a separate fuel (i.e., methane) to react with LOX to generate the gases to drive the turbines. This complicates the system and increases cost. Tapping fuel-rich gases from the solid gas generator to drive the turbines and then using the turbine outlet gas to pressurize the tank is viable, providing that the reaction between the fuel-rich turbine exhaust and the LOX can be readily controlled. Again, the problem of changing turbine outlet conditions could make this concept difficult to achieve. The results are shown in Table A-18.

Stored Reactive Gas (Tridyne) - This concept is shown schematically in Figure A-16. This system was previously described in the pressure-fed appendix of this report. The results are presented in Table A-18.

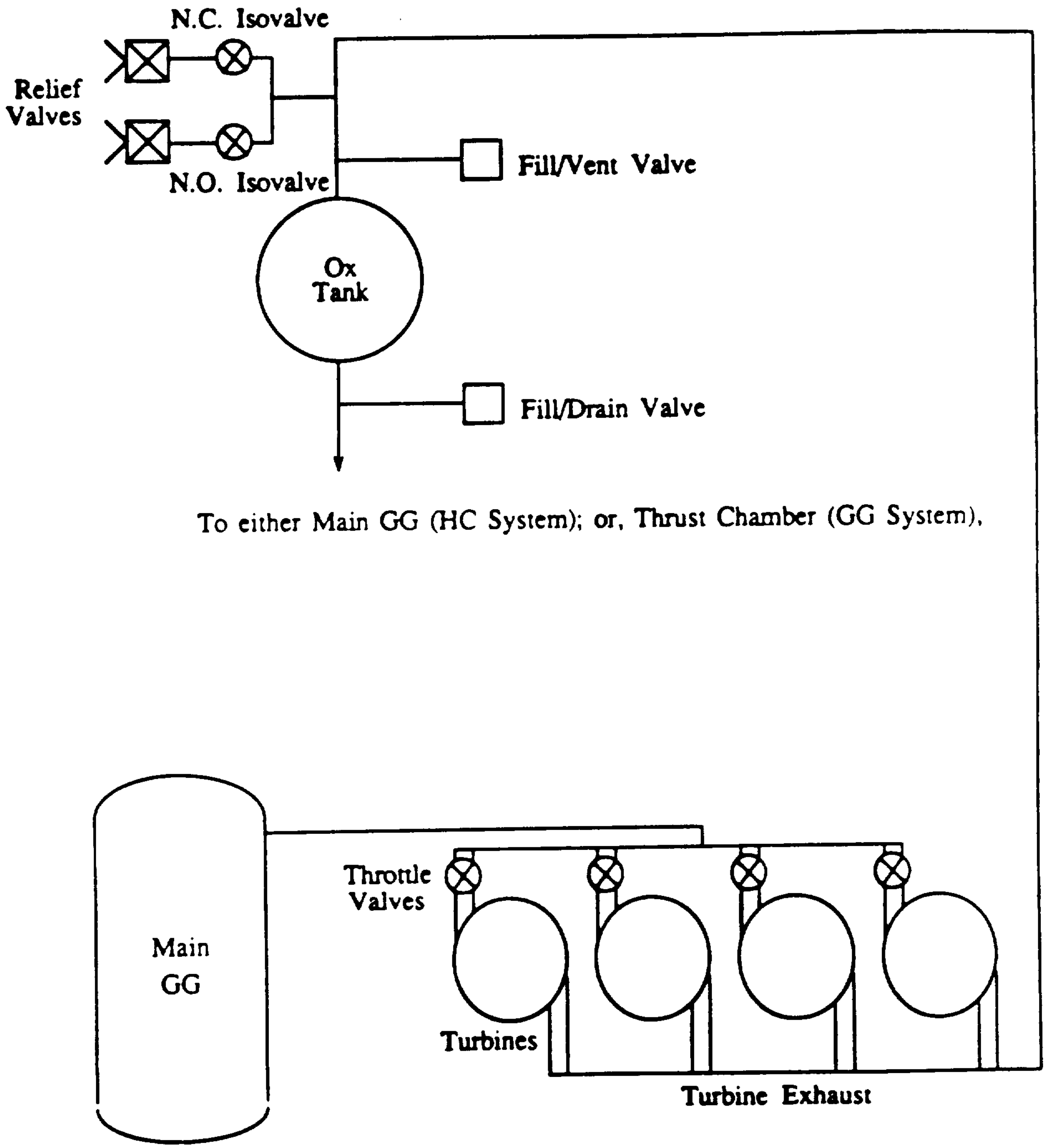

Figure A-15. Autogeneous Pressurization System.

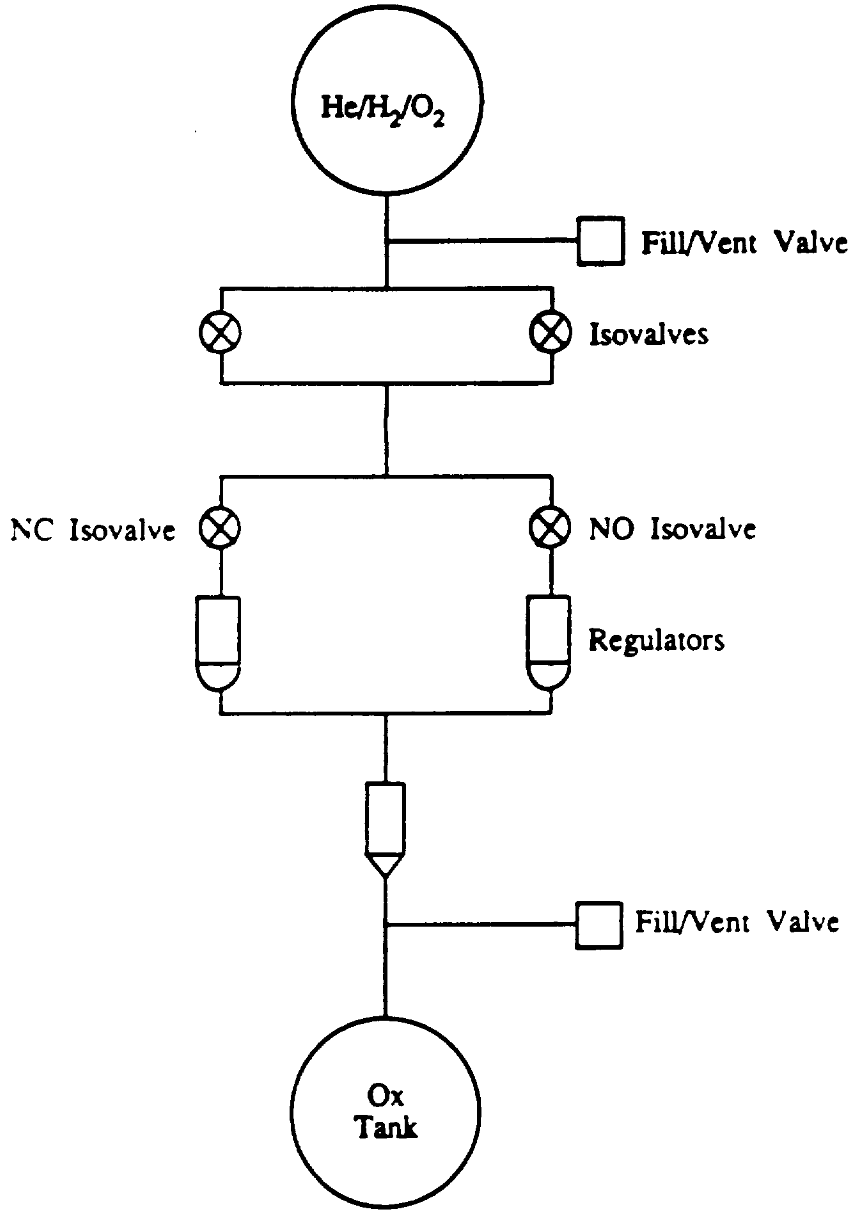

Figure A-16. Stored Reactive Gas (Tridyne) Pressurization System.

Table A-18. Comparison of Selected LOX Pressurization Concepts.

SYSTEM CONCEPT	WEIGHT*	COST INDEX	REMARKS
Stored Gas (Helium	2495	100	Largest volume - low cost.
Stored Gas (Solid Grain than plain stored Augmented)	2445	105	Slightly smaller volume gas. Also low cost.
Autogenous	3267	115	Complexity increases cost.
Stored Gas (Tridyne)	1124	105	Very low weight in conjunction with relatively low cost makes this the logical choice.

* Weight shown in kilograms.

Turbopump

The basic objectives of this subtask were:

1. Determine the feasibility of developing a turbopump capable of operating over the required throttle range.
2. Determine the most effective way to drive the turbine and where to pipe the turbine exhaust.
3. Estimate the weight and cost of a turbopump capable of meeting the overall requirements.
4. Provide an overview program schedule for turbopump development.

From a reliability viewpoint, it was determined that more than one turbopump would be required. It was also determined that these pumps would have to operate simultaneously, since in the case of two turbopumps, there would be insufficient time to get the second pump up to speed if the first pump failed, particularly just after lift-off. Since the pumps must throttle over a 1.6:1 range during normal operation, they must be capable of throttling over a wider In a 3-pump configuration, under a pump-out condition, each pump would have to operate over a 2.4:1 throttle range. In a 4-pump configuration, with one out, the design operating throttle range is 2.13:1. The 4-pump configuration is easier to accommodate, but would result in a higher life cycle cost.

Another issue was to determine the most effective way to drive the turbine. This decision will have an impact on turbopump development costs, unit production cost, as well as overall oxidizer feed system weight.

No attempt was made at this point to design a turbopump. Sufficient design studies were accomplished, however, to get an overview assessment of the degree of difficulty in accomplishing the development of a turbopump capable of satisfying system requirements.

Turbopump Requirements - The general requirements for the LOX turbopumps are presented in Table A-19.

A preliminary design study was conducted by two pump manufacturers. Acurex Corporation provided preliminary information pertaining to a pump configuration using separate gas generators to drive the turbines. This does not necessarily represent their selected approach, but was provided to evaluate one way to drive the turbine. AiResearch Division of Allied Signal provided information pertaining to a configuration using gases tapped off the

Table A-19. LOX Turbopump Operating Requirements

Maximum Flow Rate	3144 KG/sec
Pump Inlet Pressure	0.8 MPa
Pump Outlet Pressure	9.5 MPa (11.5 MPa[1])
Turbine Drive-Gas Flow Molecular Weight[2]	13.75
Turbine Drive-Gas Ratio of Specific Heats[2]	1.12
Turbine Inlet Pressure (Main GG Maximum Chamber Pressure)	7.5 MPa
Turbine Inlet Temperature (GG Chamber Temperature)	392K
Turbine Discharge Pressure	0.2 MPa
Minimum Flow Rate	1895 KG/sec
Minimum Chamber Pressure	3.5 MPa

Four Turbopumps with Single Pump Out Capability

1. Regenerative cooling version.
2. Assuming solids are filtered out using reverse pitot.

main solid gas generator to drive the turbines. The Allied Signal turbopump was selected for further evaluation.

Acurex Turbopump Configuration - The primary objective was to accomplish sufficient preliminary design work to establish a design concept that would show the feasibility of the approach, to predict weight, cost and establish a preliminary turbopump development plan. Characteristics of the Acurex main pump and booster pump are shown in Table A-20. Turbine and gas generator characteristics are presented in Tables A-21 and A-22, respectively.

The feed system will utilize 4 parallel turbopumps to feed a single thrust chamber. Each turbopump is driven by a gas generator. The turbopumps were sized such that three turbopumps could provide the design flow rate. This capability greatly improves the reliability of the overall feed system.

The basic design of the turbopump is conventional and within the state-of-the-art. Oxidizer-rich and fuel-rich gas generators were evaluated. An oxidizer-rich turbine drive does not require a positive shaft seal between the turbine and the pump. This is a great simplification and impacts inherent safety, reliability and cost of the turbopump. For this reason, the oxidizer-rich gas generator is favored to drive the turbopump.

Both the oxidizer-rich and fuel-rich gas generators are similar in size. The total flow rate for the oxidizer-rich gas generator is about three times that of the fuel-rich gas generator. However, the flow rate of methane is much less for the oxidizer-rich gas generator. A small quantity of methane means a small methane tank which is lower in weight and requires less pressurant than a fuel-rich gas generator. In this particular case, an oxidizer rich gas generator was selected primarily because it simplified the design and construction of the turbopump while increasing the inherent safety, reliability, and decreasing cost.

It should also be noted that the characteristics of the fuel-rich gas generator products of combustion are similar to those from the main solid gas generator, except for solids content. Therefore, the design of the turbine using the fuel-rich gas generator would be essentially the same for a turbine using gases tapped off the main solid gas generator.

AiResearch Turbopump Configuration - This concept uses gases tapped off the solid fuel-rich gas generator to drive the turbines. The turbopump

Table A-20. Acurex Pump Characteristics (4 Pump Configuration)

Total Flow Rate to Thrust Chamber, KG/sec.......	3144
Number of Pumps.................................	3 (1 pump out)
Flow Rate per Pump, KG/sec......................	1048
GG Ox Flow, KG/sec..............................	65
Total Pump Ox Flow, KG/sec......................	1113
Suction Pressure, MPa...........................	0.4
Vapor Pressure, MPa.............................	0.1
NPSH, MPa.......................................	0.3
NPSH, M...	30
Suction Specific Speed, Boost Pump..............	20,000
Suction Specific Speed, Main Pump...............	8,000
Main Stage Shaft Speed, rpm.....................	8,000
Boost Pump Speed, rpm...........................	5,000
Boost Pump Pressure, MPa........................	1.7
M..........................	150
Main Pump Pressure Rise, MPa....................	9.9
M......................	879
Discharge Pressure, MPa.........................	11.5
Specific Speed Boost Pump.......................	5748
Specific Speed Main Pump........................	2449
Fluid Power Boost Pump, hp......................	2075
Efficiency Boost Pump...........................	0.66
Shaft Horsepower Boost Pump, hp.................	3145
Fluid Power Main Pump, hp.......................	12,116
Efficiency Main Pump............................	0.85
Shaft Horsepower Main Pump, hp..................	14,255
Tip Speed, Main Pump, mps.......................	131
Head Coefficient, Main Pump.....................	0.5
Impeller Diameter, Main Pump, mps...............	86
Impeller Diameter, Boost Pump, cm...............	33
Suction Diameter, Boost Pump, cm................	38

Table A-21. Acurex Turbine Characteristics

HIGH SPEED

Type	Axial Flow, 2 Stage	Axial Flow, 3 Stage
Speed, rpm	8,000	8,000
Power, hp	14,255	14,255
Inlet Pressure, MPa	6.89	6.89
Flow Rate, KG/sec	66	66
Temperature, K	556	867
Tip Diameter, CM	46	46
Blade Height, 1st Stage, CM	1.5	1.5*
Tip Speed, MPS	192	192
Stage V/C_o	0.35	0.22
Efficiency	0.7	0.4

LOW SPEED

Type	Axial Flow, 1 Stage	Axial Flow, 2 Stage
Speed, rpm	5,000	5,000
Power, hp	3,145	3,145
Flow Rate, KG/sec	66	21
Exit Pressure, MPa	0.4	0.4
Exit Gas Temperature, K	389	494
Exit Gas Density, KG/M^3	4.0	1.9
Tip Speed, MPS	120	120
Blade Height, CM	10.2	7.6

* Partial Admission

Table A-22. Acurex Gas Generator Characteristics

	OX-RICH	FUEL-RICH
Propellants	LOX/Methane	LOX/Methane
Mixture Ratio, O/F	47	0.6
Temperature, K	556	867
Pressure, MPa	6.89	7.1
Flow Rate, KG/sec	66	21
Oxidizer Flow Rate, KG/sec	65	7.7
Fuel Flow Rate, KG/sec	1.4	11.8
Throat Area, CM^2	57.4	36.1
Throat Diameter, CM	8.6	6.9
Characteristic Length, CM	152	152
Chamber Volume, CM^3	8758	5510
Diameter, CM	16.5	12.7
Length, CM	48	43

characteristics are summarized in Table A-23. The gas generator must be modified to accommodate the added requirement dictated by the turbine flow. This autogenous turbine drive should theoretically be the lowest cost approach for driving the turbine since additional fluids, gas generators, storage containers, etc., are not required. However, an efficient method must be found to separate the solids out of the gas stream.

Another problem associated with using the solid gas generator for driving the turbines is that the pressure and flow rate of the gas generator change throughout the flight. Four points were taken from the flight profile, and the resulting turbopump characteristics for these conditions are shown in Table A-24.

Turbine Discharge - There are basically four choices available as to what to do with the gases coming out of the turbine.

1. Exhaust the gases to ambient through a separate nozzle or thrust chamber.
2. Use all or part of the exhaust to pressure the LOX tank.
3. Use part of the exhaust for thrust vector control.
4. Use all or part of the turbine exhaust to heat pressurization gases (cold gas system).

All of the above have some degree of merit. A preliminary selection would be to exhaust via a separate nozzle; however, final selection will require more detailed analysis.

Preliminary Oxidizer Feed System Selection

For planning purposes and to accomplish a program cost analysis, a preliminary oxidizer feed system was selected; it is presented in Figure A-17. The pressurization system consists of stored Tridyne. This gas mixture is contained in the pressure bottle by two isolation valves. Each leg is capable of handling full gas flow just in case one isolation valve fails to open. A pressure transducer is provided so that pressure in the tank is known at all times.

The gas flow then goes to a normally open isolation valve through a gas regulator and to a catalytic gas generator where the oxygen and hydrogen react to heat up the helium. The products entering the LOX tank are heated helium and steam. A second regulator is provided in parallel with the first and is

Table A-23. Airesearch Turbopump Performance Characteristics

PUMP	NOMINAL OPERATION	PUMP OUT CONDITION
Type	Single Stage Mixed Flow	
LOX Flow Rate, KG/sec	786	1048
Power, HP	9500 (11,750*)	14,000 (17,300*)
Efficiency	0.84	0.76
Mean Tip Diameter, CM	22.9 (23.6)	22.9 (23.6)
TURBINE		
Type	One Stage Impulse	
Turbine Inlet Pressure, MPa	5.2 (1)	7.5
Turbine Flow, KG/sec	7.3 (9.0*)	10.4 (12.8*)
Efficiency	0.42	0.48
Speed, rpm	16,000	17,000
Tip Diameter, CM	48.3	48.3

* Regen cooling condition
(1) Throttled from main GG pressure of 1085 psia

Table A-24. LOX Turbopump Transient Performance

Time on Duty Cycle (Seconds)	10	60	80	110
Flow Rate, KG/sec	3144	2177	2359	1905
Flow Rate per Pump, KG/sec	786	544	590	476
Gas Generator Chamber Pressure, MPa	7.5	5.2	5.6	4.4
Pump Outlet Pressure[(2)], MPa	9.5	6.5	7.1	5.6
Pump Efficiency	0.84	0.75	0.76	0.74
Speed, rpm	16,000	12,640	13,280	11,520
Required Power, HP	9500	4825	5628	3657
Turbine Inlet Pressure[(3)], MPa	4.3	2.8	3.1	2.3
Turbine Efficiency	0.43	0.37	0.39	0.35
Turbine Flow[(4)], KG/sec	7.3	4.8	5.3	3.9

(1) Using solid propellant gas generator fluid to drive turbine. Ablative cooling version.
(2) Assuming 26.7% higher than chamber pressure.
(3) Throttled down from chamber pressure.
(4) Total turbine flow for whole duty cycle is estimated to be 617 KG.

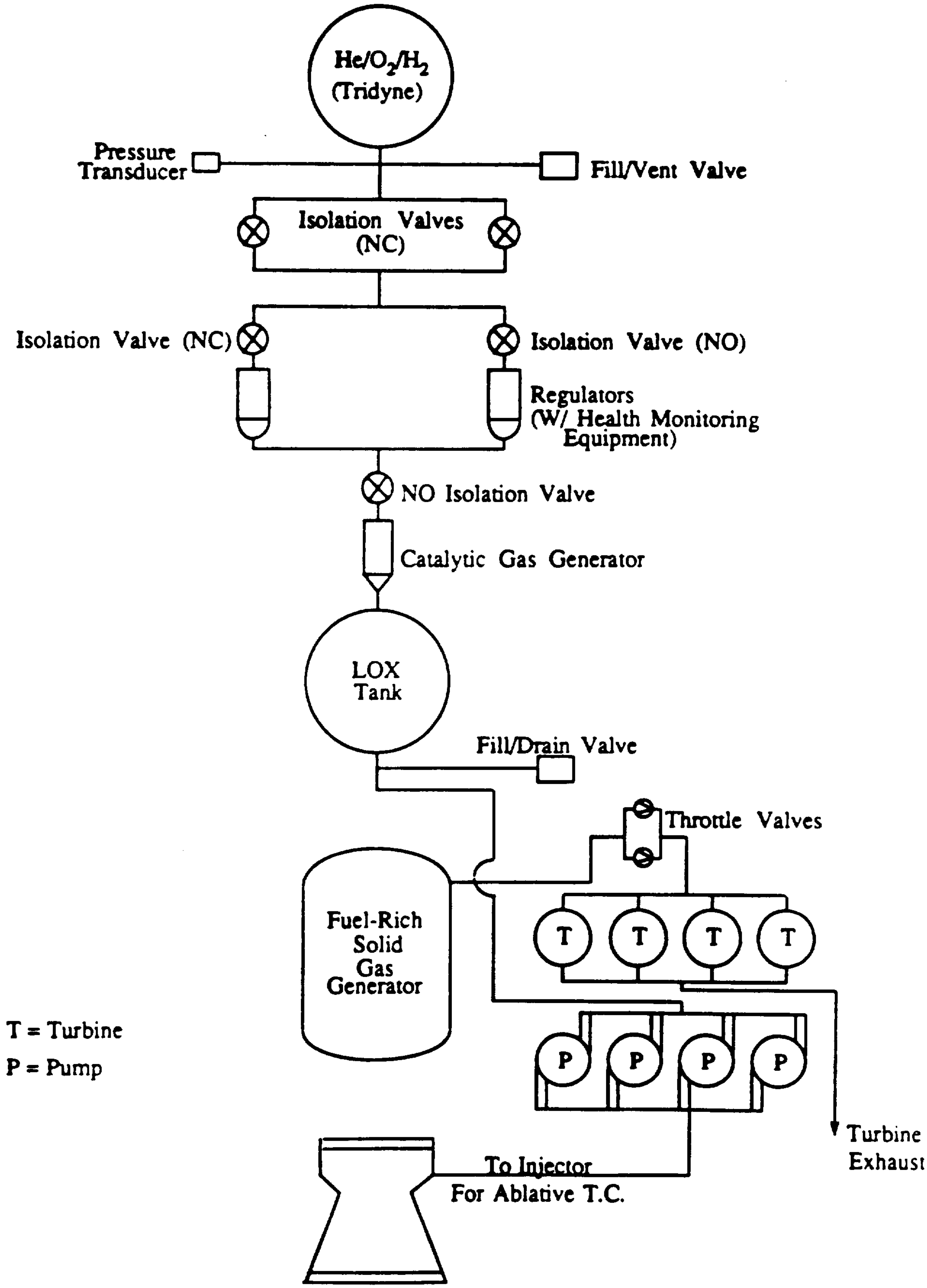

Figure A-17. Preliminary Oxidizer Pump Fed System Schematic.

connected to a normally closed isolation valve. In case the first regulator malfunctions, this isolation valve can be opened. The isolation valve with the malfunctioning regulator can be closed, and the system will continue to operate. Regulators with built-in health monitoring systems should be used in this application. In fact, this switchover should occur automatically with no outside signals required.

Since the fluid to be pressurized is cryogenic, the steam generated will liquify and eventually freeze. However, this should not cause any problem until the last bit of oxidizer is being forced out of the tank. This can be prevented by adding slightly more oxidizer than is required for the mission.

Fuel-rich gases from the solid gas generator are tapped off and sent through parallel throttle valves to power parallel turbines. The turbine exhaust will be passed through a nozzle and expanded to ambient pressure conditions. These throttle valves can be closed in the event an emergency shutdown is required. The single normally open isolation valve just upstream of the catalytic gas generator is also used for an emergency shutdown.

Oxidizer from the tank is sent directly to the pump inlet. The oxidizer pressure is greatly increased, and sent to the thrust chamber injector when an ablative thrust chamber is used or to the inlet cooling jacket when a regenerative cooled thrust chamber is used.

A system pressure schedule is shown in Table A-25. This schedule covers both the ablative and regeneratively cooled thrust chamber cases.

Table A-25. System Pressure Schedule

Tridyne Storage Pressure at 289 (MPa)	68.9
Regulator Outlet Pressure (MPa)	1.8
Catalytic Gas Generator Pressure (MPa)	1.8
Tank Pressure* (MPa)	0.8 (min)
	0.9 (max)
Inlet Pressure to Pump (MPa)	0.4
Pump Outlet Pressure (Ablative) (MPa)	9.5
Pump Outlet Pressure (Regen) (MPa)	11.5

*Includes static head. Minimum tank pressure is 65 psia.

APPENDIX B

HYBRID LIFE CYCLE COST

Hybrid LCC Mode 1 Costing Methodology

The hybrid LCC model was developed using a wide array of cost experience from launch vehicle programs, spacecraft/probes, upper stages, tactical/strategic missiles, and commercial aircraft. As in most parametric cost models, weight is the primary input into the costing algorithms. The ability of the lower level relationships (i.e., the "pieces") to predict cost is less accurate than the model in total. Typically, the more detail design data that is available, as in a later phase of the hybrid booster program, the more accurate the component costs. However, the total costs produced in preliminary studies such as this one are usually very representative and comparable.

The cost algorithms for the hybrid booster are comprised of several elements. Cost estimating relationships (CERS) are used to predict the hardware engineering design costs and the hardware manufacturing costs. Other costs, called support costs, account for items not directly attributable to the hardware itself. Finally, cost figures related to facilities, operations, and support equipment are calculated.

The component hardware design engineering include the tasks of basic component and subsystem design, drafting, developmental shop, testing, finance support, and supervision and clerical. The component hardware manufacturing CERS include the tasks of basic factory labor, quality control, and subcontract and material costs. The support set of cost relationships are systems engineering, software engineering, system test, tooling, and everything else called other. Some of the other costs include logistics, engineering liaison, facilities engineering, and data.

Cost Element Definitions

Design Engineering: The function concerned with applying understanding and knowledge of materials, natural phenomena, and the industrial arts to configure and design systems of hardware and software which satisfies known or anticipated needs of customers. It includes the effort to prepare hardware/systems drawings, data, specifications, and required design reviews, and design confirmation by utilizing mockups, breadboards, prototypes, etc.

Developmental Shop Labor: The shop support to engineering during the design, development, test and production activities. It includes the planning, building, and maintenance of models, breadboards, mockups, test articles, tools, assistance to engineers in the conduct of laboratory and development tests, and inplant liaison to remote activities.

Subsystem Integration & Test: Includes the effort to integrate components into subsystems. Specifically, it includes the effort to test and verify electrical and structural interfaces and specification compliance.

The following manufacturing CERS include the following task direct functions:

Manufacturing Engineering: It includes the activities of tool and production planning, special charges, manufacturing development, and shipping. Some of the tasks include converting engineering designs into manufacturing plans, identifying factory equipment and tools required for the manufacture of the hardware, reviewing supplier manufacturing capabilities, providing numerical control plans and programs, the charge of items damaged in transit, refining and reporting on the manufacturing process, fabricating shipping containers, and packaging and crating parts for in-house and customer delivery.

Quality Assurance: The effort required to perform non-destructive tests on hardware to see if it meets engineering requirements, specifications, T.O. requirements, and ensure that vendor products and procedures meet quality requirements.

Subsystem Assembly: The effort of joining components into a sub-assembly. Included would be any subsystem testing.

Basic Factor Labor (BFL): The shop activity required to fabricate, assemble, and functional test an end item of hardware to include fabrication, minor assembly, and major assembly.

Final Assembly and Checkout: The effort of joining subassemblies into a final assembly. Included would be the final functional test of the end item.

The following definitions relate to the support costs categories of the hybrid booster cost model.

Systems Engineering: All activities directed at assuring a totally integrated engineering effort. It includes the effort to establish system, subsystem, GSE, and test requirements and criteria; to define and integrate technical interfaces to optimize total system definition and design; to allocate performance parameters to the subsystem level; to identify, define, and control interface requirements between system elements, to monitor design and equipment to determine CEI compliance; to provide and maintain inertial properties analyses, support and documentation; to develop and maintain system specification to provide parts, standards and materials and processes surveillance and to integrate product assurance activities. Fundamental to this element is the documentation of system level design requirements and derived from customer established requirements and guidelines and through functional analysis. System engineering effort includes, for example, system definition, overall system design, design integrity analysis, system optimization, cost effectiveness analysis, weight and balance analysis and intrasystem and intersystem compatibility analysis. It also includes reliability, maintainability, safety, and survivability program requirements, human engineering and manpower factors, program preparation of equipment and component performance specifications, security requirements, logistics support integration, and design of test and demonstration plans.

Software Engineering: All effort to design, develop, test, deliver, and maintain (for the program phase being estimated) computer software; with software including all associated programs, data, procedures, rules and documentation required for system operation. Software may be subdivided into the three categories of test, ground operational, and flight operational.

System Test: All manpower required to plan for and test prototype equipment as a system in order to acquire engineering data, confirm engineering hypotheses and qualify the system design in total. This element is limited to environmental, space chamber (space programs), wind tunnel, ground based tests, and includes static, dynamic, fatigue, subsystem performance, qualification, and reliability tests.

Tooling & Special Test Equipment: Tooling includes all effort to plan, design, fabricate, assemble, inspect, install, test, modify, maintain, and rework jigs, dies, fixtures, molds, patterns, and other manufacturing aids that are of a special nature necessary for the manufacture of mission hardware. Special test equipment includes all effort to design and/or manufacture that unique equipment which is used for testing during the development or production of mission hardware.

Other: The other category is comprised of liaison engineering, logistics, data, and other miscellaneous effort such as facilities engineering, safety, training, etc.

The hardware design and manufacturing CERS are defined to the level of thermal protection, tanks, control box, actuation system, valves, etc. Many of the same CERS will be used in a variety of subsystems. For example, the control box CER can be used in the cold gas pressurization subsystem, the liquid tank subsystem, and the nozzle subsystem.

The process of using the cost model begins with a careful accounting of all components. The weights routine must supply weight (in pounds) for each line item.

Design engineering costs follow the form:

Engineering Dollars = A(wt)**B

The answer is subsequently modified by linear multipliers of this equation that account for hardware complexity, technological maturity, and the degree of "off-the-shelf" hardware designs.

The off-the-shelf (OTS) factor is a correction factor that accounts for previous design efforts that could be applied to a new component, thus reducing the cost of engineering design. At the lowest and finest level of component definitions (nuts, bolts, chips, etc) virtually everything would be off-the-shelf. The other extreme, the macroscopic end item level, virtually nothing is off-the-shelf. To determine where the OTS factor would fall in this spectrum, we try to estimate what percentage of the total engineering drawings/specifications are available for a given component. Figure B-1 simply converts this percentage to an OTS factor. By way of example, suppose a valve required 15 engineering drawings, and 3 drawings from a similar valve were applicable and valid. The percentage of available drawings, 3/15, or 20%, corresponds to an OTS factor (from Figure B-1) of 0.8.

Similarly, the curve for design complexity factor, Figure B-2, relates an experienced judgement of component complexity to an appropriate multiplier for the design cost equation.

The third design cost multiplier reflects the impact of the level of maturity for the selected technology. A judgement is made concerning the status of the hardware's technology development. Figure B-3 provides a maturity design factor to use as a multiplier to the design cost equation.

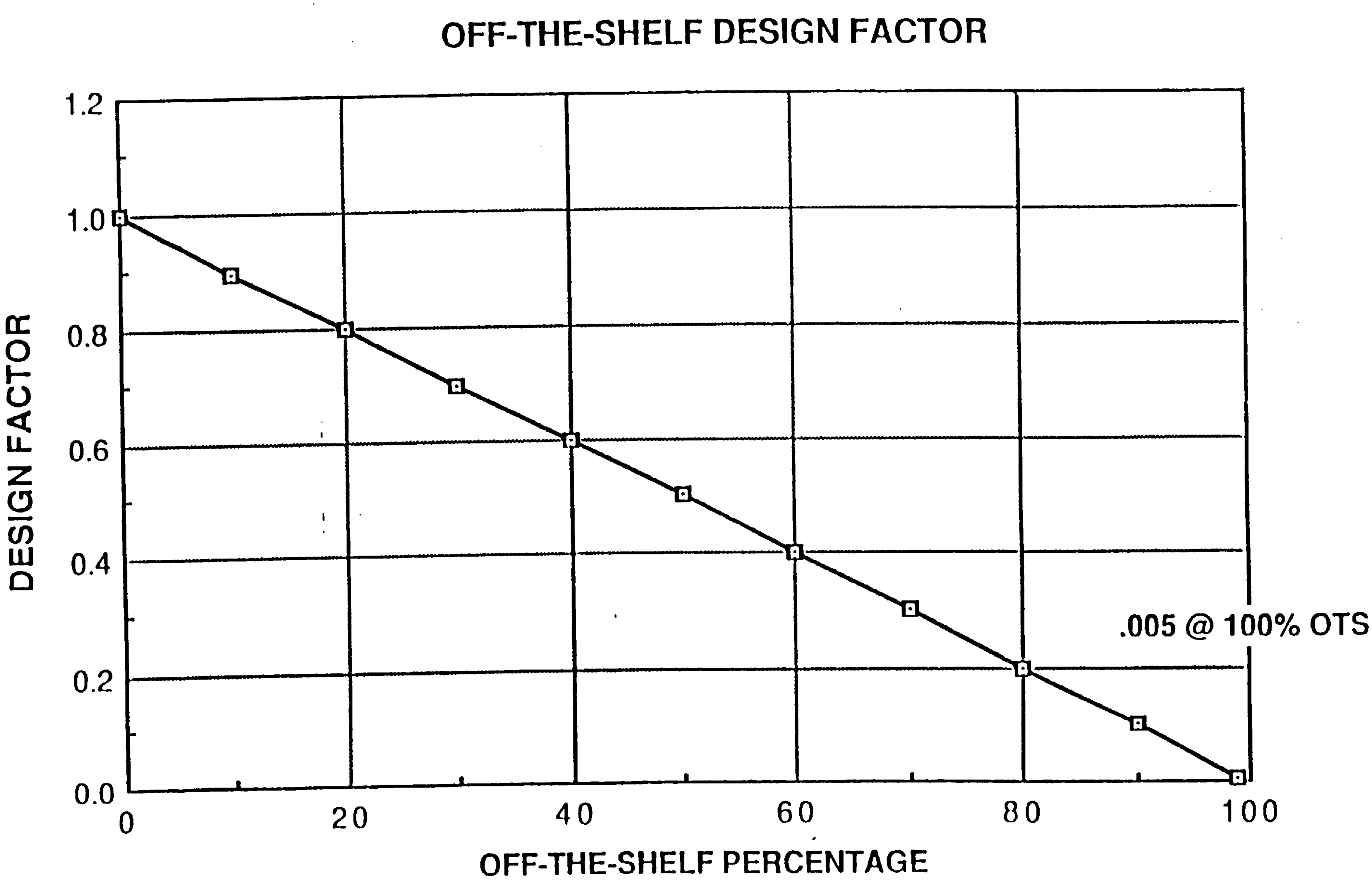

Figure B-1. Off-the-Shelf Design Factor.

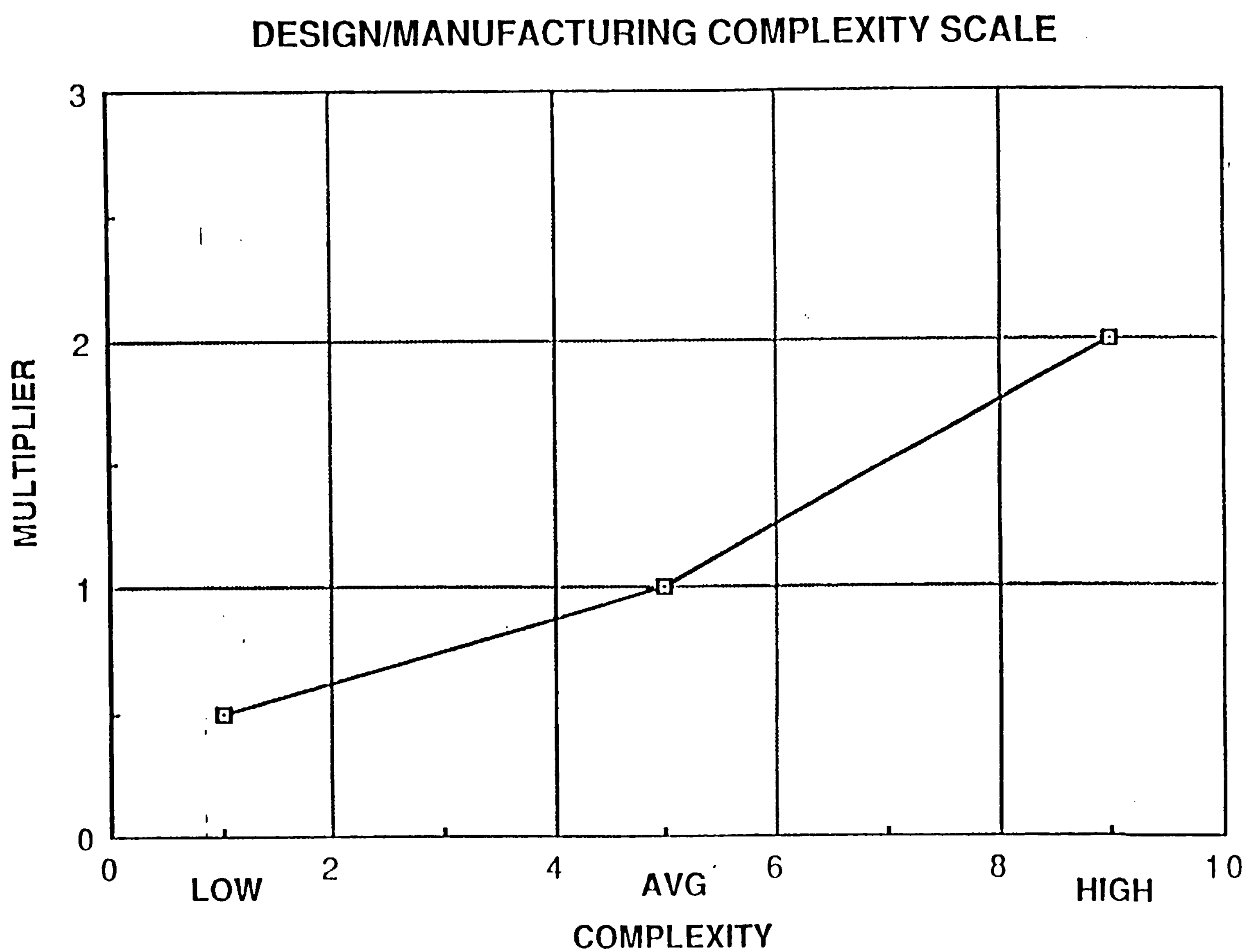

Figure B-2. Design/Manufacturing Complexity Factor.

Engineering Technology Maturity Level

Level	Description	Design factor
1*	Qualified off-the-shelf hardware design	.30
2	Engineering model tested in actual mission environment	.45
3	Prototype engineering model tested in relevant environment	.65
4	Preprototype, engineering model tested	.75
5	Component brassboard tested	.80
6	Critical function—characteristic demonstrated	.85
7	Conceptual design tested analytically or experimentally	.90
8	Concept design formulated	1.00
9	Basic principles observed and reported	NA
10	Basic principles not identified	NA

*At this maturity level, an appropriate "% of components available OTS" should be used on the hardware specifics input sheet.

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Figure B-3. Maturity Design Factor.

The following CERS are for the engineering design of hybrid booster components. Note that all the equations have the form:
(A*(wt)**B)*(complexity factor)*(OTS factor)*(Maturity factor)

Nose Cone:

Structural shell - includes all structure and fasteners for nose cone section and attachment provisions to oxidizer tank.

A = 8609.5
B = 0.7647

Thermal - includes thermal protection and insulation and attachments to structure of nose.

A = 5,470.6
B = 0.6200

Cold gas pressurization system:

Tank - includes all structure, liners, insulation, and attachment fittings for high pressure tanks for pressurant storage.

Small tank:	A = 57,700
(less than 200 lb)	B = 0.7643
Large tank:	A = 158,059
(more than 200 lb)	B = 0.479

Valves - includes isolation valves, service relief valves, and and pressure regulators.

A = 72,220	(less than 35 lb)
B = 0.7034	
A = 87,420	(more than 35 lb)
B = 0.5163	

Control box - includes structure, electronics, wiring, and attachment of controller for pressurization system.

A = 173,300
B = 0.7031

Oxidizer tank:

Structural shell - includes all structure, stringers, attachments, and interface flanges to nose cone and interstage.

A = 9188.4
B = 0.7638

Valves - includes isolation valves, pyro valves, and service valves associated with oxidizer tank.

A =72,220	(less than 35 lb)
B = 0.7034	
A = 87,420	(more than 35 lb)
B = 0.5163	

Control box - includes structure, electronics, wiring, and attachment of controller for oxidizer system.

A = 173,300
B = 0.7031

Thermal - includes external thermal protection, insulation, liners, and attachments to tank structure.

A = 5,470.6
B = 0.6200

<u>Nozzle:</u>

Structure - includes nozzle structure and attachment provisions to combustion chamber and gimbal activation.

Based on total solid motor design (Note: gimballed vs. fixed nozzle difference is 1.28 times higher)

Actuation assembly - includes actuators, sensors, hydraulic control, accumulators, and attachments for gimballed TVC nozzle concepts.

A = 68,740
B = 0.8764

Fluid injection system - includes all plumbing, sensors, and injectors associated with a fluid injection TVC concept.

A = 130,000
B = 0.4100

Valves - includes all valves associated with fluid injection TVC concepts.

A = 72,220 (less than 35 lb)
B = 0.7034
A = 87,420 (more than 35 lb)
B = 0.5163

<u>Thrust Control valve:</u>

Valves - includes variable (throttleable) valves associated with thrust control.

A = 72,220 (less than 35 lb)
B = 0.7034
A = 87,420 (more than 35 lb)
B = 0.5163

Control box - includes structure, electronics, wiring, and attachment of controller for thrust control valve.

A = 173,300
B = 0.7031

Lines

Lines - includes oxidizer lines, bypass lines, pressurant system lines, and turbopump fuel feed lines.

A = 17,640
B = 0.4951

Other structures:

Aft skirt - includes all structure and fasteners, interfaces and attachments with nozzle, actuators and gas generator case, and load paths/hold downs for interfacing with a launch pad.

A = 218,000
B = 0.3305

Interstage - includes all structure and interface flanges and attachments to the oxidizer tank and gas generator case.

A = 75,125
B = 0.4569

Attach struts - includes all fore and aft attachment struts and fittings required to handle loads between the hybrid booster and parallel core vehicle.

A = 795,000
B = 0.273

Separation motors:

Rocket (cluster) motors - includes all rocket motors, ignitors, attachments, safe and arm, and sequencers for separation system.

A = 1,610,784
B = 0.553

Electrical Systems:

Electronics and Instrumentation - includes all electronics hardware and software, software development, monitoring instrumentation, sequencing, range safety, and control algorithms.

A = 221,800
B = 0.5276

Electrical power supply - includes all power storage, conditioning, and distribution hardware for electrical power to electronics, valves, and any electrical actuators for the period of time from ground umbilical disconnect to vehicle recovery.

A = 242,500
B = 0.7009

Cabling - includes all wires and interface connectors associated with electrical power and signal distribution.

A = 87,389
B = 0.693

Turbopumps:

Oxidizer (and hydrocarbon) turbopumps - includes turbopump assembly, exhaust system, and mounting provisions.

A = 35,000
B = 1.000

Gas Generator:

Solid motor - includes all structure, insulation, propellant, ignitor, safe and arm, and injector hardware.

A = 261,000 (General equation for total solid
B = 0.4100 rocket motor)

Injector - includes all structure and interfaces.

A = 279,796
B = 0.4900

Catalyst Bed:

Catalyst Bed - includes case, catalyst, interfaces, and mounting provisions.

A = 195,857
B = 0.490

After calculating the engineering costs, a 20% addition is made to account for the subsystem integration effort.

The manufacturing dollars are calculated using the same general form of the engineering dollars equation. The equation is then modified by a series of linear multipliers that account for the hardware manufacturing complexity, a material factor, and the learning curve cum factor.

For the manufacturing CERS, the first of these linear multipliers, the complexity factor, uses the same curve as for engineering design. Refer to Figure B-2 to select the appropriate factor for a selected complexity level.

The manufacturing costs are also modified by a material factor which accounts for the relative cost of manufacturing and raw materials for typical booster hardware. The material factors used are as follows:

Aluminum	= 1.0
Aluminum Lithium	= 2.64
Titanium	= 1.45
Stainless Steel	= 2.0
Carbon Composite	= 1.14
Steel	= 1.0

The third multiplier accounts for the learning curve effect. The learning curve (LC) cum factor includes both the "slope" of the learning curve, as well as the quantity of units produced.

The value of the Nth unit, call it Y, can be expressed as

$$Y = AN^{\frac{\log_{10}(\text{slope}) - 2}{\log_{10}(2)}}$$

Where: A = theoretical first unit (TFU) value

Slope = learning curve slope values

The cumulative curve, which results in a LC cum factor, is calculated from:

$$\text{LC cum factor} = \frac{1}{Z+1}\left((N + \tfrac{1}{2})^{Z+1} - (\tfrac{1}{2})^{Z+1}\right)$$

Where: $Z = \frac{Y}{AN}$

By way of example, building 300 valves using a 92% curve results in a LC cum factor value of approximately 171.3.

The following CERS are for the manufacturing of hybrid booster components. Note that all the equations have the form:

(C* (wt) ** D) * (complexity factor) * (material factor) * (LC cum factor).

The component descriptions, as far as what each item entails, is the same as the descriptions previously given for the design CERS.

<u>Nose Cone:</u>

Structural shell -	C = 12,140 D = 0.6727	
Thermal -	C = 2,156 D = 0.7505	

<u>Cold gas pressurization system:</u>

Tank -		
Small tank - (less than 200 lb)	C = 22,390 D = 0.5713	
Large tank - (more than 200 lb)	C = 14,863 D = 0.654	
Valves -	C = 4,254.7 D = 0.8617	(less than 35 lb)
	C = 3,520.9 D = 0.5228	(more than 35 lb)
Control Box -	C = 52,540 D = 0.5669	

<u>Oxidizer tank:</u>

Structural shell -	C = 3183.84 D = 0.8076	
Valves -	C = 4,254.7 D = 0.8617	(less than 35 lb)
	C = 3,520.9 D = 0.5228	(more than 35 lb)
Control box -	C = 52,540 D = 0.5669	
Thermal -	C = 2,156 D = 0.7505	

Nozzle:

Moveable:	0.5 x (325250 + 108 x nozzle wt.) + 0.5 x (273250 + 0.97 x avg. thrust)
Fixed:	0.5 x (85005 + 131 x nozzle wt.) + 0.5 (57701 + 1.03 x avg. thrust)

Actuation assembly -	C = 10,821.8 D = 0.5454	
Control box -	C = 52,540 D = 0.5669	
Fluid injection system -	C = 5,400 D = 0.5454	
Valves -	C = 4,254.7 D = 0.8617	(less than 35 lb)
	C = 3,520.9 D = 0.5228	(more than 35 lb)

Thrust Control Valve:

Valves -	C = 4,254.7 D = 0.8617	(less than 35 lb)
	C = 3,520.9 D = 0.5228	(more than 35 lb)
Control box -	C = 52,540 D = 0.5669	

Lines:

Lines -	C = 11,550 D = 0.3143

Other Structures:

Aft skirt -	C = 25,360 D = 0.4961
Interstage -	C = 11,905 D = 0.571
Attach struts -	C = 4120.3 D = 0.6593

Separation Motors:

Rocket (cluster) motors - C = 17,894
D = 0.544

Electrical Systems:

Electronics and instrumentation - C = 34,130
D = 0.7524

Electrical power supply - C = 22,720
D = 0.4477

Cabling - C = 3445.2
D = 0.927

Turbopumps:

Oxidizer (and hydrocarbon) turbopumps C = 1,000
D = 0.800

Gas Generator:

Solid motor case

((-291,291 + 330.86* volume
+ 382,584* Reuse)*
(-539,179 + 50.57* weight
+ 737,152* Reuse))^.5
Reuse = 1 Expendable
Reuse = 2 Reusable

Injector - C = 33,932
D = 0.613

Catalyst Bed:

Catalyst Bed - C = 16,966
D = 0.613

After calculating the manufacturing dollars, a 5% addition is made to account for the subsystem assembly effort.

To account for final assembly and checkout to arrive at a complete system, the manufacturing dollars are added to the 5% subsystem assembly factor and the sum is multiplied by 15%.

The support function costs are calculated based on the resultant design and manufacturing costs. Refer to the previous definitions of what activities are included in each support area.

Systems engineering dollars are computed as:

0.323 * (Design $) ** 0.9802

Software engineering dollars are computed as:

1.370 * (Design $) ** 0.8944

System test dollars are computed as:

0.0006 * (Design $) ** 1.3226

Tooling costs are manufacturing dependant:

0.0045 * (Manufacturing$) ** 1.1526

Miscellaneous costs are computed as:

(0.1138 * (Design $) ** 1.0185) ÷ (0.03 * (Manufacturing $))

The remaining costs that need to be accounted for are for the ground complex and launch operations (GCLO). The basis for the algorithms and estimating relationships is a collection of historical booster system data. This data is related to nonrecurring investment and recurring costs for launch facilities, ground support equipment (GSE), booster launch operations, and recovery/refurbishment operations. Figure B-4 describes the GCLO cost breakdown structure.

Cost data were escalated to Fiscal Year 1988 levels using NASA JSC escalation tables. Costs in millions of FY 88 $ were tabularized and regressed against significant launch system technical or programmatic characteristics. All algorithms included herein provide solutions in FY88 millions of dollars. The algorithms are loosely structured into a preliminary cost model architecture which defines nonrecurring investment as the sum of launch facilities costs and ground support equipment costs. recurring costs are defined as the sum of launch operations costs and refurbishment costs. Complexity factors are available within the detailed algorithms to tailor cost solutions to a particular booster and its launch requirements. Appendix A lists the sources used for GCLO data.

All cost estimates are at price/outlay level in constant FY88 dollars (millions). All facilities algorithms cover construction of new installations. If existing facilities at ETR or PMR are to be modified/converted for advanced launch systems, complexity adjustments reflecting the relative percentage of modification must be applied. Facilities and Ground Support Equipment algorithms related to the pad area represent unit pad expense. Typically, a system launch complex may contain two or more individual pads to support maximum launch rates and provide backup in the event on on-pad explosions and other contingencies.

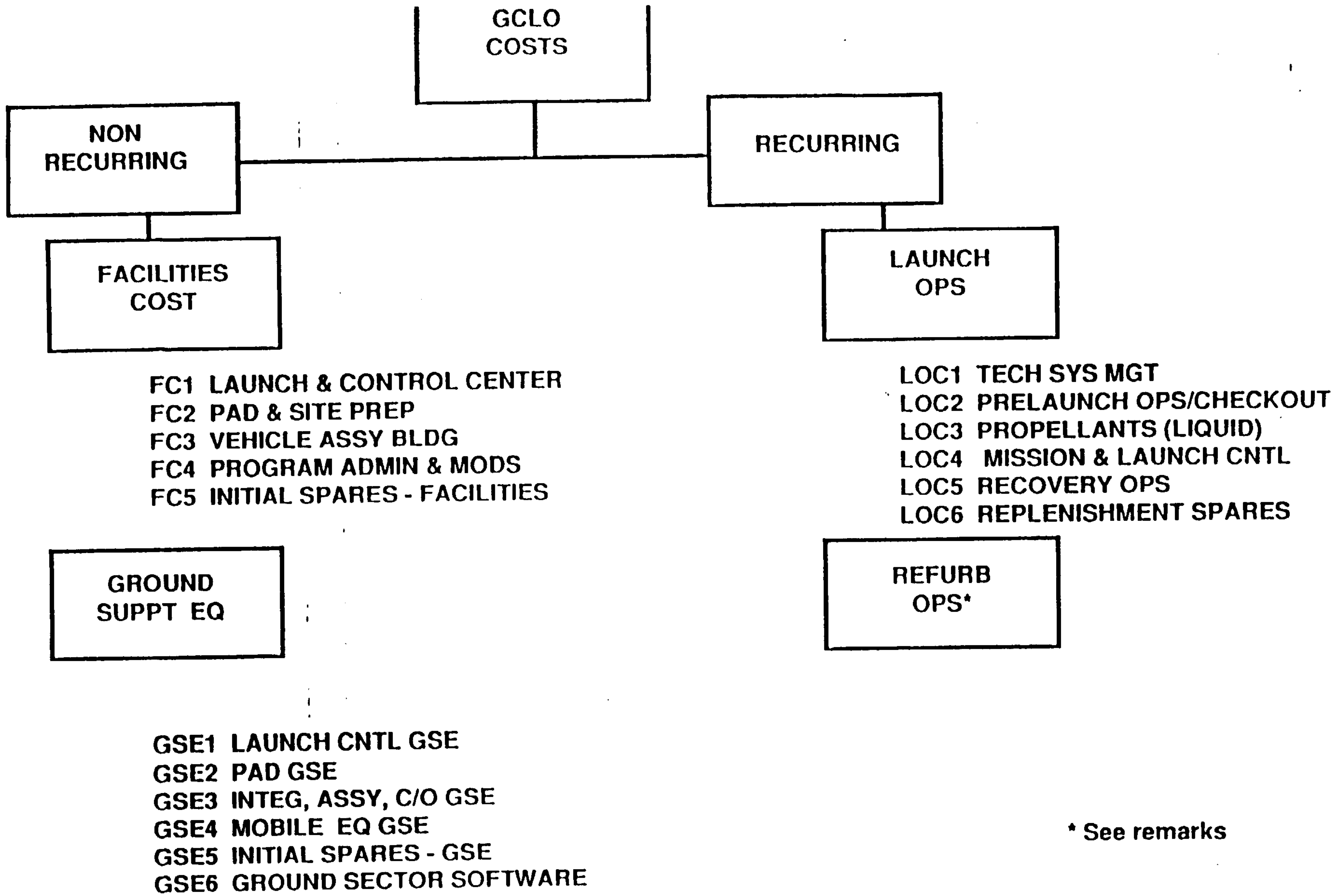

Figure B-4. Ground Complex and Launch Operations (GCLO) Cost Breakdown Structure.

Facilities include brick and mortar and real property installed equipment (RPIE). Any support item which is mobile or transportable is classified herein as Ground Support Equipment (GSE). Real Property Installed Equipment (RPIE) is permanently emplaced during construction of the launch complex.

Ground Support Equipment is that population of support items used to launch, service, checkout, maintain, and provide training which are mobile or transportable.

Launch Operations includes costs of technical system management, prelaunch operations and checkout, propellant charges for liquid fueled systems, mission and launch control operations, recovery operations, and sustaining spares requirements of GSE and Facilities.

The following nonrecurring cost algorithms are for facilities costs (FC).

Launch & Control Center: (See Figure B-5)

$$FC1 = 0.010 * ((TOGW) ** 0.474) * (K1)$$

Where TOGW = Takeoff Gross Weight
K1 = Complexity Factor

Pad & Site Preparation: (See Figure B-6)

$$FC2 = N_p * 0.037 * ((TOGW) ** 0.545) * (K2)$$

Where N_p = Number of pads
TOGW = Takeoff Gross Weight
K_2 = Complexity Factor

Vehicle Assembly Building: (See Figure B-7)

$$FC3 = 0.004 * ((TOGW) ** 0.733) * (K3)$$

Where TOGW = Takeoff Gross Weight
K3 = Complexity Factor

Program Administration & Facility Modifications: (See Figure B-8)

$$FC4 = 0.094 * ((FC1 + FC2 + FC3)) ** 1.224) * (K4)$$

Where FC1 = Launch Control Center Facility Cost
FC2 = Pad and Site Facility Cost
FC3 = Vehicle Assy Bldg Facility Cost
K_4 = Complexity Factor

$$Y(\text{in millions}) = 0.011322 \cdot x^{0.456713}$$

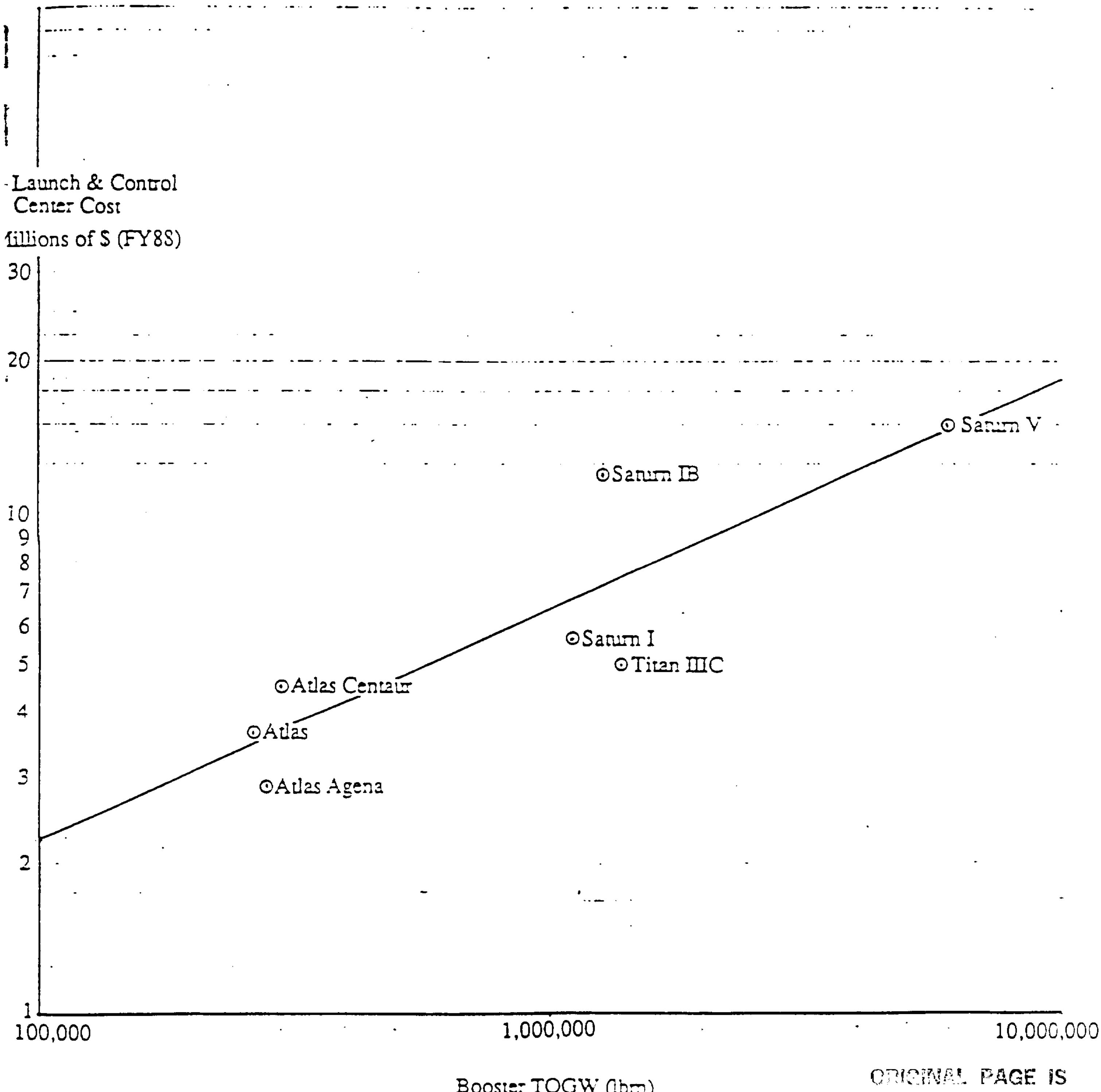

Figure B-5. Launch & Control Center Facilities Cost (FC1) vs. TOGW.

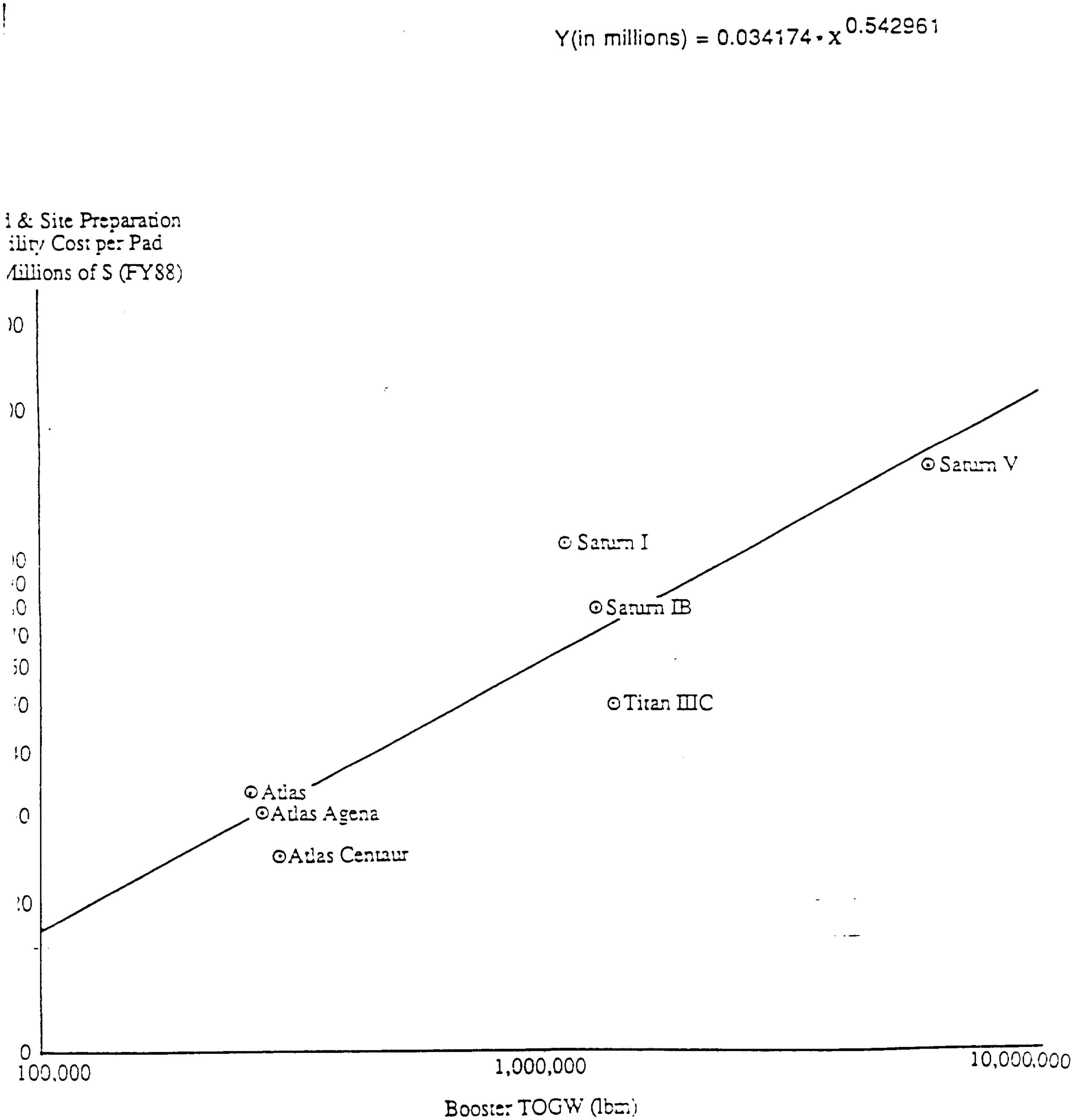

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Figure B-6. Pad & Site Preparation Facility Cost (FC2) vs. TOGW.

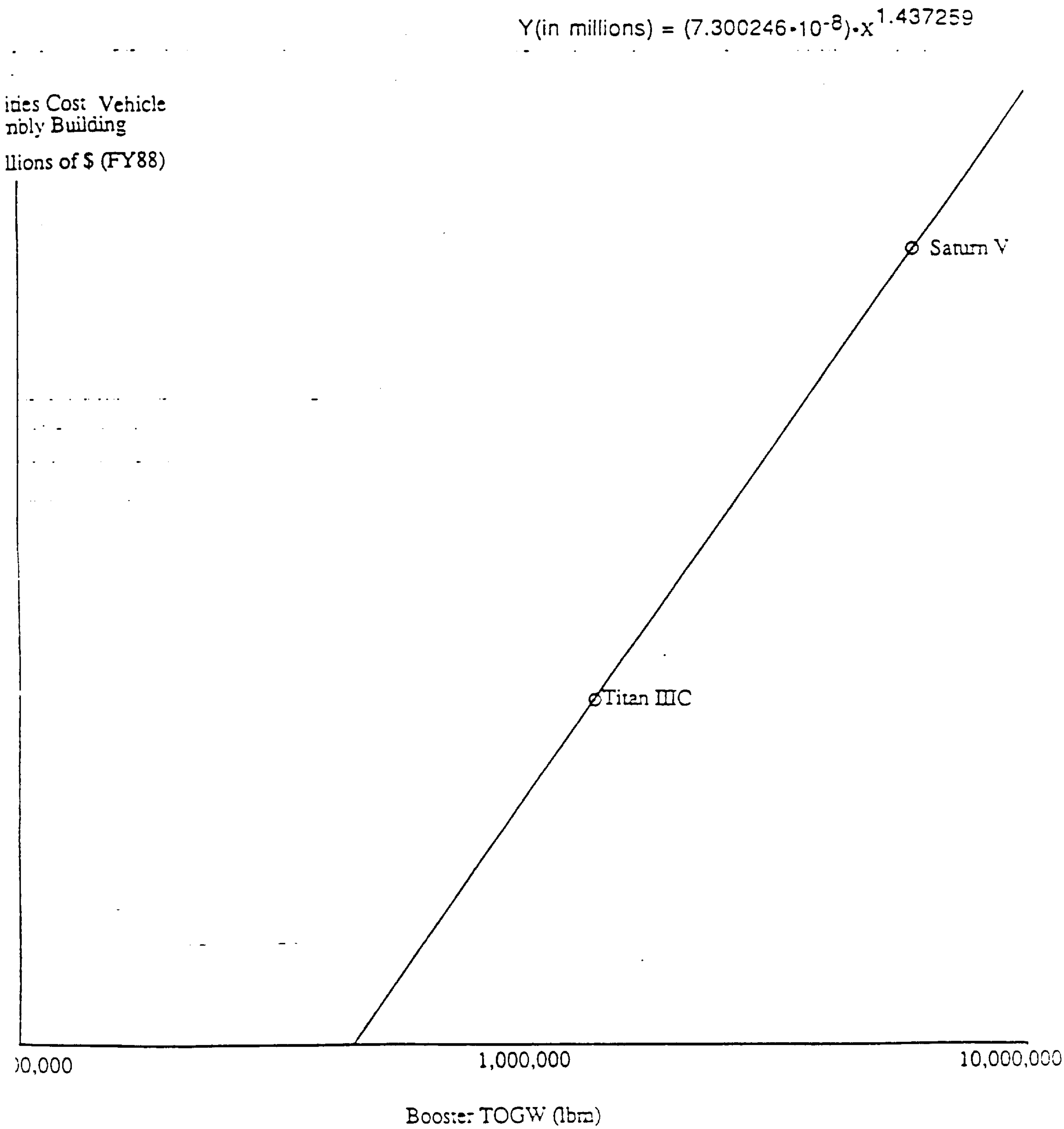

Figure B-7. Vehicle Assembly Building Facilities Cost (FC3) vs. TOGW.

$$Y(\text{in millions}) = (7.300246 \cdot 10^{-8}) \cdot x^{1.437259}$$

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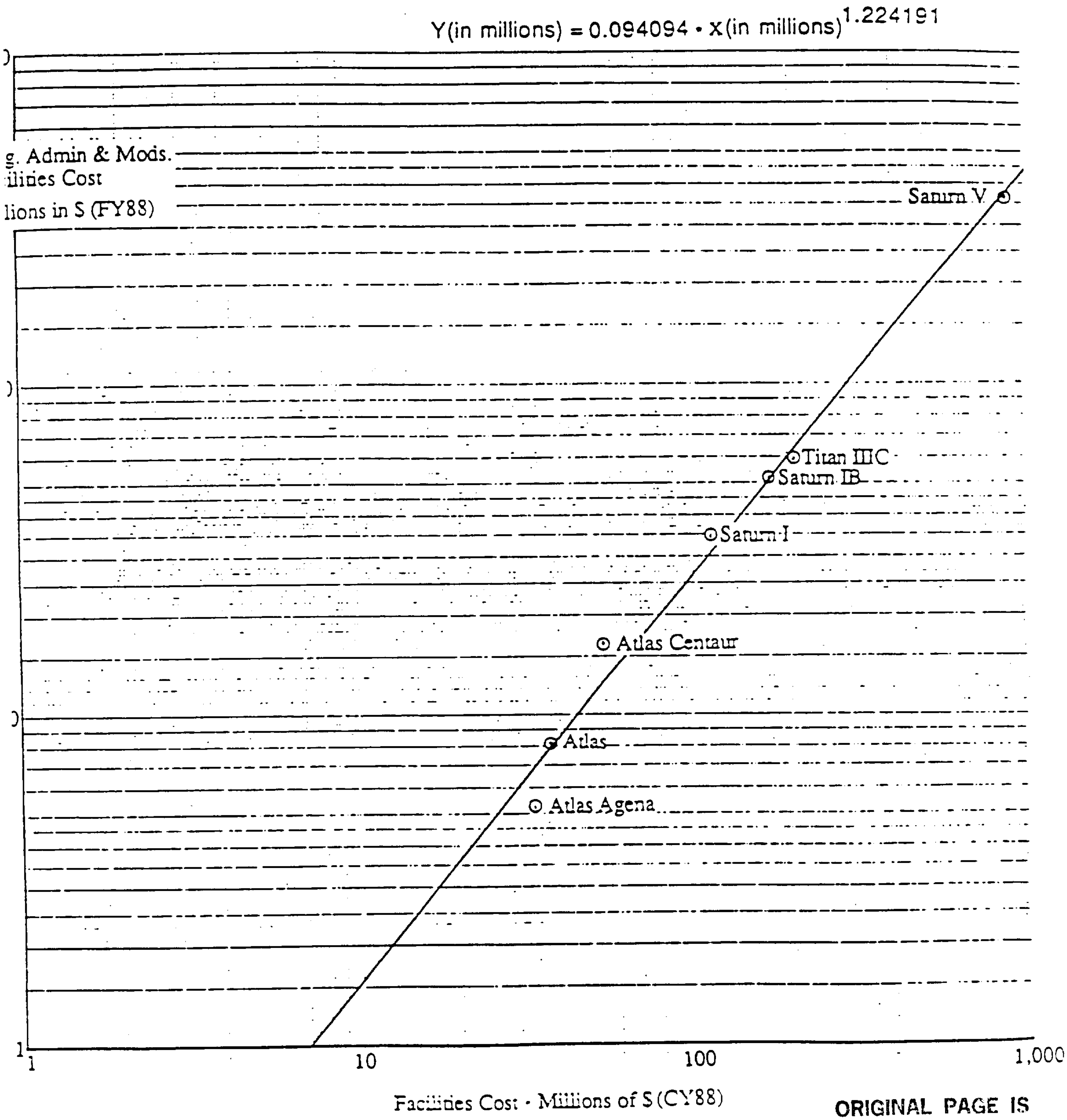

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Figure B-8. Program Administration & Modifications Facilities Cost (FC4) vs. Facilities Costs.

Facilities Initial Spares:

$$FC5 = 0.02 \cdot (FC1) + 0.07 \cdot (FC2) + 0.02 \cdot (FC3)$$

Where FC1 = Launch Control Center Facility Cost
FC2 = Pad and Site Facility Cost
FC3 = Vehicle Assy Bldg Facility Cost

The following nonrecurring cost algorithms are for GSE:

Launch Control GSE: (See Figure B-9)

$$GSE1 = 0.355 \cdot ((ALR)^{1.264}) \cdot (K5)$$

Where ALR = Maximum Annual Launch Rate
K5 = Complexity Adjustment

The following nonrecurring cost algorithms are for GSE:

Launch Control GSE: (See Figure B-9)

$$GSE1 = 0.355 \cdot ((ALR)^{1.264}) \cdot (K5)$$

Where ALR = Maximum Annual Launch Rate
K5 = Complexity Adjustment

Pad GSE: (See Figure B-10)

$$GSE2 = 0.011 \cdot ((TOGW)^{0.612}) \cdot (K6) \cdot (Np)$$

Where TOGW = Takeoff Gross Weight
K6 = Complexity Adjustment
N_p = Number of Pads

IACO GSE: (See Figure B-11)

$$GSE3 = 0.003 \cdot ((TOGW)^{0.743}) \cdot (K7)$$

Where TOGW = Takeoff Gross Weight
K7 = Complexity Adjustment

Mobile Equipment: (See Figure B-12)

$$GSE4 = 16.23 \cdot ((TOGW)^{0.228}) \cdot (K8)$$

Where TOGW = Takeoff Gross Weight
K8 = Complexity Adjustment

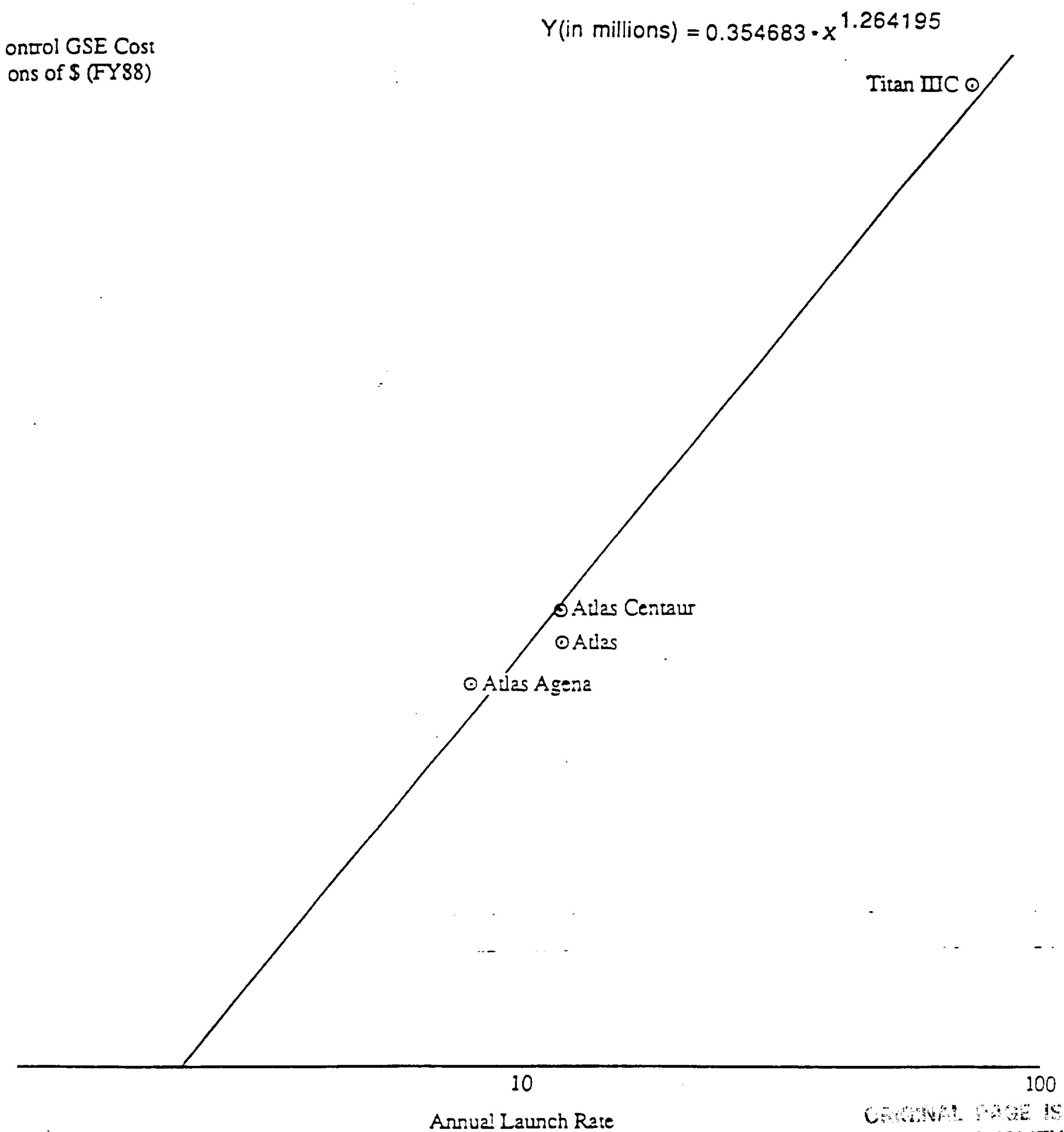

Figure B-9. Launch Control GSE Cost (GSE1) vs. Annual Launch Rate.

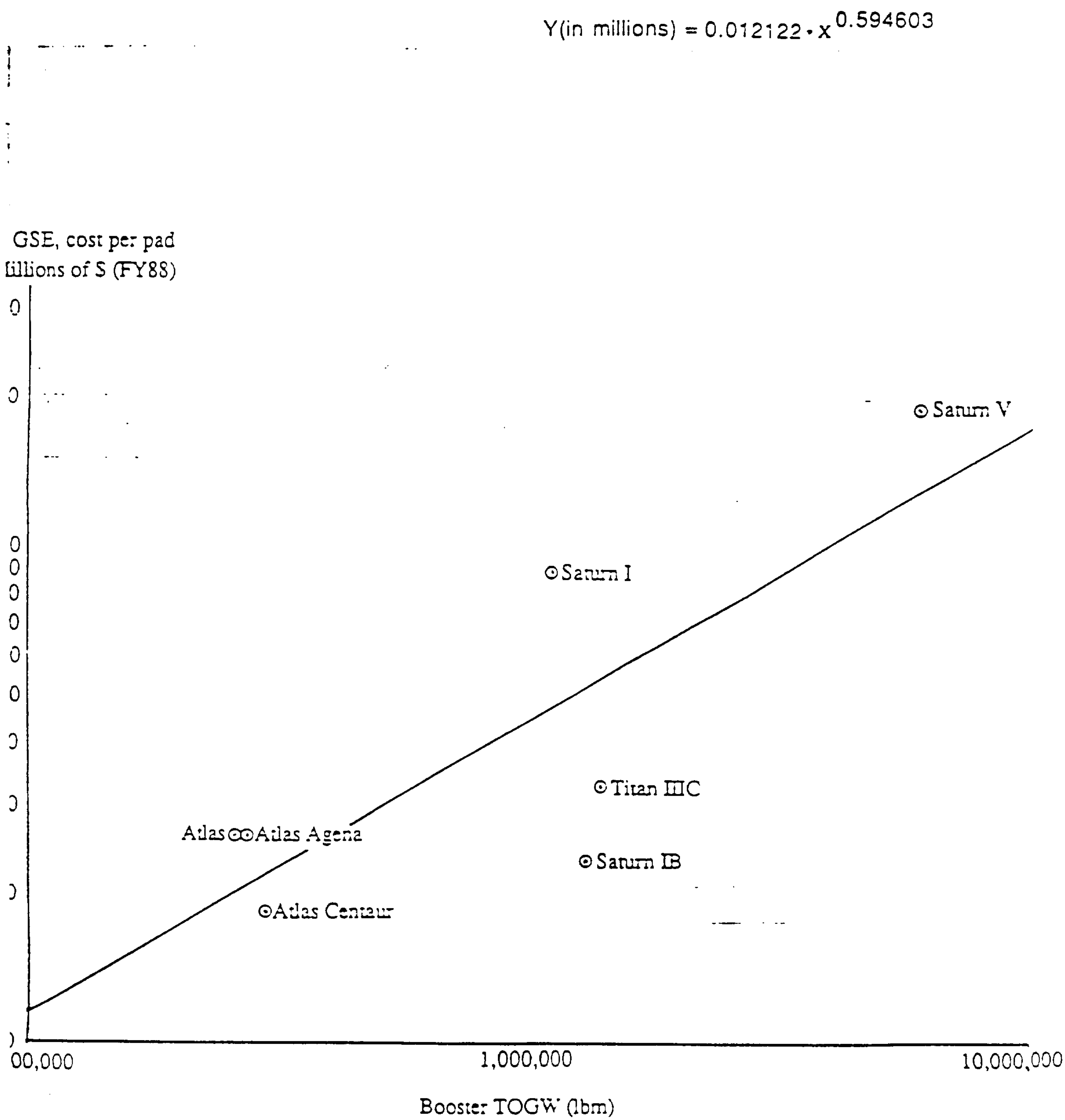

Figure B-10. Pad GSE, cost per pad (GSE2) vs. TOGW.

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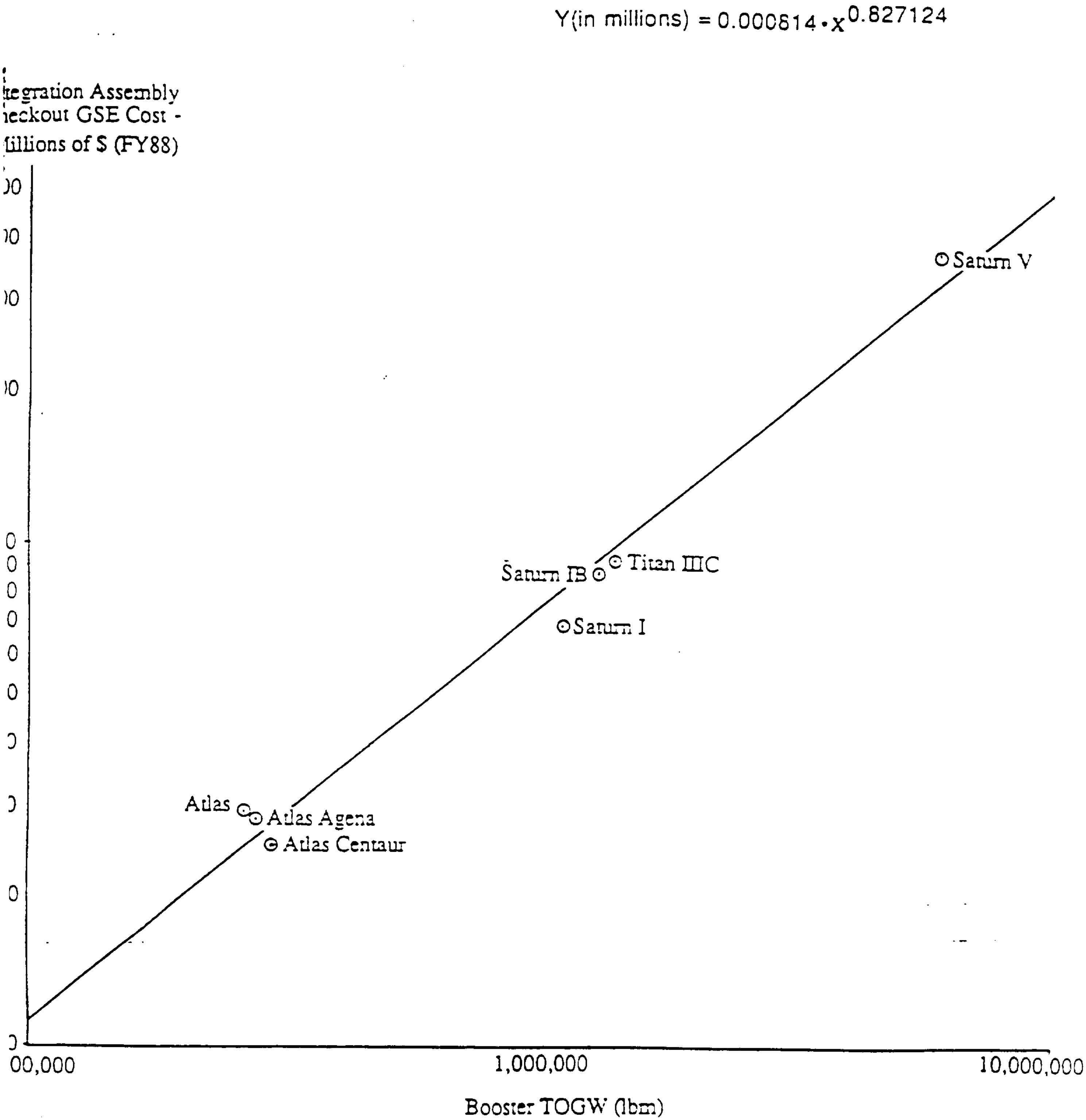

Figure B-11. IACO GSE Cost (GSE3) vs. TOGW.

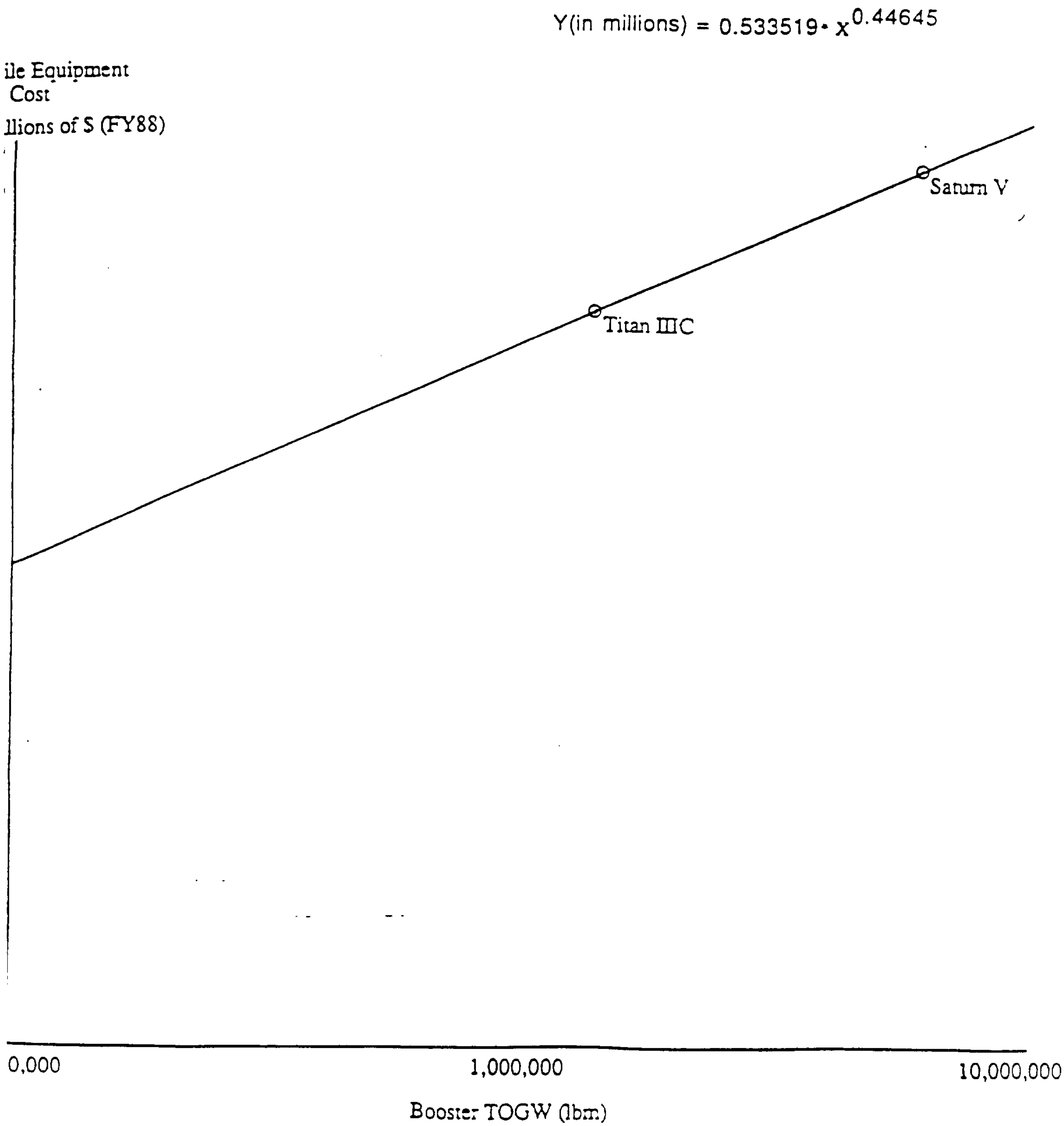

Figure B-12. Mobile Equipment Cost (GSE4) vs. TOGW.

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Initial Spares:

$$GSE5 = 0.05 * (GSE1) + 0.15 * (GSE2) + 0.07 * (GSE3) + 0.5 * (GSE4)$$

Where GSE1 = Launch Control GSE Cost
GSE2 = Pad GSE Cost
GSE3 = IACO GSE Cost
GSE4 = Mobile Equip GSE Cost

Ground Sector Software:

$$GSE6 = 0.036 * ((KSLOC_{TST}) ** 1.12) * (K9) + 0.043 * ((KSLOC_{INST}) ** 1.20)$$

Where K9 = Complexity Adjustment
$KSLOC_{TST}$ =
Thousands of source lines of code, test & checkout

$KSLOC_{INST}$ =
Thousands of source lines of code, real time instrumentation

The following recurring cost algorithms are for annual launch operations costs (LOC):

Tech System Management:

$$LOC1 = 0.009 * ((TOGW) ** 0.516) * ((L) ** 0.360)$$

Where TOGW = Takeoff Gross Weight
L = Annual Launch Rate

Prelaunch Operations Checkout: (see above)

$$LOC2 = 0.025 * ((TOGW) ** 0.516) * ((L) ** 0.360)$$

Propellant Cost:

$$LOC3 = L * (W_F * C_F * B_F) + (W_O * C_O * B_O)) * 10^{-6}$$

Where F = Fuel
O = Oxidizer
W = Propellant weight per flight (lbs)
C = Cost per lb
B = Boiloff factor
L = Annual Launch Rate

Note: Solid propellants are included in assembly costs.

Mission & Launch Control: (see above)

$$LOC4 = 0.010 * ((TOGW) ** 0.516) * ((L) ** 0.360)$$

Recovery Cost:

$$LOC5 = 1.77 * ((L) ** 0.534)$$

Where L = Annual Launch Rate

Note: Sea recovery of 1st stage booster assumed.

Replenishment Spares - FC/GSE:

$$LOC6 = 0.10 * (FC5) + 0.20 * (GSE5) * (L ** 0.05)$$

Where FC5 = Facilities Initial Spares Cost

GSE5 = Ground Support Equipment Initial Spares Cost

L = Annual Launch Rate

GLCO Data Sources

- SP-224 - Launch Complexes for Space Missions: Economic and Operational Considerations. Frederic and Yates, General Electric, Santa Barbara, California, 1963
- ELV database - Space Cost Advisory Group (SCAG). NASA, JSC, 1986
- Cost Model for Space Transportation Systems Development, Fabrication, and Operations, (TRANSCOST), D. E. Koelle, MBB, 1980
- Facilities - Program Population. Atlas Agena, Atlas Centaur, Titan IIIC, Saturn 1, Saturn 1B, Saturn V
- Ground Support Equipment - Program Population. Atlas, Atlas Agena, Atlas Centaur, Titan IIIC, Saturn 1, Saturn 1B, Saturn V
- Launch Operations. Scout, Atlas, Atlas Centaur, Delta, Titan 34D, Ariane

LCC Computer Model

The Boeing Hypervelocity Aerospace Vehicle Conceptual Design (HAVCD) computer program was utilized to assess the impacts of hybrid components and design considerations on hybrid booster cost, reliability, and performance.

Boeing, under independent IR&D, developed this specialized analysis program in 1986 and 1987. HAVCD combines launch vehicle design subprograms with a modified version of a previously developed optimization technique to perform the optimization analysis with only a small fraction of a number of design evaluations required by traditional parametric comparison methods.[1] In 1988, HAVCD was further developed under IR&D to support an all liquid booster propellant study under NASA contract.[2]

HAVCD uses specialized conceptual/preliminary design subprograms. The hybrid study required modifications and additions to the previous subprograms. A flow diagram of the hybrid booster model is shown in Figure B-13. The subprograms that were used in this study are:

- AIREZ - aerodynamics
- WITNEW - consolidating weights routine and configuration determinator.
- SOLID - hybrid performance plus required oxidizer and solid propellant required.
- NOSE - nose structure, avionics, recovery system.
- TANK - ox tank, solid case, interstage sizing, both structure and dimensions.
- PRESS - sizes the pressurant tanks for all of the configurations.

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1. G. T. Eckard and M. J. Healy, "Airplane Responsive Engine Selection," Air Force Aero Propulsion Laboratory, Wright Patterson Air Force Base, Ohio, April 1978, AFAPL-TR-78-13.

2. V. Weldon, M. Dunn, L. Fink, D. Phillips, E. Wetzel, "Final Report Booster Propulsion/Vehicle Impact Study," Boeing Aerospace, Seattle, Washington, June 1988, NAS8-36944.

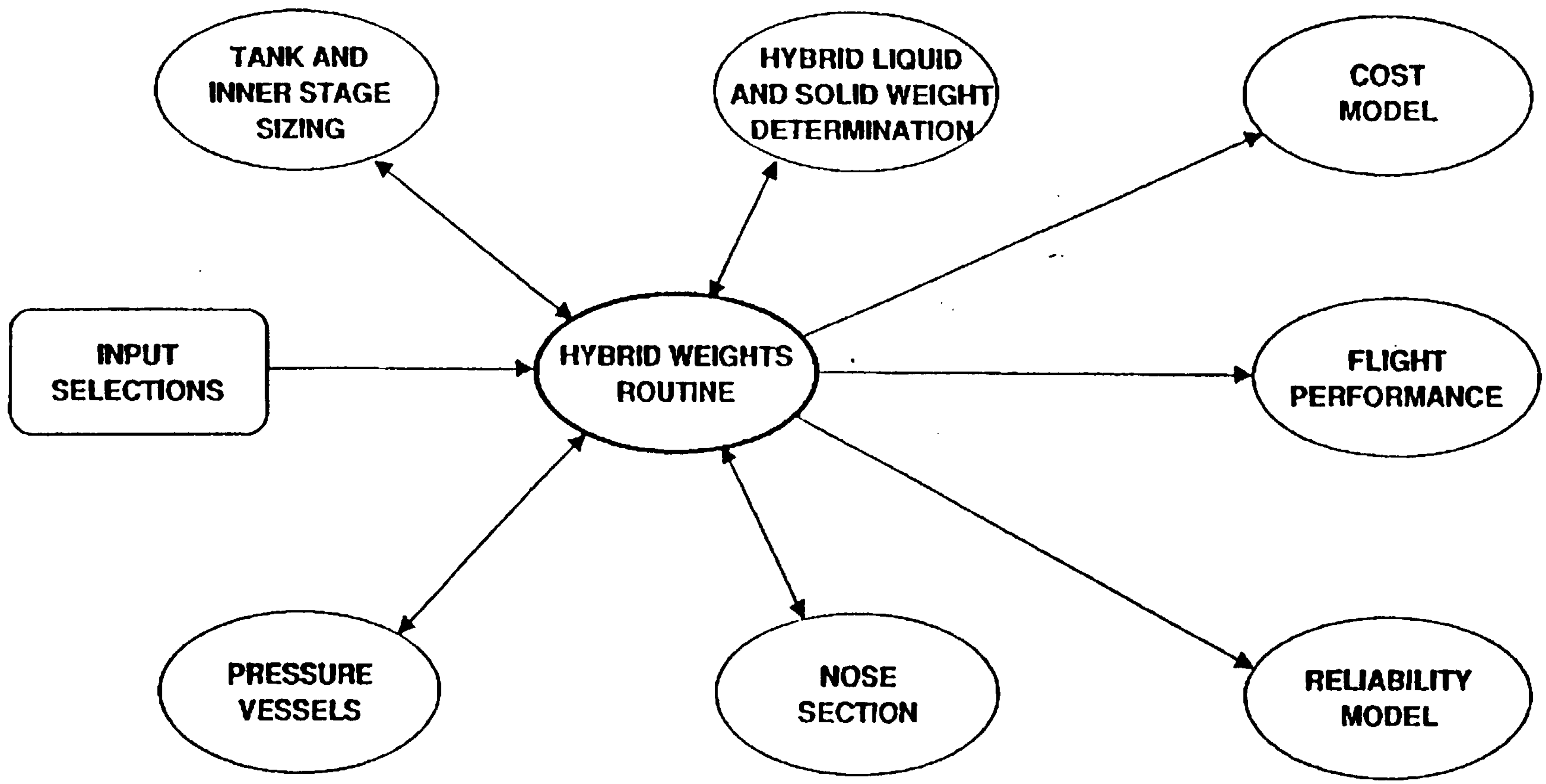

Figure B-13. Hybrid Computer Flow Diagram.

- COSTIT - design, manufacturing through life cycle cost.
- RELIB - single booster subsystem and system reliability.
- NTOP - trajectory performance.

AIREZ relies on a blend of simplified aerodynamic theory and empirical relationships which result in acceptable agreement with wind tunnel test data. The subprogram generates a table of axial and normal aerodynamic force coefficients as a function of Mach number (Mach 0.3 to 20) and angle of attach (-10° to 60°) based on airframe geometry determined from WITNEW. The performance of the full-size hybrid booster was evaluated as a replacement for the shuttle SRBs. The core vehicle matched the thrust level and drag of the shuttle and external tank. The aerodynamic drag routines were modified to account for the wave drag impact from the external tank to the hybrid boosters or from the hybrid boosters to the external tank. This lowered the drag coefficients with booster length.

WITNEW is the collection routine for the output from the subprograms. It sets up the configuration to be evaluated and calls on the appropriate subprograms to get a physical size, component weights, component locations, center of gravity travel, gross liftoff weight (GLOW), empty weight, shutdown weight, etc. The program cycles through all of the subprograms until system and subsystem weights converge to a constant number. Files are set up that would be used by COSTIT, RELIB, and NTOP.

SOLID determines the flight oxidizer and solid propellant load from the given ASRM thrust trace, the specific impulse (I_{sp}) tables and the input variables (such as mixture ratio, operating pressure, expansion ratio, etc.) SOLID adjusts the I_{sp} for fluids lost overboard such as turbine exhaust gas (from the gas generator or from a methane/LOX preburner) and/or thrust vector control (TVC) fluid (from either the gas generator or from the oxidizer). This program sets up the time, thrust, I_{sp}, and expansion ratio file that NTOP used to determine booster performance during ascent.

PRESS determines the pressurant tank volume, tank size and shape, and pressurate weight initially in the pressurant tank to the pressurant in the ox tank at thrust termination. The program can use either pure helium or tridyne (a mix of helium, hydrogen, and oxygen) as the pressurant. TANK is called to determine the wall thickness, ellipsoidal ratio of the dome and the vessel weight.

COSTIT is a program that uses cost algorithms for each component generated from WITNEW to calculate the design cost, first unit manufacturing cost, and the total manufacturing cost based on the delivered component quantity. Total acquisition and DDT&E costs is calculated based on the design and manufacturing costs. Operational cost is based on the total system weight of the boosters and total missions to be flown.

RELIB computes the reliability of each subsystem and the reliability of the overall system. Depending on the number of required components and the number of components used in the system, each delivered component reliability is calculated and is available to be integrated into the subsystem reliability and the overall system reliability.

NTOP flies the hybrid boosters to their separation point, and a shuttle and an external tank to a low Earth orbit (150 nm circular at 28° East). The shuttle and external tank liftoff weight was determined to be 1,840,600 pounds with 1,578,600 pounds of propellant and a delivered vacuum I_{sp} of 452.4 seconds. No fluids were assumed to be lost from the shuttle during ascent except thrusting propellant. The flight profile used in this study was a vertical ascent to a point where a continued gravity turn would deliver the shuttle to an apogee altitude of 50 nm. The booster thrust profile that was used for each mission is shown in Figure B-14. As a point of reference, the program was set up to fly representative ASRM boosters with the shuttle, and together they delivered about 73,500-pounds payload to the above orbit. The staging velocity was 4,800 ft/sec. Peak dynamic pressure was determined to be 680 lb/ft^2 (see Figure B-15), with a peak acceleration of 2.67 g's (see Figure B-16). Time did not permit core vehicle constructions for the quarter-size boosters; these were not flown.

Optimization equations can be generated using the method of steepest descent. The main feature of this optimization technique is that a minimal number of designs have to be run on the HAVCD program, thereby allowing optimized designs to be derived quickly. The latin squares method is used for optimization and requires (n+a)2 where "n" is the number of independent variable and "a" is 1 when "n" is not a prime number and is 2 when "n" is a prime number. For 8 independent variables (8+1)2=21 cases are required to be

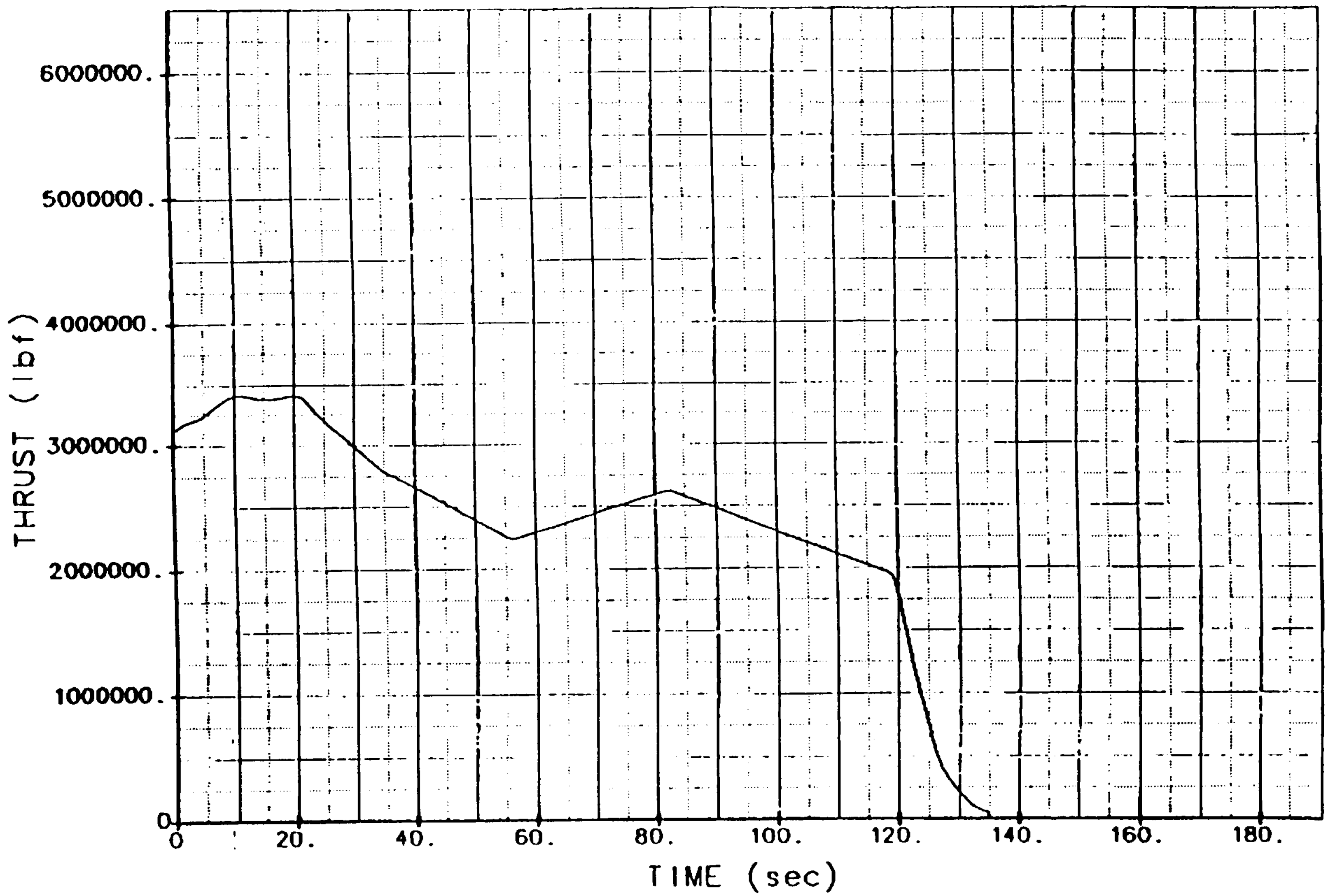

Figure B-14. ASRM Thrust Versus Time.

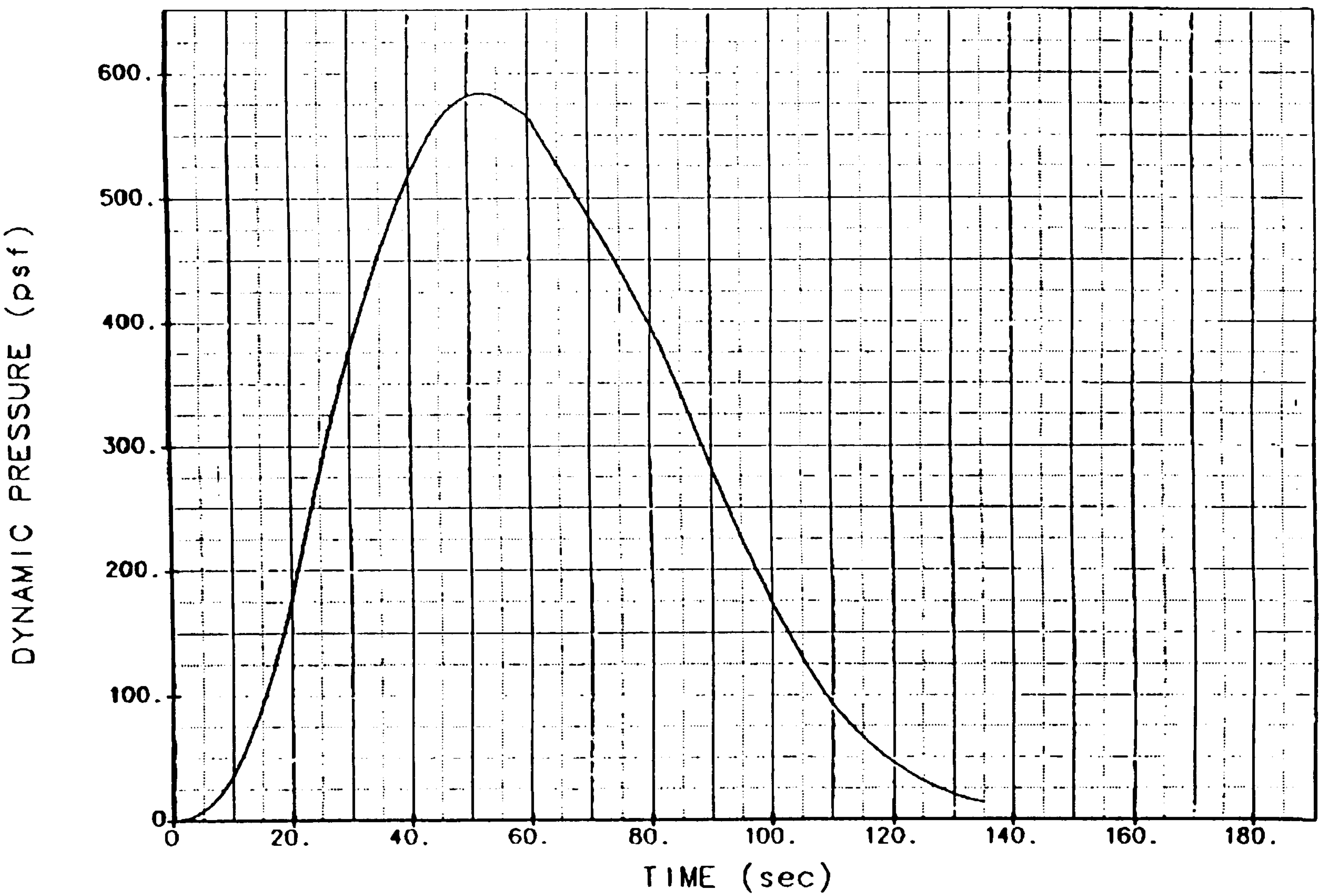

Figure B-15. ASRM Dynamic Pressure Versus Time.

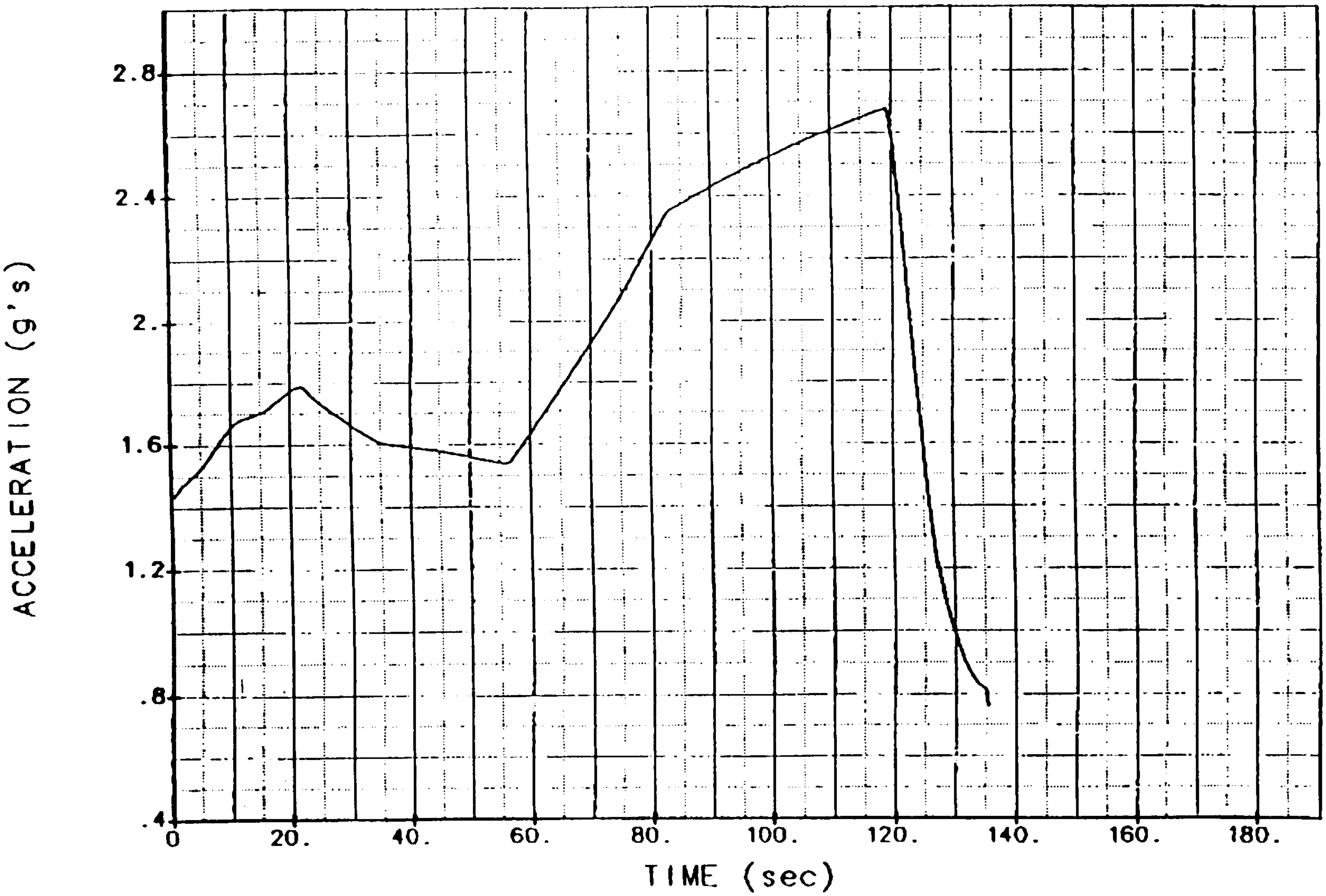

Figure B-16. ASRM Acceleration Versus Time.

run. The time savings is evident when one considers that a traditional carpet plot approach would require 65,536 designs to be evaluated for 8 variables (4 levels per variable requires 4 to the 8th power number of cases). At about 30 minutes to derive a design on a VAC 8300 computer, the time savings is substantial. Once the equations are obtained, an optimization can be performed in under ten seconds. Any of the dependent variables can be optimized or used as a constraint.

Computer Model Assumptions

Oxidizer Tanks

Upper and lower dome thicknesses were determined based on liquid level pressure from the minimum required pressure to the hydrostatic head pressure developed due to a 3g maximum allowed ascent acceleration. The cylinder wall thickness was based on the average of upper and lower dome pressure, assuming in practical application the cylinder walls would be tapered based on a representative pressure gradient. The cylinder wall thicknesses were evaluated for local buckling, and stiffeners were added or a slight increase in wall thickness would occur if required to prevent wall buckling. Side loads due to booster to core vehicle moments and gust loads were not considered in tank wall sizing.

Tank ullage was assumed to be 2 percent of the total loading oxidizer volume. Reserved propellant was calculated at 2 percent of the flight oxidizer load weight.

IM7 tank dome ellipsoidal ratio and tank weight equations were provided by ARC. No provisions were made available for local wall buckling.

Turbopumps

Turbopump equations assumed that a boost pump and a main pump plus turbines for each would be required. Horsepower required by each was calculated based on the maximum oxidizer flow rate and the total head pressure each pump was required to deliver. Turbine flow reduced the I_{sp} of the system and added solid propellant or methane/LOX was added as a function of the turbine inlet temperature and delivered horsepower. Turbine exhaust was assumed to not contribute to the overall system thrust.

Structural Weights

The aft skirt on the full-size booster used the weight of the current shuttle solid rocket boosters. It was assumed that a vehicle of this type would require support structure such as the aft skirt, but it was not clear how this would be modeled for different body diameters. An aft skirt weight of 13,722 pounds was assumed for all full-size boosters. No aft skirt was used on the quarter-size boosters.

The connecting truss, between the core vehicle and the boosters, weights were calculated based on the weight of each booster (full or quarter size) and the maximum thrust level of each and along with the maximum thrust level of the shuttle.

Interstage wall thickness was based on localized buckling and the load it was supporting at 3g's. No bending moments were considered.

Quarter-Size Boosters

The quarter-size boosters used the same weight and sizing algorithms as the full-sized booster. Thrust levels were reduced by one-fourth, but insulation thickness in the motor case and combustion chambers remained the same as the full-sized booster. Avionics, batteries, and wiring was also assumed to remain that of the full-sized booster. Single string components were assumed in the quarter-size booster, such as one pump, one throttle valve, one isolation valve, etc.

Sample Computer Model Variable Inputs

Figure B-17 shows the list of variables that were available to be changed from run-to-run. The values shown were those used for the full-sized reference vehicle.

Typical Computer Model Output

Figure B-18 is a brief list of the component size and weights for the reference expendable booster. Cost and reliability results are also included and are shown in Figures B-19 and B-20, respectively.

Reuseable Booster LCC

Calculation of the LCC of recoverable/reuseable boosters requires the definition of the reuseable components' design life. The attrition rate, and the cost of refurbishment. An example section of a reuseable booster input sheet is shown on Figure B-21.

Refurbishment costs for SRM components were obtained from STACEM. Refurbishment costs for liquid oxidizer components were assumed to be 25 percent of TFU.

Reference Trade Data

Attached are summary tables of the reference conditions and materials trades that were completed. Also attached are the results of the LITVC, structural margins, reserve propellant, and propellant volumetric loading trade studies.

```
DEDIAM=2.           !DIAMETER OF OX TANK - FT
DISBTK=2.           !DISTANCE BETWEEN TANK ASSEMBLIES - FT
RPTINS=.0127        !EXTERNAL TANK INSULATION DENSITY LB/IN3
SPWFT=2.            !CONCENTRIC TANK SEPARATION DISTANCE - IN
TFSOFI=0.           !FUEL TANK INSULATION THICKNESS - IN
TOSOFI=0.           !OX TANK INSULATION THICKNESS - IN
ELRTNK=1.4          !ELLIPSOIDAL RATIO FOR TANKS IF NOT CALCULATED
ELRPTN=1.4          !ELLIPSOIDAL RATIO FOR PRESS TANKS IF NOT CALCULATED
EXPR=15.            !BOOSTER NOZZLE EXPANSION RATIO
RSOLID=6.5          !OUTSIDE RADIUS OF SOLID CASE - FT
RZERO=2.333333      !STARTING GRAIN RADIUS  - FT
CONANG=18.          !NOSE SECTION CONE ANGLE - DEGREES
ALTHK=.030          !WALL THICKNESS OF AL LINER - IN
SLINSU=.15          !INSUL. THK FOR SOLID UPPER SECT. IN
SLINSL=5.0          !INSUL. THK FOR SOLID LOWER SECT. IN
THKINJ=8.           !THICKNESS OF INJECTOR - IN.
THRINL=.5           !RATIO OF THROAT AREA TO THROAT INLET FOR GG
CMBRAT=5.           !COMBUSTION CHAMBER LENGTH - FT
HLFANG=20.          !NOZZLE HALF ANGLE - DEGREES
GYMBOL=5.           !NOZZLE GYMBOL ANGLE - DEGREES
DTYCYL=.5           !TVC DUTY CYCLE
GASTVC=1.           !1-GG POWERED; 2-OX POWERED TVC FLUID INJ
GAPNOZ=.5           !GAP NOZZLE WILL MISS SKIRT WHEN GYMBOLED - FT


TRBPRS=10.          !TURBINE PRESSURE RATIO INLET/OUTLET
RSABPC=1.1          !PRESSURE INLET RATIO TO COMBUSTION CHAMBER
RSABPE=1.15         !PRESSURE INLET RATIO TO PRE-BURNER
PDOLIN=21.          !DP OF OX LINE - PSI
PDFLIN=5.           !DP OF FUEL LINE - PSI
PMAX=10000.         !HE TANK INITIAL PRESSURE
GPRATO=1.           !PRESSURE RATIO ABOVE TANK PRESSURE
FTPRES=15.          !MINIMUM FUEL TANK PRESSURE - PSIA
OTNPSP=25.          !OX PUMP SUPPLY PRESSURE - PSIA
PC=1000.            !STARTING PC
DPINJ=250.          !DP OF INJECTOR
DPOXV=32.           !DP OF MAIN OX ISO VALVE
DPMISC=14.          !DP OF MISC ITEMS
DPGLIN=10.          !DP OF GAS LINE
DPREG=22.           !DP OF REGULATOR
REGUNC=.01          !PRESSURE REGULATOR UNCERTAINTY
REGRAT=1.15         !MINIMUM REGULATOR OPERATING PRESSURE RATIO


PTYPE=11.           !CONFIGURATION TYPE
PMRO=1.5            !STARTING MIXTURE RATIO
AXMAX=3.            !MAXIMUM ALLOWED ACCELERATION - G'S
FS=1.6              !STRUCTURAL SAFETY FACTOR
TAUTO=100.          !AUTOGENOUS GAS TEMPERATURE - F
CONH20=95.          !PEROXIDE CONCENTRATION IN PERCENT
ELSTAR=10.          !COMBUSTION CHAMBER 'L' STAR
GG=1                !GAS GENERATOR=1, NORMAL HYBRID=0
TEMPIN=60.          !HE TANK INITIAL GAS TEMP
TEMP=-15.           !TEMP OF HE GAS IN OX TANK DEG F
ITERAT=300          !STEPS IN THRUST PROFILE INTEGRATION
TEMTRB=1800.        !TURBINE GAS TEMP FROM GG
EFFISP=.925         !IMPUSE EFFICENCY
EFFCST=.925         !CSTAR EFFICENCY
```

Figure B-17. Computer Input Variables.

```
OXRSRV=.02          !OX RESERVE
FLRSRV=.02          !FUEL RESERVE
OULLG=.02           !OX GAS ULLAGE
FULLG=.02           !FUEL GAS ULLAGE
FACNOZ=1.3          !NOZZLE WEIGHT FACTOR FOR TVC
DENSL1=.04          !SOLID INSUL. DENSITY #/IN3
DENINJ=.1           !DENSITY OF INJECTOR #/IN3

RCBOOS=31.          !2X RECOV.; 3X NOT RECOV.:XO NO TVC; X1 GYMBOLED, X2 INJ.
CPOXG=.17           !SP HT GOX
CPOXL=.4            !SP HT LOX
TRBFAC=3.           !TURBINE POWER              1-TOPPING, 2-METHANE,
CYIELD=1.           !YIELD STRENGTH FACTOR    * 3-GG REVERSED OX DOME
CULT=1.             !ULTIMATE STRENGTH FACTOR * 4-GG NORMAL OX DOME

PTMAT=5             !PRESS TANK MATERIAL                1=ALUMINUM
OTMAT=5             !OX TANK MATERIAL                   2=AL-LI
FTMAT=5             !FUEL TANK MATERIAL                 3=TITANIUM
STRMAT=1            !STRUCTURE MATERIAL                 4=STAINLESS STEEL
LNMAT=4             !LINE MATERIAL                      5=IM7 CARBON FIBER
INSMAT=1            !INNER STAGE MATERIAL               6=A6AC CARBON STEEL
SLMAT=5             !SOLID CASE MATERIAL
SKTMAT=1            !SKIRT MATERIAL
CMBMAT=5            !COMBUSTION CHAMBER MATERIAL

FAX=0.              !NON AXIAL FORCES
FSIDE=0.
DSIDEI=0.
FVERT=0.
DVERT1=0.
DHIGH1=0.

QNTY=1              !# OF HALF THE TOTAL BOOSTERS PER VEHICLE
CASES=1             !# OF HYBRID MOTORS PER BOOSTER
PUMPS=4             !# OF TURBO PUMPS
PUMPSR=3            !# OF TURBO PUMPS REQUIRED
HEVLV=3             !# OF HELIUM VALVE
HEVLVR=3            !# OF REQ'D HELIUM VALVE
HEPYR=2             !# OF HELIUM PYRO ISO VALVES
HEPYRR=1            !# OF REQ'D HELIUM PYRO ISO VALVES
HEREG=1             !# OF HELIUM REGULATORS
HEREGR=1            !# OF REQ'D HELIUM REGULATORS
HERLF=1             !# OF HELIUM RELIEF VALVES
HERLFR=1            !# OF REQ'D HELIUM RELIEF VALVES
HESRV=1             !# OF HELIUM SERVICE VALVES
HESRVR=1            !# OF REQ'D HELIUM SERVICE VALVES
OXVLV=4             !# OF OX VALVES IN SYSTEM
OXVLVR=4            !# OF REQ'D OX VALVES IN SYSTEM
OXPYR=4             !# OF OX PYRO ISO VALVES
OXPYRR=4            !# OF REQ'D OX PYRO ISO VALVES
THVLV=4             !# OF THROTTLE VALVES
THVLVR=4            !# OF REQ'D THROTTLE VALVES
OXSRV=1             !# OF OX SERVICE VALVES
OXSRVR=1            !# OF REQ'D OX SERVICE VALVES
```

Figure B-17. Computer Input Variables (Cont'd).

```
PRVLV=1       !# OF METHANE VALVES IN SYSTEM
PRVLVR=1      !# OF REQ'D METHANE VALVES IN SYSTEM
PRPYR=2       !# OF METHANE PYRO ISO VALVES
PRPYRR=1      !# OF REQ'D METHANE PYRO ISO VALVES
PRRLF=1       !# OF METHANE RELIEF VALVES
PRRLFR=1      !# OF REQ'D METHANE RELIEF VALVES
PRSRV=1       !# OF METHANE SERVICE VALVES
PRSRVR=1      !# OF REQ'D METHANE SERVICE VALVES
OXREG=1       !# OF OX REGULATORS
OXREGR=1      !# OF REQ'D OX REGULATORS
OXRLF=1       !# OF OX RELIEF VALVES
OXRLFR=1      !# OF REQ'D OX RELIEF VALVES
AVION=1       !# OF AVIONICS
AVIONR=1      !# OF REQ'D AVIONICS
WIRES=1       !# OF WIRES
WIRESR=1      !# OF REQ'D WIRES
BATRY=1       !# OF BATTERIES
BATRYR=1      !# OF REQ'D BATTERIES
INSTR=1       !# OF INSTRUMENTATION
INSTRR=1      !# OF REQ'D INSTRUMENTATION
PARAC=1       !# OF PARACHUTES
PARACR=1      !# OF REQ'D PARACHUTES
NOSES=1       !# OF NOSE SHELLS
NOSESR=1      !# OF REQ'D NOSE SHELLS
OXTNK=1       !# OF OX TANKS
OXTNKR=1      !# OF REQ'D OX TANKS
OXLIN=4       !# OF OX LIQ LINES
OXLINR=4      !# OF REQ'D OX LIQ LINES
GOXLN=1       !# OF GAS OX LINES
GOXLNR=1      !# OF REQ'D GAS OX LINES
HELIN=1       !# OF HE LINES
HELINR=1      !# OF REQ'D HE LINES
SLDIG=1       !# OF SOLID MOTOR IGNITERS
SLDIGR=1      !# OF REQ'D SOLID MOTOR IGNITERS
HETNK=1       !# OF HELIUM TANKS
HETNKR=1      !# OF REQ'D HE TANKS
PPTNK=1       !# OF METHANE TANKS
PPTNKR=1      !# OF REQ'D TANKS
TVCVS=4       !# OF FLUID INJECTION TVC VALVES
TVCVSR=4      !# OF REQ'D FLUID INJECTION TVC VALVES
```

Figure B-17. Computer Input Variables (Cont'd).

TOTAL INITIAL WEIGHT=1213138.14Lb
EMPTY WEIGHT= 83218.23Lb
EXPENDED OX WEIGHT= 660190.69Lb
TVC OX PROP.= 0.00Lb
TURBINE FUEL= 5976.30Lb
INITAL C.G.= 79.71Ft
EMPTY C.G.= 124.57Ft

OVERALL LENGTH= 166.64Ft
CUT OFF WT= 106842.77Lb
EXPENDED FUEL WEIGHT= 446104.69Lb
TVC FUEL PROP.= 0.00Lb
TOTAL EXPENDED PROPELLANT= 1106295.38LB
CUT OFF C.G.= 101.72Ft

STARTING M.R.= 1.50
SAFETY FACTOR= 1.60

STARTING PC=1000.00 PSIA
NUMBER OF HYBRED UNITS=1

--- NOSE SECTION SIZE ---
BASE DIA. = 14.00Ft
NOSE TIP RAD= 1.27FT
C.G. FROM NOSE TIP= 10.21Ft
LOCATION FROM NOSE TIP= 0.00Ft

OVERALL LENGTH=18.91Ft
CYL LEN= 0.00FT
WEIGHT= 1523.65Lb
TO BOTTOM = 18.91Ft

--- HELIUM TANK SIZE ---
MATERIAL: IM7 CARBON FIBER
OUTSID DIAMETER= 8.53Ft
DOME HT= 2.92Ft
DOME THICK.= 2.869In
VESSEL WEIGHT= 3354.23Lb
INIT WEIGHT= 4697.94Lb
HE WEIGHT= 1176.00Lb
INIT PRESS=10000.PSIA
LOCATION FROM NOSE TIP= 13.08Ft

LENGTH= 5.83Ft
CYL LEN= 0.00Ft
CYL THICK.= 0.000In
ALUMINUM LINER= 44.20Lb
SHUTDOWN WEIGHT= 3544.04Lb
C.G. FROM CYL TOP= 2.91Ft
FINAL PRESS= 144.PSIA
TO BOTTOM= 18.91Ft

--- HELIUM TANK VALVING SYSTEM ---
HE PYRO VAVLE WT= 14.91Lb
PRESSURE REGULATOR WT= 17.81Lb
HE SERVICE VALVE WT= 29.81Lb
TOTAL VALVE WT= 77.43Lb

QUANTITY= 2
QUANTITY= 1
QUANTITY= 1

--- INTER STAGE (NOSE TO OX TANK) ---
MATERIAL: 2219-T87 ALUMINUM
DIA TOP= 14.00Ft
LENGTH= 5.00Ft
WALL THICK.= 0.040In
CG FROM TOP= 2.50Ft
LOCATION FROM NOSE TIP= 18.91Ft

DIA BOT= 14.00FT
WEIGHT= 137.03Lb
STIFFINERS REQUIRED= 0

TO BOTTOM= 23.91Ft

--- OXIDIZER TANK ---
MATERIAL: IM7 CARBON FIBER
DIAMETER= 14.00Ft
DOME HT= 4.51Ft
UPPER DOME THICK.= 0.036In
CYL THICK.= 0.111In
OX TANK VOL= 9639.66FT3
TOT OXIDIZER WEIGHT= 673593.63Lb
RESIDUAL OXIDIZER= 199.13Lb
INSULATION= 0.00Lb
INIT WEIGHT= 677312.63Lb
INIT. C.G. FROM CYL TOP= 29.27Ft
UPPER DOME PRESS= 93.PSIA
LOCATION FROM NOSE TIP= 19.40Ft

TANK LENGTH= 70.31Ft
CYL LEN= 61.29Ft
LOWER DOME THICK.= 0.072In
STIFFINERS REQUIRED= 0.
VESSEL WEIGHT= 2442.97Lb
RESERVE OXIDIZER=13203.81Lb
PRES GAS WEIGHT= 1153.91Lb
OX LINER= 1276.04Lb
EMPTY WEIGHT= 3719.01Lb
FINAL C.G.= 21.16Ft
LOWER DOME PRESS= 187.PSIA
TO BOTTOM= 85.20Ft

Figure B-18. LOX With Turbopumps, Gas Generator System.

--- LOX VALVING SYSTEM ---

OXIDIZER VALVE WT	= 241.17Lb	QUANTITY= 4
OXIDIZER PYRO WT	= 120.59Lb	QUANTITY= 4
METHANE THROTTLE VALVE WT	= 39.54Lb	QUANTITY= 4
OX SERVICE VALVE WT	= 120.59Lb	QUANTITY= 1
OX RELIEF VALVE WT	= 4.44Lb	QUANTITY= 1
TOTAL VALVE WT	= 1730.23Lb	

--- BOOST PUMP SIZE ---

DIAMETER= 1.38Ft	LENGTH= 1.54Ft
WEIGHT/PUMP= 292.16Lb	TOTAL WT= 1168.66Lb
PUMPS= 4	FLOWRATE/PUMP= 1628.89Lb/Sec
DELTA P= 54.47PSIA	SPEED= 2960RPM
HORSE POWER= 417	PUMP EFFICIENCY= 77.97%
NS= 8825	INLET PRESS.= 25.00PSIA
VAPOR PRES= 14.34PSIA	PUMP CG FROM TOP= 0.77Ft
LOCATION FROM NOSE TIP= 152.69Ft	TO BOTTOM= 154.23Ft

--- MAIN PUMP ---

DIAMETER= 1.56Ft	LENGTH= 2.27Ft
WEIGHT/PUMP= 623.94Lb	TOTAL WT= 2495.77Lb
PUMPS= 4	FLOWRATE/PUMP= 1628.89Lb/Sec
DELTA P= 1034.88PSIA	SPEED= 6092RPM
HORSE POWER= 7455	PUMP EFFICIENCY= 83.06%
NS= 1995	INLET PRESS.= 79.47PSIA
VAPOR PRES= 14.34PSIA	PUMP CG FROM TOP= 1.14Ft
LOCATION FROM NOSE TIP= 154.23Ft	TO BOTTOM= 156.50Ft

--- TURBINE ---

TURBINE FLOWRATE= 752.01Lb/Sec	TURBINE TEMPERATURE= 1800F
ISP REDUCED BY 0.54%	FUEL REQUIRED= 5976.30Lb

TOTAL PUMP ASSEM. LEN= 3.81Ft	TOTAL PUMP ASSEM. WEIGHT= 3664.43Lb

--- OXIDIZER PROPELLANT LINE TO COMBUSTION CHAMBER ---

MATERIAL: AISI 301 STAINLESS

OX LINE DIA.= 7.00In	LENGTH= 68.89Ft
NUMBER OF LINES= 4	WEIGHT/LINE= 348.97Lb
TOTAL LINE WT= 1395.89Lb	

--- SOLID FUEL CASE ---

MATERIAL: IM7 CARBON FIBER

DIAMETER= 13.00Ft	LENGTH= 55.41Ft
DOME HT= 4.19Ft	CYL LEN= 51.22Ft
RATIO PORT TO THROAT AREA= 1.46	INIT PORT RAD.= 2.33Ft
SOLID CASES= 1	GRAIN LENGTH= 51.22Ft
UPPER DOME THICK.= 0.370In	CYL THICK.= 0.627In
STIFFENERS REQUIRED= 0.	AVG DEL ISP=293.66SEC
CASE WEIGHT= 11112.80Lb	TOTAL FUEL WEIGHT= 455026.78Lb
RESERVE FUEL= 8922.09Lb	INSULATION= 3039.68Lb
INIT WEIGHT= 469679.25Lb	EMPTY WEIGHT= 14652.47Lb
IGNITER= 500.0 Lb	
INIT. C.G. FROM CYL TOP= 25.61Ft	EMPTY C.G.= 25.61Ft
STARTING PRESS= 1000.PSIA	MAXIMUM PRESS= 1089.PSIA
LOCATION FROM NOSE TIP= 81.00Ft	TO BOTTOM= 136.41Ft

Figure B-18. LOX With Turbopumps, Gas Generator System (Cont'd).

ORIGINAL PAGE IS
OF POOR QUALITY

--- CONVERGENT SECTION ---
MATERIAL: IM7 CARBON FIBER
CASE WEIGHT= 349.51Lb
TOTAL WT= 3579.41Lb
LENGTH= 3.80Ft
LOCATION FROM NOSE TIP= 136.41Ft

INSULATION= 3229.90Lb
CG FROM TOP= 1.52Ft
OUTLET DIA.= 5.47Ft
TO BOTTOM= 140.22Ft

--- GG INJECTOR ---
INJECTOR DIA.= 5.47Ft
WEIGHT= 2704.18Lb
LOCATION FROM NOSE TIP= 140.22Ft

LENGTH= 8.00In
TO BOTTOM= 140.89Ft

--- COMBUSTION CHAMBER ---
MATERIAL: IM7 CARBON FIBER
WEIGHT CHAMBER= 138.16Lb
TOTAL WT= 2421.62Lb
WALL THICK.= 0.20In
LENGTH= 5.00Ft
LOCATION FROM NOSE TIP= 140.89Ft

WEIGHT INS= 2283.46Lb
CG FROM TOP= 2.50Ft
INSULATION THICK.= 5.00In
OUTSIDE DIA.= 5.47Ft
TO BOTTOM= 145.89Ft

--- THROAT SIZE ---
THROAT ID DIAMETER= 3.87Ft
WEIGHT=16656.26Lb
LOCATION FROM NOSE TIP= 145.89Ft

LENGTH= 4.40Ft
CG FROM TOP= 26.39Ft
TO BOTTOM= 150.28Ft

--- NOZZLE SIZE ---
DIA. NOZZLE EXIT= 14.97Ft
WEIGHT= 8711.62Lb
CG FROM TOP= 8.18Ft
LOCATION FROM NOSE TIP= 150.28Ft

LENGTH= 16.36Ft
EXP RATIO= 15.0
TO BOTTOM= 166.64Ft

--- TVC ACTUATOR ---
WEIGHT= 2328.00Lb

--- BASE SKIRT SIZE ---
MATERIAL: 2219-T87 ALUMINUM
DIA TOP= 13.00Ft
LENGTH= 20.95Ft
CG FROM TOP= 10.47Ft
LOCATION FROM NOSE TIP= 136.41Ft

DIA BASE= 13.68FT
WEIGHT=13722.00Lb
TO BOTTOM= 157.36Ft

--- BOOSTER TO CORE TRUSS ---
TRUSS WEIGHT= 1165.56Lb

--- BOOSTER SEPARATION SYSTEM ---
SEPARATION SYSTEM WEIGHT= 1487.00Lb

--- RANGE SAFETY ---
RANGE SAFETY WEIGHT=144.00Lb

Figure B-18. LOX With Turbopumps, Gas Generator System (Cont'd).

REQUIRED ASSEMBLIES= 300 INCLUDING 0 SPARES

--- COMPONENT WEIGHT, DESIGN COST, FIRST UNIT COST, AND QUANTITY COST ---

COMPONENT	WEIGHT-LB	DESIGN-K$	MANUF(1)-K$	MANUF:TOTAL-K$	QTY
AVIONICS	77	951	904	133,987	300
WIRING	260	3,582	273	40,462	300
BATTERIES	45	213	124	18,378	300
INSTRUMENTATION	45	101	598	88,633	300
NOSE SHELL	1,095	474	1,345	199,350	300
HE TANK	3,354	5,785	3,423	507,343	300
HE LINER	44	70	155	22,973	300
HE PYRO VALVE	14	1	43	11,486	600
HE REGULATOR	17	1	50	7,410	300
HE SERVICE VLV	29	1	79	11,709	300
INTER STAGE	137	92	114	16,896	300
OXIDIZER TANK	2,442	2,665	1,975	292,726	300
OX LINER	1,276	918	1,490	220,841	300
OX ISO VALVE	241	3	61	29,351	1,200
OX PYRO VALVE	120	2	43	20,690	1,200
OX THROT. VALVE	39	1	24	11,548	1,200
OX SERVICE VLV	120	2	43	6,373	300
OX REGULATOR	17	1	50	7,410	300
OX RELIEF VLV	4	0	15	2,223	300
BOOST TURBINE	146	2,299	53	25,502	1,200
BOOST PUMP	146	2,299	53	25,502	1,200
MAIN TURBINE	311	4,910	98	47,155	1,200
MAIN PUMP	311	4,910	98	47,155	1,200
SOLID PROPEL.	455,026	32,908	1,281	189,864	300
OX LINE	348	41	84	40,418	1,200
SOLID IGNITER	500	2,014	100	14,821	300
SOLID INSUL.	3,039	4,221	439	65,066	300
SOLID CASE	11,112	7,182	1,558	230,920	300
CONVRG CASE	349	1,739	176	26,086	300
CONVRG INSL	3,229	494	12	1,778	300
INJECTOR	2,704	13,435	4,311	638,959	300
COMB. CASE	138	1,188	186	27,568	300
COMB. INSL	2,283	3,754	20	2,964	300
THROAT	16,656	8,479	1,634	242,185	300
NOZZLE	8,711	6,500	1,338	198,313	300
TVC ACT	2,328	793	2,009	297,766	300
AFT SKIRT	13,722	2,284	2,860	423,897	300
TRUSS	1,165	2,457	432	64,029	300
SEP SYS	1,487	3,525	2,446	362,536	300
COLUMN TOTALS		120,314	31,706	4,875,593	

SUBSYSTEM INTEGRATION	=	24,062K$
SUBSYSTEM ASSEMBLY	=	243,779K$
FINAL ASSEMBLY AND CHECKOUT	=	767,905K$
MANUF. COST PER UNIT	=	19,624K$
SYSTEMS ENGINEERING	=	32,147K$
SOFTWARE ENGINEERING	=	27,200K$
SYSTEMS TEST	=	37,145K$
TOOLING	=	820,382K$
MISC.	=	199,877K$
TOTAL SUPPORT FUNCTION COST	=	1,116,753K$
TOTAL ACQUISTION COST	=	7,148,410K$
DDT&E COST	=	1,084,512K$

Figure B-19. LOX With Turbopumps, Gas Generator System.

--- COST ASSUMPTION FACTORS FOR DESIGN AND MANUFACTURE ---

COMPONENT	DESIGN COMPLEXITY	OFF-THE-SHELF%	DESIGN MATURITY	MANUFACTURE COMPLEXITY	MATERIAL TYPE	LEARNING CURVE SLOPE%
AVIONICS	7	0	1	5	1	90
WIRING	5	0	1	5	1	90
BATTERIES	5	80	1	5	1	90
INSTRUMENTATION	5	80	1	5	1	90
NOSE SHELL	4	30	2	5	1	90
HE TANK	5	0	4	5	5	90
HE LINER	5	0	2	5	1	90
HE PYRO VALVE	5	100	2	5	1	90
HE REGULATOR	5	100	2	5	1	90
HE SERVICE VLV	5	100	2	5	1	90
INTER STAGE	2	50.	2	2	1	90
OXIDIZER TANK	5	0	4	5	5	90
OX LINER	5	0	2	5	1	90
OX ISO VALVE	5	100	2	5	1	90
OX PYRO VALVE	5	100	2	5	1	90
OX THROT. VALVE	5	100	2	5	1	90
OX SERVICE VLV	5	100	2	5	1	90
OX REGULATOR	5	100	2	5	1	90
OX RELIEF VLV	5	100	2	5	1	90
BOOST TURBINE	5	0	2	5	1	90
BOOST PUMP	5	0	2	5	1	90
MAIN TURBINE	5	0	2	5	1	90
MAIN PUMP	5	0	2	5	1	90
SOLID PROPEL.	5	0	2	7	1	90
OX LINE	2	50	2	2	4	90
SOLID IGNITER	5	0	2	5	1	90
SOLID INSUL.	5	0	2	5	1	90
SOLID CASE	5	0	2	5	5	90
CONVRG CASE	5	0	2	5	5	90
CONVRG INSL	5	0	2	5	1	90
INJECTOR	5	0	8	5	1	90
COMB. CASE	5	0	2	5	5	90
COMB. INSL	5	0	2	5	1	90
THROAT	5	0	2	5	1	90
NOZZLE	5	0	2	5	1	90
TVC ACT	2	80	5	5	1	90
AFT SKIRT	5	0	2	5	1	90
TRUSS	5	0	2	5	1	90
SEP SYS	5	0	2	5	1	90
MISC	9	100	8	9	1	90

--- FACTOR DEFINITION FOR ABOVE TABLE ---

DESIGN COMPLEXITY: 1-9; "1" FOR LOW, "9" FOR HIGH COMPLEXITY
OFF-THE-SHELF%: PERCENTAGE OF DESIGN THAT IS OFF-THE-SHELF.
DESIGN MATURITY: 1-8; "1" QUALIFIED, "8" CONCEPTUAL ONLY
MANUFACTURE COMPLEXITY: 1-9; "1" FOR LOW, "9" FOR HIGH COMPLEXITY
MATERIAL TYPE: "1" FOR ALUMINUM OR REFERENCE MATERIAL
"2" FOR ALUMINUM LITHIUM
"3" FOR TITANIUM
"4" FOR STAINLESS STEEL
"5" FOR GRAPHITE FIBER
"6" FOR D6AC STEEL
LEARNING CURVE SLOPE IN PERCENT

Figure B-19. LOX With Turbopumps, Gas Generator System (Cont'd).

--- NON-RECURRING OPERATIONS COST ---

LAUNCH & CONTROL CENTER	=	10,628K$
PAD & SITE PREPARATION	=	111,684K$
VEHICLE ASSY BUILDING	=	109,265K$
PROGRAM ADMINS. & FACIL. MODS	=	346,352K$
FACILITIES INITIAL SPARES	=	10,215K$
LAUNCH CONTROL GSE	=	8,209K$
PAD GSE	=	88,915K$
IACO GSE	=	166,391K$
MOBILE EQUIPMENT	=	377,440K$
INITIAL SPARES	=	44,267K$
GROUND SECTOR SOFTWARE	=	17,057K$
TOTAL NON-RECURRING OPTS COST	=	1,290,427K$

--- RECURRING OPERATIONS COST ---

FOR YEARS LAUNCHES PER YEAR		0 - 1 3	1 - 2 6	2 - 3 9	3 - 14 12
TECH SYSTEM MANAGEMENT	=	26,341K$	33,806K$	39,120K$	43,388K$
PRELAUNCH OPERATIONS/CHECKOUT	=	73,169K$	93,908K$	108,666K$	120,524K$
PROPELLANT COST - LIQUID ONLY	=	210K$	420K$	630K$	840K$
HELIUM COST	=	55K$	111K$	167K$	222K$
MISSION & LAUNCH CONTROL	=	29,267K$	37,563K$	43,466K$	48,209K$
REPLENISHMENT SPARES - FC/GSE	=	10,374K$	10,704K$	10,903K$	11,046K$
YEARLY OPERATIONS COST	=	139,419K$	176,514K$	202,954K$	224,232K$

TOTAL OPERATIONS COST = 2,985,444K$

TOTAL LIFE CYCLE COST = 11,424,283K$

Figure B-19. LOX With Turbopumps, Gas Generator System (Cont'd).

COMPONENT	UNITS REQ'D	UNITS OPER.	PRE-LAUNCH FAILS/ MILLION	PRE-LAUNCH OPER. HRS-CYC	BOOST FAILS/ MILLION	BOOST OPER. HRS-CYC	POST-BOOST FAILS/ MILLION	POST-BOOST OPER. HRS-CYC
AVIONICS	1	1	2.00	0.000	20.00	0.038	0.00	0.000
WIRING	1	1	0.13	0.000	1.50	0.038	0.00	0.000
BATTERIES	1	1	33.80	0.000	169.00	0.038	0.00	0.000
INSTRUMENTATION	1	1	6.80	0.000	155.00	0.038	0.00	0.000
NOSE SHELL	1	1	1.00	0.000	45.00	0.038	0.00	0.000
HE TANK	1	1	3.80	0.000	37.50	0.038	0.00	0.000
HE LINER	1	1	3.80	0.000	37.50	0.038	0.00	0.000
HE PYRO VALVE	1	2	1.10	0.000	0.00	0.000	0.00	0.000
HE REGULATOR	1	1	2.60	0.000	55.30	0.038	0.00	0.000
HE SERVICE VLV	1	1	1.60	0.000	0.00	0.000	0.00	0.000
INTER STAGE	1	1	0.40	0.000	1.00	0.038	0.00	0.000
AFT SKIRT	1	1	0.40	0.000	1.00	0.038	0.00	0.000
TRUSS	1	1	0.10	0.000	1.00	0.038	0.00	0.000
OX ISO VALVE	4	4	11.00	0.000	0.00	0.000	11.00	0.000
OX PYRO VALVE	4	4	8.00	0.000	0.00	0.000	0.00	0.000
OX THROT. VALVE	4	4	9.60	0.000	165.70	0.038	0.00	0.000
OX SERVICE VLV	1	1	1.60	0.000	0.00	0.000	0.00	0.000
OX REGULATOR	1	1	2.60	0.000	55.30	0.038	0.00	0.000
OX RELIEF VLV	1	1	1.60	0.000	9.80	0.038	1.00	0.000
OX LINE	4	4	0.00	0.000	1.10	0.038	0.00	0.000
OXIDIZER TANK	1	1	3.80	0.000	37.50	0.038	0.00	0.000
OX LINER	1	1	3.80	0.000	37.50	0.038	0.00	0.000
MAIN PUMP	3	4	0.00	0.000	267.75	1.000	0.00	0.000
BOOST TURBINE	3	4	0.00	0.000	267.75	1.000	0.00	0.000
BOOST PUMP	3	4	0.00	0.000	267.75	1.000	0.00	0.000
MAIN TURBINE	3	4	0.00	0.000	267.75	1.000	0.00	0.000
SOLID INSUL.	1	1	0.00	0.000	6.30	1.000	0.00	0.000
SOLID CASE	1	1	0.70	0.000	134.00	1.000	0.00	0.000
CONVRG CASE	1	1	0.70	0.000	134.00	1.000	0.00	0.000
CONVRG INSL	1	1	0.00	0.000	6.30	1.000	0.00	0.000
INJECTOR	1	1	0.00	0.000	45.00	1.000	0.00	0.000
COMB. CASE	1	1	0.70	0.000	134.00	1.000	0.00	0.000
COMB. INSL	1	1	0.00	0.000	6.30	1.000	0.00	0.000
SOLID PROPEL.	1	1	0.00	0.000	56.00	1.000	0.00	0.000
SOLID IGNITER	1	1	0.70	0.000	85.00	1.000	0.00	0.000
TVC ACT	1	1	0.00	0.000	321.00	1.000	0.00	0.000
THROAT	1	1	0.00	0.000	248.50	1.000	0.00	0.000
NOZZLE	1	1	0.00	0.000	248.50	1.000	0.00	0.000
SEP SYS	1	1	0.00	0.000	0.00	0.000	0.00	0.000

Figure B-20. LOX With Turbopumps, Gas Generator System.

COMPONENT	UNITS REQ'D	UNITS OPER.	RELIABILITY THRU PRE-LAUNCH	RELIABILITY THRU BOOST	RELIABILITY THRU POST-BOOST
AVIONICS	1	1	1.000000000	0.999999250	0.999999250
WIRING	1	1	1.000000000	0.999999944	0.999999944
BATTERIES	1	1	1.000000000	0.999993663	0.999993663
INSTRUMENTATION	1	1	1.000000000	0.999994188	0.999994188
NOSE SHELL	1	1	1.000000000	0.999998313	0.999998313
NOSE SECTION SUBSYSTEM			1.000000000	0.999985356	0.999985356
HE TANK	1	1	1.000000000	0.999998594	0.999998594
HE LINER	1	1	1.000000000	0.999998594	0.999998594
HE PYRO VALVE	1	2	1.000000000	1.000000000	1.000000000
HE REGULATOR	1	1	1.000000000	0.999997926	0.999997926
HE SERVICE VLV	1	1	1.000000000	1.000000000	1.000000000
HELIUM PRESS. SUBSYSTEM			1.000000000	0.999995114	0.999995114
INTER STAGE	1	1	1.000000000	0.999999963	0.999999963
AFT SKIRT	1	1	1.000000000	0.999999963	0.999999963
TRUSS	1	1	1.000000000	0.999999963	0.999999963
STRUCTURE SUBSYSTEM			1.000000000	0.999999888	0.999999888
OX ISO VALVE	4	4	1.000000000	1.000000000	1.000000000
OX PYRO VALVE	4	4	1.000000000	1.000000000	1.000000000
OX THROT. VALVE	4	4	1.000000000	0.999975145	0.999975145
OX SERVICE VLV	1	1	1.000000000	1.000000000	1.000000000
OX REGULATOR	1	1	1.000000000	0.999997926	0.999997926
OX RELIEF VLV	1	1	1.000000000	0.999999633	0.999999633
OX LINE	4	4	1.000000000	0.999999835	0.999999835
OXIDIZER TANK	1	1	1.000000000	0.999998594	0.999998594
OX LINER	1	1	1.000000000	0.999998594	0.999998594
OXIDIZER SUBSYSTEM			1.000000000	0.999969727	0.999969727
MAIN PUMP	3	4	1.000000000	0.999999570	0.999999570
BOOST TURBINE	3	4	1.000000000	0.999999570	0.999999570
BOOST PUMP	3	4	1.000000000	0.999999570	0.999999570
MAIN TURBINE	3	4	1.000000000	0.999999570	0.999999570
TURBO-PUMP SUBSYSTEM			1.000000000	0.999998281	0.999998281
SOLID INSUL.	1	1	1.000000000	0.999993700	0.999993700
SOLID CASE	1	1	1.000000000	0.999866009	0.999866009
CONVRG CASE	1	1	1.000000000	0.999866009	0.999866009
CONVRG INSL	1	1	1.000000000	0.999993700	0.999993700
INJECTOR	1	1	1.000000000	0.999955001	0.999955001
COMB. CASE	1	1	1.000000000	0.999866009	0.999866009
COMB. INSL	1	1	1.000000000	0.999993700	0.999993700
SOLID PROPEL.	1	1	1.000000000	0.999944002	0.999944002
SOLID IGNITER	1	1	1.000000000	0.999915004	0.999915004
SOLID MOTOR SUBSYSTEM			1.000000000	0.999393284	0.999393284

Figure B-20. LOX With Turbopumps, Gas Generator System (Cont'd).

COMPONENT	UNITS REQ'D	UNITS OPER.	RELIABILITY THRU PRE-LAUNCH	RELIABILITY THRU BOOST	RELIABILITY THRU POST-BOOST
TVC ACT	1	1	1.000000000	0.999679052	0.999679052
THROAT	1	1	1.000000000	0.999751531	0.999751531
NOZZLE	1	1	1.000000000	0.999751531	0.999751531
NOZZLE SUBSYSTEM			1.000000000	0.999182334	0.999182334
SEP SYS	1	1	1.000000000	1.000000000	1.000000000
ONE BOOSTER SYSTEM RELIABILITY			1.000000000	0.998524554	0.998524554

Figure B-20. LOX With Turbopumps, Gas Generator System (Cont'd).

REUSABLE BOOSTER INPUTS

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DEFAULT COMPONENT DESIGN LIFE 10

DEFAULT RECOVERY ATTRITION RATE 10%

	REUSE LIFE	RECOVERY ATTRITION	REFURB COST (% TFU)	REFURB COST (K $)	TOTAL COST (K $)
	==========	==========	==========	==========	==========
E & I				$445,113	$887,973
=======================					
AVIONICS	10	10%	25%	$266,927	$402,226
WIRING	1			$0	$152,961
BATTERIES	1			$0	$64,282
INSTRUMENTATION	10	10%	25%	$178,186	$268,504
RECOVERY SYSTEM				$70,589	$255,801
=======================					
PARACHUTES	10	10%	39%	$70,589	$255,801
STRUCTURES				$104,284	$768,539
=======================					
NOSE	10	10%	16%	$31,417	$231,535
FORWARD SKIRT	10	10%	16%	$0	$0
AFT SKIRT	10	10%	16%	$59,128	$435,753
ATTACHMENTS	10	10%	16%	$10,089	$74,350
INTER STAGE	10	10%	16%	$3,650	$26,902
OXIDIZER SYSTEM				$56,337	$4,049,234
=======================					
OXIDIZER TANK	1			$0	$1,017,039
TANK LINER	1			$0	$528,171
OXIDIZER PIPING	10	10%	25%	$29,842	$421,446
OX ISO VALVE	10	10%	25%	$2,213	$31,253
OX PYRO VALVE	10	10%	25%	$1,537	$21,706
OX SERVICE VALVE	10	10%	25%	$1,660	$8,383
OX REGUALTOR	10	10%	25%	$1,858	$9,384
OX RELIEF VALVE	10	10%	25%	$534	$2,697
PROPANE TANK	1	10%	50%	$0	$0
PRO THROTTLE VALVE	10	10%	25%	$854	$12,061
PRO ISO VALVE	10	10%	25%	$0	$0
PRO PYRO VALVE	10	10%	25%	$0	$0
PRO RELIEF VALVE	10	10%	25%	$0	$0
HE TANK	1	10%	50%	$0	$1,763,062
HE TANK LINER	1	10%	50%	$0	$34,813
HE ISO VALVE	10	10%	25%	$0	$0
HE PYRO VALVE	10	10%	25%	$1,572	$13,029
HE REGULATOR	10	10%	25%	$1,858	$9,384

Figure B-21. Reusable Booster Inputs.

BOEING HYBRID COMPUTER DATA

MIXTURE RATIO TRADE ALL PRESSURIZED

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.300	1000.	15.0	LOX	PRES	14.0	IM7	22810.	13.0	IM7	1257.K	173.	1618.M	10.880B	1.327B	3.053B	15.260B	88020.	1156.
1.400	1000.	15.0	LOX	PRES	14.0	IM7	23430.	13.0	IM7	1252.K	173.	1620.M	10.900B	1.323B	3.048B	15.270B	88590.	1149.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	IM7	1253.K	173.	1631.M	10.990B	1.324B	3.050B	15.370B	88370.	1160.
1.600	1000.	15.0	LOX	PRES	14.0	IM7	24880.	13.0	IM7	1257.K	173.	1644.M	11.090B	1.327B	3.056B	15.480B	87820.	1175.
1.700	1000.	15.0	LOX	PRES	14.0	IM7	25510.	13.0	IM7	1258.K	173.	1654.M	11.170B	1.328B	3.057B	15.550B	87730.	1182.
1.300	1000.	15.0	LOX	PRES	14.0	ALLI	122100.	13.0	IM7	1355.K	177.	4886.M	31.540B	1.409B	3.174B	36.120B	63170.	3812.
1.400	1000.	15.0	LOX	PRES	14.0	ALLI	125300.	13.0	IM7	1353.K	176.	4963.M	32.010B	1.407B	3.172B	36.590B	63070.	3868.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	IM7	1357.K	176.	5056.M	32.580B	1.411B	3.178B	37.170B	62230.	3982.
1.600	1000.	15.0	LOX	PRES	14.0	ALLI	132800.	13.0	IM7	1364.K	176.	5162.M	33.220B	1.417B	3.187B	37.820B	61010.	4133.
1.700	1000.	15.0	LOX	PRES	14.0	ALLI	136000.	13.0	IM7	1368.K	176.	5248.M	33.740B	1.420B	3.192B	38.350B	60350.	4236.
1.300	1000.	15.0	LOX	PRES	14.0	AL	159500.	13.0	IM7	1393.K	178.	2931.M	19.080B	1.441B	3.220B	23.740B	54440.	2907.
1.400	1000.	15.0	LOX	PRES	14.0	AL	163800.	13.0	IM7	1392.K	177.	2961.M	19.280B	1.440B	3.219B	23.930B	54130.	2947.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	IM7	1397.K	177.	3005.M	19.550B	1.444B	3.225B	24.220B	53080.	3042.
1.600	1000.	15.0	LOX	PRES	14.0	AL	173500.	13.0	IM7	1405.K	177.	3054.M	19.870B	1.451B	3.236B	24.550B	51650.	3169.
1.700	1000.	15.0	LOX	PRES	14.0	AL	177700.	13.0	IM7	1410.K	177.	3094.M	20.120B	1.455B	3.242B	24.810B	50800.	3256.
1.300	1000.	15.0	LOX	PRES	14.0	IM7	22810.	13.0	D6AC	1310.K	174.	1668.M	11.190B	1.371B	3.119B	15.680B	74180.	1409.
1.400	1000.	15.0	LOX	PRES	14.0	IM7	23430.	13.0	D6AC	1303.K	173.	1668.M	11.200B	1.365B	3.111B	15.680B	75320.	1388.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	D6AC	1302.K	173.	1678.M	11.280B	1.365B	3.111B	15.760B	75590.	1390.
1.600	1000.	15.0	LOX	PRES	14.0	IM7	24880.	13.0	D6AC	1305.K	173.	1690.M	11.370B	1.367B	3.115B	15.850B	75490.	1400.
1.700	1000.	15.0	LOX	PRES	14.0	IM7	25510.	13.0	D6AC	1304.K	173.	1698.M	11.440B	1.366B	3.115B	15.920B	75810.	1400.
1.300	1000.	15.0	LOX	PRES	14.0	ALLI	122100.	13.0	D6AC	1409.K	177.	4945.M	31.860B	1.454B	3.238B	36.550B	50900.	4787.
1.400	1000.	15.0	LOX	PRES	14.0	ALLI	125300.	13.0	D6AC	1404.K	176.	5019.M	32.320B	1.450B	3.234B	37.000B	51350.	4804.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	D6AC	1406.K	176.	5111.M	32.870B	1.452B	3.237B	37.560B	50980.	4912.
1.600	1000.	15.0	LOX	PRES	14.0	ALLI	132800.	13.0	D6AC	1412.K	177.	5215.M	33.500B	1.457B	3.244B	38.200B	50200.	5073.
1.700	1000.	15.0	LOX	PRES	14.0	ALLI	136000.	13.0	D6AC	1414.K	177.	5299.M	34.010B	1.458B	3.247B	38.720B	49920.	5171.
1.300	1000.	15.0	LOX	PRES	14.0	AL	159500.	13.0	D6AC	1447.K	178.	2985.M	19.390B	1.486B	3.283B	24.160B	42640.	3777.
1.400	1000.	15.0	LOX	PRES	14.0	AL	163800.	13.0	D6AC	1443.K	177.	3014.M	19.580B	1.483B	3.279B	24.340B	42880.	3784.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	D6AC	1446.K	177.	3055.M	19.850B	1.485B	3.284B	24.610B	42300.	3879.
1.600	1000.	15.0	LOX	PRES	14.0	AL	173500.	13.0	D6AC	1453.K	178.	3103.M	20.150B	1.491B	3.293B	24.930B	41280.	4026.
1.700	1000.	15.0	LOX	PRES	14.0	AL	177700.	13.0	D6AC	1456.K	178.	3141.M	20.390B	1.494B	3.296B	25.180B	40810.	4113.

BOEING HYBRID COMPUTER DATA

BODY DIAMETER TRADE

M.R.	PC	ER	OX	PRESS /PUMP	OXT-DIA	OXT-MAT	OXT-WT	SOL-DIA	SOL-MAT	TOT-WT	LCN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	1000.	15.0	LOX	PUMP	10.0	IM7	3716.	10.0	IM7	1216.K	265.	1117.M	7.373B	1.293B	2.989B	11.650B	101900.	762.
1.500	1000.	15.0	LOX	PUMP	12.0	IM7	2905.	12.0	IM7	1214.K	196.	1095.M	7.223B	1.291B	2.986B	11.500B	102200.	750.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	14.0	IM7	1214.K	159.	1086.M	7.156B	1.291B	2.986B	11.430B	101100.	754.
1.500	1000.	15.0	LOX	PUMP	16.0	IM7	2124.	16.0	IM7	1215.K	137.	1081.M	7.128B	1.292B	2.988B	11.410B	100200.	759.
1.500	1000.	15.0	LOX	PUMP	18.0	IM7	1881.	18.0	IM7	1217.K	124.	1081.M	7.124B	1.293B	2.990B	11.410B	98410.	773.
1.500	1000.	15.0	LOX	PUMP	10.0	ALLI	13600.	10.0	IM7	1224.K	266.	1534.M	10.310B	1.299B	2.999B	14.610B	99620.	978.
1.500	1000.	15.0	LOX	PUMP	12.0	ALLI	11510.	12.0	IM7	1221.K	197.	1463.M	9.818B	1.297B	2.995B	14.110B	100200.	939.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	14.0	IM7	1220.K	159.	1422.M	9.540B	1.296B	2.994B	13.830B	99270.	929.
1.500	1000.	15.0	LOX	PUMP	16.0	ALLI	9408.	16.0	IM7	1221.K	137.	1399.M	9.376B	1.297B	2.995B	13.670B	98430.	926.
1.500	1000.	15.0	LOX	PUMP	18.0	ALLI	8808.	18.0	IM7	1223.K	124.	1385.M	9.279B	1.298B	2.997B	13.570B	96740.	935.
1.500	1000.	15.0	LOX	PUMP	10.0	AL	17660.	10.0	IM7	1228.K	266.	1248.M	8.229B	1.303B	3.004B	12.540B	98470.	849.
1.500	1000.	15.0	LOX	PUMP	12.0	AL	14940.	12.0	IM7	1224.K	197.	1214.M	8.000B	1.300B	2.999B	12.300B	99190.	827.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	14.0	IM7	1223.K	159.	1197.M	7.886B	1.299B	2.998B	12.180B	98390.	825.
1.500	1000.	15.0	LOX	PUMP	16.0	AL	12210.	16.0	IM7	1224.K	137.	1189.M	7.832B	1.299B	2.999B	12.130B	97640.	828.
1.500	1000.	15.0	LOX	PUMP	18.0	AL	11430.	18.0	IM7	1225.K	124.	1186.M	7.815B	1.300B	3.001B	12.120B	96010.	842.
1.500	1000.	15.0	LOX	PUMP	10.0	IM7	3716.	10.0	D6AC	1269.K	265.	1165.M	7.682B	1.337B	3.055B	12.070B	87300.	922.
1.500	1000.	15.0	LOX	PUMP	12.0	IM7	2905.	12.0	D6AC	1264.K	197.	1141.M	7.516B	1.333B	3.049B	11.900B	88250.	899.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	14.0	D6AC	1264.K	160.	1130.M	7.445B	1.333B	3.049B	11.830B	87280.	904.
1.500	1000.	15.0	LOX	PUMP	16.0	IM7	2124.	16.0	D6AC	1266.K	138.	1127.M	7.424B	1.335B	3.052B	11.810B	86020.	915.
1.500	1000.	15.0	LOX	PUMP	18.0	IM7	1881.	18.0	D6AC	1271.K	124.	1129.M	7.438B	1.339B	3.058B	11.830B	83630.	943.
1.500	1000.	15.0	LOX	PUMP	10.0	ALLI	13600.	10.0	D6AC	1277.K	266.	1584.M	10.620B	1.343B	3.065B	15.030B	85150.	1177.
1.500	1000.	15.0	LOX	PUMP	12.0	ALLI	11510.	12.0	D6AC	1271.K	197.	1510.M	10.110B	1.339B	3.058B	14.510B	86350.	1120.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	14.0	D6AC	1270.K	160.	1469.M	9.831B	1.338B	3.057B	14.230B	85540.	1109.
1.500	1000.	15.0	LOX	PUMP	16.0	ALLI	9408.	16.0	D6AC	1272.K	138.	1446.M	9.674B	1.340B	3.060B	14.070B	84400.	1111.
1.500	1000.	15.0	LOX	PUMP	18.0	ALLI	8808.	18.0	D6AC	1277.K	124.	1435.M	9.594B	1.344B	3.066B	14.000B	82060.	1137.
1.500	1000.	15.0	LOX	PUMP	10.0	AL	17660.	10.0	D6AC	1281.K	266.	1297.M	8.539B	1.347B	3.070B	12.960B	84080.	1028.
1.500	1000.	15.0	LOX	PUMP	12.0	AL	14940.	12.0	D6AC	1274.K	197.	1260.M	8.293B	1.341B	3.062B	12.700B	85430.	991.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	14.0	D6AC	1273.K	160.	1242.M	8.176B	1.341B	3.061B	12.580B	84730.	990.
1.500	1000.	15.0	LOX	PUMP	16.0	AL	12210.	16.0	D6AC	1275.K	138.	1235.M	8.129B	1.342B	3.064B	12.540B	83660.	999.
1.500	1000.	15.0	LOX	PUMP	18.0	AL	11430.	18.0	D6AC	1280.K	124.	1235.M	8.129B	1.346B	3.069B	12.540B	81390.	1027.

BOEING HYBRID COMPUTER DATA

EXPANSION RATIO TRADE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	1000.	6.0	LOX	PUMP	14.0	IM7	2783.	13.0	IM7	1312.K	169.	1123.M	7.367B	1.373B	3.111B	11.850B	98580.	801.
1.500	1000.	10.0	LOX	PUMP	14.0	IM7	2564.	13.0	IM7	1248.K	167.	1098.M	7.222B	1.320B	3.030B	11.570B	101900.	757.
1.500	1000.	14.0	LOX	PUMP	14.0	IM7	2466.	13.0	IM7	1220.K	167.	1088.M	7.171B	1.296B	2.994B	11.460B	101000.	756.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B	100700.	757.
1.500	1000.	18.0	LOX	PUMP	14.0	IM7	2404.	13.0	IM7	1203.K	167.	1084.M	7.149B	1.282B	2.972B	11.400B	98440.	772.
1.500	1000.	22.0	LOX	PUMP	14.0	IM7	2361.	13.0	IM7	1191.K	168.	1082.M	7.145B	1.272B	2.957B	11.370B	95000.	798.
1.500	1000.	6.0	LOX	PUMP	14.0	ALLI	11490.	13.0	IM7	1320.K	169.	1496.M	9.991B	1.379B	3.119B	14.490B	96580.	1000.
1.500	1000.	10.0	LOX	PUMP	14.0	ALLI	10690.	13.0	IM7	1255.K	167.	1447.M	9.692B	1.325B	3.039B	14.060B	100000.	937.
1.500	1000.	14.0	LOX	PUMP	14.0	ALLI	10330.	13.0	IM7	1227.K	167.	1428.M	9.572B	1.302B	3.003B	13.880B	99150.	933.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	IM7	1220.K	167.	1423.M	9.543B	1.296B	2.994B	13.830B	98840.	933.
1.500	1000.	18.0	LOX	PUMP	14.0	ALLI	10100.	13.0	IM7	1209.K	168.	1416.M	9.506B	1.287B	2.980B	13.770B	96600.	950.
1.500	1000.	22.0	LOX	PUMP	14.0	ALLI	9944.	13.0	IM7	1198.K	169.	1410.M	9.470B	1.278B	2.965B	13.710B	93190.	981.
1.500	1000.	6.0	LOX	PUMP	14.0	AL	14920.	13.0	IM7	1323.K	169.	1246.M	8.174B	1.382B	3.124B	12.680B	95640.	884.
1.500	1000.	10.0	LOX	PUMP	14.0	AL	13880.	13.0	IM7	1258.K	167.	1213.M	7.980B	1.328B	3.043B	12.350B	99110.	831.
1.500	1000.	14.0	LOX	PUMP	14.0	AL	13410.	13.0	IM7	1230.K	167.	1200.M	7.906B	1.304B	3.006B	12.220B	98280.	829.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	IM7	1223.K	167.	1197.M	7.888B	1.298B	2.998B	12.180B	97970.	829.
1.500	1000.	18.0	LOX	PUMP	14.0	AL	13110.	13.0	IM7	1212.K	168.	1194.M	7.870B	1.290B	2.984B	12.140B	95750.	845.
1.500	1000.	22.0	LOX	PUMP	14.0	AL	12900.	13.0	IM7	1201.K	169.	1190.M	7.856B	1.280B	2.969B	12.100B	92350.	873.
1.500	1000.	6.0	LOX	PUMP	14.0	IM7	2783.	13.0	D6AC	1366.K	170.	1171.M	7.680B	1.418B	3.175B	12.270B	84410.	969.
1.500	1000.	10.0	LOX	PUMP	14.0	IM7	2564.	13.0	D6AC	1300.K	167.	1144.M	7.520B	1.363B	3.094B	11.980B	87950.	908.
1.500	1000.	14.0	LOX	PUMP	14.0	IM7	2466.	13.0	D6AC	1270.K	167.	1133.M	7.462B	1.338B	3.057B	11.860B	87230.	906.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1263.K	167.	1131.M	7.448B	1.332B	3.048B	11.830B	86920.	907.
1.500	1000.	18.0	LOX	PUMP	14.0	IM7	2404.	13.0	D6AC	1252.K	168.	1128.M	7.436B	1.323B	3.035B	11.790B	84750.	927.
1.500	1000.	22.0	LOX	PUMP	14.0	IM7	2361.	13.0	D6AC	1240.K	169.	1126.M	7.428B	1.313B	3.020B	11.760B	81400.	963.
1.500	1000.	6.0	LOX	PUMP	14.0	ALLI	11490.	13.0	D6AC	1373.K	170.	1546.M	10.310B	1.424B	3.184B	14.910B	82550.	1204.
1.500	1000.	10.0	LOX	PUMP	14.0	ALLI	10690.	13.0	D6AC	1306.K	167.	1495.M	9.992B	1.368B	3.103B	14.460B	86170.	1119.
1.500	1000.	14.0	LOX	PUMP	14.0	ALLI	10330.	13.0	D6AC	1277.K	167.	1474.M	9.865B	1.343B	3.065B	14.270B	85490.	1113.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	D6AC	1270.K	167.	1469.M	9.834B	1.338B	3.057B	14.230B	85190.	1114.
1.500	1000.	18.0	LOX	PUMP	14.0	ALLI	10100.	13.0	D6AC	1259.K	168.	1463.M	9.795B	1.328B	3.043B	14.170B	83040.	1138.
1.500	1000.	22.0	LOX	PUMP	14.0	ALLI	9944.	13.0	D6AC	1247.K	169.	1456.M	9.755B	1.318B	3.028B	14.100B	79710.	1179.
1.500	1000.	6.0	LOX	PUMP	14.0	AL	14920.	13.0	D6AC	1377.K	170.	1295.M	8.488B	1.427B	3.188B	13.100B	81690.	1069.
1.500	1000.	10.0	LOX	PUMP	14.0	AL	13880.	13.0	D6AC	1310.K	167.	1260.M	8.279B	1.371B	3.106B	12.760B	85340.	997.
1.500	1000.	14.0	LOX	PUMP	14.0	AL	13410.	13.0	D6AC	1280.K	167.	1246.M	8.198B	1.346B	3.069B	12.610B	84680.	993.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	D6AC	1273.K	167.	1242.M	8.178B	1.340B	3.060B	12.580B	84380.	994.
1.500	1000.	18.0	LOX	PUMP	14.0	AL	13110.	13.0	D6AC	1262.K	168.	1239.M	8.158B	1.331B	3.047B	12.540B	82240.	1017.
1.500	1000.	22.0	LOX	PUMP	14.0	AL	12900.	13.0	D6AC	1250.K	169.	1235.M	8.139B	1.321B	3.031B	12.490B	78920.	1055.

BOEING HYBRID COMPUTER DATA

CHAMBER PRESSURE TRADE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	600.	15.0	LOX	PUMP	14.0	IM7	2456.	13.0	IM7	1214.K	172.	1132.M	7.467B	1.292B	2.987B	11.750B	88110.	889.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B	100700.	757.
1.500	1400.	15.0	LOX	PUMP	14.0	IM7	2433.	13.0	IM7	1216.K	164.	1068.M	7.034B	1.293B	2.989B	11.320B	104400.	723.
1.500	1800.	15.0	LOX	PUMP	14.0	IM7	2427.	13.0	IM7	1221.K	162.	1062.M	6.985B	1.297B	2.995B	11.280B	105500.	713.
1.500	2200.	15.0	LOX	PUMP	14.0	IM7	2419.	13.0	IM7	1226.K	161.	1061.M	6.967B	1.301B	3.001B	11.270B	105500.	712.
1.500	600.	15.0	LOX	PUMP	14.0	ALLI	10300.	13.0	IM7	1221.K	172.	1472.M	9.863B	1.297B	2.996B	14.160B	86250.	1094.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	IM7	1220.K	167.	1423.M	9.543B	1.296B	2.994B	13.830B	98840.	933.
1.500	1400.	15.0	LOX	PUMP	14.0	ALLI	10210.	13.0	IM7	1222.K	164.	1403.M	9.411B	1.298B	2.997B	13.710B	102600.	891.
1.500	1800.	15.0	LOX	PUMP	14.0	ALLI	10190.	13.0	IM7	1227.K	162.	1396.M	9.356B	1.302B	3.003B	13.660B	103700.	878.
1.500	2200.	15.0	LOX	PUMP	14.0	ALLI	10160.	13.0	IM7	1232.K	161.	1395.M	9.333B	1.306B	3.009B	13.650B	103700.	878.
1.500	600.	15.0	LOX	PUMP	14.0	AL	13360.	13.0	IM7	1224.K	172.	1245.M	8.201B	1.300B	2.999B	12.500B	85380.	976.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	IM7	1223.K	167.	1197.M	7.888B	1.298B	2.998B	12.180B	97970.	829.
1.500	1400.	15.0	LOX	PUMP	14.0	AL	13250.	13.0	IM7	1225.K	164.	1179.M	7.762B	1.301B	3.001B	12.060B	101700.	791.
1.500	1800.	15.0	LOX	PUMP	14.0	AL	13220.	13.0	IM7	1230.K	163.	1173.M	7.711B	1.305B	3.007B	12.020B	102800.	780.
1.500	2200.	15.0	LOX	PUMP	14.0	AL	13190.	13.0	IM7	1235.K	162.	1171.M	7.691B	1.309B	3.013B	12.010B	102800.	779.
1.500	600.	15.0	LOX	PUMP	14.0	IM7	2456.	13.0	D6AC	1244.K	173.	1164.M	7.672B	1.316B	3.025B	12.010B	79780.	1004.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1263.K	167.	1131.M	7.448B	1.332B	3.048B	11.830B	86920.	907.
1.500	1400.	15.0	LOX	PUMP	14.0	IM7	2433.	13.0	D6AC	1286.K	164.	1125.M	7.394B	1.351B	3.077B	11.820B	85380.	923.
1.500	1800.	15.0	LOX	PUMP	14.0	IM7	2427.	13.0	D6AC	1312.K	163.	1129.M	7.406B	1.372B	3.108B	11.890B	81290.	975.
1.500	2200.	15.0	LOX	PUMP	14.0	IM7	2419.	13.0	D6AC	1338.K	162.	1138.M	7.447B	1.395B	3.141B	11.980B	76220.	1048.
1.500	600.	15.0	LOX	PUMP	14.0	ALLI	10300.	13.0	D6AC	1251.K	173.	1505.M	10.070B	1.322B	3.033B	14.420B	77990.	1233.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	D6AC	1270.K	167.	1469.M	9.834B	1.338B	3.057B	14.230B	85190.	1114.
1.500	1400.	15.0	LOX	PUMP	14.0	ALLI	10210.	13.0	D6AC	1293.K	164.	1462.M	9.773B	1.357B	3.085B	14.210B	83700.	1132.
1.500	1800.	15.0	LOX	PUMP	14.0	ALLI	10190.	13.0	D6AC	1318.K	163.	1466.M	9.781B	1.378B	3.116B	14.270B	79680.	1194.
1.500	2200.	15.0	LOX	PUMP	14.0	ALLI	10160.	13.0	D6AC	1344.K	162.	1474.M	9.816B	1.400B	3.148B	14.360B	74660.	1282.
1.500	600.	15.0	LOX	PUMP	14.0	AL	13360.	13.0	D6AC	1254.K	173.	1276.M	8.407B	1.324B	3.037B	12.770B	77160.	1103.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	D6AC	1273.K	167.	1242.M	8.178B	1.340B	3.060B	12.580B	84380.	994.
1.500	1400.	15.0	LOX	PUMP	14.0	AL	13250.	13.0	D6AC	1296.K	165.	1236.M	8.122B	1.359B	3.089B	12.570B	82920.	1011.
1.500	1800.	15.0	LOX	PUMP	14.0	AL	13220.	13.0	D6AC	1321.K	163.	1240.M	8.133B	1.380B	3.120B	12.630B	78920.	1067.
1.500	2200.	15.0	LOX	PUMP	14.0	AL	13190.	13.0	D6AC	1347.K	162.	1249.M	8.172B	1.403B	3.152B	12.730B	73930.	1148.

BOEING HYBRID COMPUTER DATA

MIXTURE RATIO TRADE - CONTINUED

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDTLE	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.300	1000.	15.0	LOX	PUMP	14.0	ALLI	9522.	13.0	D6AC	1279.K	169.	1455.M	9.714B	1.346B	3.068B	14.130B	83220.	1132.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	D6AC	1270.K	167.	1469.M	9.834B	1.338B	3.057B	14.230B	85190.	1114.
1.700	1000.	15.0	LOX	PUMP	14.0	ALLI	10970.	13.0	D6AC	1270.K	167.	1487.M	9.969B	1.338B	3.057B	14.360B	85960.	1114.
1.900	1000.	15.0	LOX	PUMP	14.0	ALLI	11830.	13.0	D6AC	1286.K	168.	1516.M	10.180B	1.351B	3.078B	14.610B	84420.	1154.
2.100	1000.	15.0	LOX	PUMP	14.0	ALLI	12690.	13.0	D6AC	1306.K	170.	1547.M	10.390B	1.368B	3.104B	14.870B	82360.	1204.
2.100	1000.	15.0	LOX	PUMP	14.0	ALLI	12690.	13.0	D6AC	1306.K	170.	1547.M	10.390B	1.368B	3.104B	14.870B	82360.	1204.
2.500	1000.	15.0	LOX	PUMP	14.0	ALLI	14010.	13.0	D6AC	1325.K	171.	1588.M	10.680B	1.384B	3.128B	15.200B	80860.	1253.
2.700	1000.	15.0	LOX	PUMP	14.0	ALLI	14780.	13.0	D6AC	1346.K	173.	1615.M	10.870B	1.402B	3.155B	15.430B	78770.	1306.
2.900	1000.	15.0	LOX	PUMP	14.0	ALLI	15170.	13.0	D6AC	1345.K	172.	1622.M	10.930B	1.400B	3.153B	15.480B	79470.	1299.
1.300	1000.	15.0	LOX	PUMP	14.0	AL	12360.	13.0	D6AC	1282.K	169.	1241.M	8.154B	1.348B	3.071B	12.570B	82470.	1016.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	D6AC	1273.K	167.	1242.M	8.178B	1.340B	3.060B	12.580B	84380.	994.
1.700	1000.	15.0	LOX	PUMP	14.0	AL	14240.	13.0	D6AC	1273.K	167.	1247.M	8.219B	1.340B	3.061B	12.620B	85100.	989.
1.900	1000.	15.0	LOX	PUMP	14.0	AL	15350.	13.0	D6AC	1290.K	168.	1260.M	8.316B	1.354B	3.083B	12.750B	83480.	1018.
2.100	1000.	15.0	LOX	PUMP	14.0	AL	16470.	13.0	D6AC	1310.K	170.	1276.M	8.425B	1.371B	3.109B	12.900B	81370.	1057.
2.100	1000.	15.0	LOX	PUMP	14.0	AL	16470.	13.0	D6AC	1310.K	170.	1276.M	8.425B	1.371B	3.109B	12.900B	81370.	1057.
2.500	1000.	15.0	LOX	PUMP	14.0	AL	18180.	13.0	D6AC	1329.K	171.	1293.M	8.551B	1.387B	3.133B	13.070B	79770.	1092.
2.700	1000.	15.0	LOX	PUMP	14.0	AL	19180.	13.0	D6AC	1351.K	173.	1306.M	8.645B	1.405B	3.160B	13.210B	77630.	1134.
2.900	1000.	15.0	LOX	PUMP	14.0	AL	19700.	13.0	D6AC	1349.K	173.	1306.M	8.655B	1.404B	3.159B	13.220B	78300.	1126.

BOEING HYBRID COMPUTER DATA

MIXTURE RATIO TRADE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DUTLE	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.300	1000.	15.0	LOX	PUMP	14.0	IM7	2248.	13.0	IM7	1219.K	168.	1088.M	7.154B	1.295B	2.992B	11.440B	99660.	765.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B	100700.	757.
1.700	1000.	15.0	LOX	PUMP	14.0	IM7	2641.	13.0	IM7	1216.K	166.	1086.M	7.174B	1.293B	2.990B	11.460B	100700.	759.
1.900	1000.	15.0	LOX	PUMP	14.0	IM7	2875.	13.0	IM7	1234.K	168.	1094.M	7.233B	1.308B	3.014B	11.550B	98610.	781.
2.100	1000.	15.0	LOX	PUMP	14.0	IM7	3114.	13.0	IM7	1256.K	169.	1103.M	7.299B	1.326B	3.041B	11.670B	96090.	810.
2.300	1000.	15.0	LOX	PUMP	14.0	IM7	3349.	13.0	IM7	1278.K	171.	1111.M	7.363B	1.344B	3.070B	11.780B	93530.	840.
2.500	1000.	15.0	LOX	PUMP	14.0	IM7	3484.	13.0	IM7	1278.K	171.	1111.M	7.373B	1.344B	3.070B	11.790B	93590.	840.
2.700	1000.	15.0	LOX	PUMP	14.0	IM7	3703.	13.0	IM7	1299.K	172.	1118.M	7.425B	1.362B	3.097B	11.880B	91350.	867.
2.900	1000.	15.0	LOX	PUMP	14.0	IM7	3816.	13.0	IM7	1299.K	172.	1116.M	7.422B	1.362B	3.097B	11.880B	91690.	864.
1.300	1000.	15.0	LOX	PUMP	14.0	ALLI	9522.	13.0	IM7	1225.K	168.	1405.M	9.397B	1.300B	3.000B	13.700B	97940.	911.
1.500	1000.	15.0	LOX	PUMP	14.0	ALLI	10250.	13.0	IM7	1220.K	167.	1423.M	9.543B	1.296B	2.994B	13.830B	98840.	933.
1.700	1000.	15.0	LOX	PUMP	14.0	ALLI	10970.	13.0	IM7	1223.K	166.	1443.M	9.698B	1.299B	2.999B	14.000B	98750.	945.
1.900	1000.	15.0	LOX	PUMP	14.0	ALLI	11830.	13.0	IM7	1242.K	168.	1474.M	9.919B	1.314B	3.023B	14.260B	96490.	985.
2.100	1000.	15.0	LOX	PUMP	14.0	ALLI	12690.	13.0	IM7	1264.K	169.	1507.M	10.150B	1.333B	3.051B	14.530B	93830.	1032.
2.300	1000.	15.0	LOX	PUMP	14.0	ALLI	13530.	13.0	IM7	1286.K	171.	1539.M	10.370B	1.351B	3.080B	14.800B	91140.	1083.
2.500	1000.	15.0	LOX	PUMP	14.0	ALLI	14010.	13.0	IM7	1287.K	171.	1551.M	10.460B	1.352B	3.081B	14.900B	91120.	1090.
2.700	1000.	15.0	LOX	PUMP	14.0	ALLI	14780.	13.0	IM7	1309.K	172.	1579.M	10.660B	1.370B	3.109B	15.140B	88770.	1137.
2.900	1000.	15.0	LOX	PUMP	14.0	ALLI	15170.	13.0	IM7	1309.K	172.	1588.M	10.730B	1.370B	3.109B	15.210B	89040.	1139.
1.300	1000.	15.0	LOX	PUMP	14.0	AL	12360.	13.0	IM7	1228.K	168.	1192.M	7.838B	1.303B	3.004B	12.110B	97130.	833.
1.500	1000.	15.0	LOX	PUMP	14.0	AL	13300.	13.0	IM7	1223.K	167.	1197.M	7.888B	1.298B	2.998B	12.180B	97970.	829.
1.700	1000.	15.0	LOX	PUMP	14.0	AL	14240.	13.0	IM7	1226.K	166.	1204.M	7.949B	1.301B	3.003B	12.250B	97820.	835.
1.900	1000.	15.0	LOX	PUMP	14.0	AL	15350.	13.0	IM7	1245.K	168.	1220.M	8.060B	1.317B	3.027B	12.400B	95500.	866.
2.100	1000.	15.0	LOX	PUMP	14.0	AL	16470.	13.0	IM7	1267.K	169.	1237.M	8.180B	1.336B	3.056B	12.570B	92780.	903.
2.300	1000.	15.0	LOX	PUMP	14.0	AL	17560.	13.0	IM7	1290.K	171.	1253.M	8.294B	1.355B	3.085B	12.730B	90040.	943.
2.500	1000.	15.0	LOX	PUMP	14.0	AL	18180.	13.0	IM7	1291.K	171.	1258.M	8.332B	1.355B	3.086B	12.770B	89970.	946.
2.700	1000.	15.0	LOX	PUMP	14.0	AL	19180.	13.0	IM7	1313.K	172.	1272.M	8.430B	1.374B	3.114B	12.920B	87570.	984.
2.900	1000.	15.0	LOX	PUMP	14.0	AL	19700.	13.0	IM7	1313.K	172.	1273.M	8.451B	1.374B	3.115B	12.940B	87810.	982.
1.300	1000.	15.0	LOX	PUMP	14.0	IM7	2248.	13.0	D6AC	1273.K	168.	1137.M	7.469B	1.341B	3.060B	11.870B	84810.	933.
1.500	1000.	15.0	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1263.K	167.	1131.M	7.448B	1.332B	3.048B	11.830B	86920.	907.
1.700	1000.	15.0	LOX	PUMP	14.0	IM7	2641.	13.0	D6AC	1263.K	167.	1128.M	7.443B	1.332B	3.049B	11.820B	87820.	897.
1.900	1000.	15.0	LOX	PUMP	14.0	IM7	2875.	13.0	D6AC	1279.K	168.	1134.M	7.488B	1.345B	3.069B	11.900B	86410.	918.
2.100	1000.	15.0	LOX	PUMP	14.0	IM7	3114.	13.0	D6AC	1298.K	170.	1141.M	7.544B	1.361B	3.094B	12.000B	84480.	947.
2.100	1000.	15.0	LOX	PUMP	14.0	IM7	3114.	13.0	D6AC	1298.K	170.	1141.M	7.544B	1.361B	3.094B	12.000B	84480.	947.
2.500	1000.	15.0	LOX	PUMP	14.0	IM7	3484.	13.0	D6AC	1316.K	171.	1146.M	7.591B	1.376B	3.117B	12.090B	83200.	969.
2.700	1000.	15.0	LOX	PUMP	14.0	IM7	3703.	13.0	D6AC	1337.K	173.	1152.M	7.639B	1.394B	3.143B	12.180B	81210.	1000.
2.900	1000.	15.0	LOX	PUMP	14.0	IM7	3816.	13.0	D6AC	1335.K	172.	1149.M	7.625B	1.392B	3.141B	12.160B	81990.	989.

BOEING HYBRID COMPUTER DATA

EXPANSION RATIO TRADE ALL PRESSURIZED

M.R.	PC	ER	OX	PRESS /PUMP	OXT-DIA	OXT-MAT	OXT-WT	SOL-DIA	SOL-MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	1000.	6.0	LOX	PRES	14.0	IM7	26310.	13.0	IM7	1355.K	176.	1705.M	11.450B	1.409B	3.178B	16.040B	90030.	1188.
1.500	1000.	10.0	LOX	PRES	14.0	IM7	24970.	13.0	IM7	1289.K	173.	1657.M	11.150B	1.354B	3.096B	15.600B	89350.	1164.
1.500	1000.	14.0	LOX	PRES	14.0	IM7	24240.	13.0	IM7	1260.K	173.	1635.M	11.020B	1.329B	3.058B	15.400B	88670.	1158. (1)
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	IM7	1253.K	173.	1631.M	10.990B	1.324B	3.050B	15.370B	88370.	1160.
1.500	1000.	18.0	LOX	PRES	14.0	IM7	23870.	13.0	IM7	1242.K	173.	1625.M	10.950B	1.314B	3.036B	15.300B	86210.	1183.
1.500	1000.	22.0	LOX	PRES	14.0	IM7	23610.	13.0	IM7	1230.K	174.	1619.M	10.920B	1.305B	3.021B	15.250B	82890.	1227.
1.500	1000.	6.0	LOX	PRES	14.0	ALLI	140000.	13.0	IM7	1468.K	180.	5398.M	34.570B	1.504B	3.312B	39.380B	58770.	4467.
1.500	1000.	10.0	LOX	PRES	14.0	ALLI	133200.	13.0	IM7	1397.K	176.	5187.M	33.340B	1.444B	3.226B	38.010B	62790.	4036.
1.500	1000.	14.0	LOX	PRES	14.0	ALLI	129400.	13.0	IM7	1364.K	176.	5075.M	32.690B	1.417B	3.187B	37.290B	62480.	3979. (5)
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	IM7	1357.K	176.	5056.M	32.580B	1.411B	3.178B	37.170B	62230.	3982.
1.500	1000.	18.0	LOX	PRES	14.0	ALLI	127500.	13.0	IM7	1345.K	177.	5020.M	32.360B	1.401B	3.163B	36.930B	60210.	4089.
1.500	1000.	22.0	LOX	PRES	14.0	ALLI	126200.	13.0	IM7	1332.K	177.	4984.M	32.150B	1.390B	3.147B	36.690B	57010.	4290.
1.500	1000.	6.0	LOX	PRES	14.0	AL	183000.	13.0	IM7	1512.K	181.	3184.M	20.610B	1.541B	3.362B	25.520B	49370.	3446.
1.500	1000.	10.0	LOX	PRES	14.0	AL	174100.	13.0	IM7	1438.K	177.	3072.M	19.950B	1.479B	3.275B	24.700B	53530.	3076.
1.500	1000.	14.0	LOX	PRES	14.0	AL	169100.	13.0	IM7	1404.K	177.	3014.M	19.610B	1.450B	3.235B	24.290B	53330.	3036. (3)
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	IM7	1397.K	177.	3005.M	19.550B	1.444B	3.225B	24.220B	53080.	3042.
1.500	1000.	18.0	LOX	PRES	14.0	AL	166700.	13.0	IM7	1384.K	178.	2987.M	19.450B	1.434B	3.210B	24.090B	51090.	3143.
1.500	1000.	22.0	LOX	PRES	14.0	AL	164900.	13.0	IM7	1371.K	179.	2969.M	19.340B	1.422B	3.194B	23.960B	47890.	3335.
1.500	1000.	6.0	LOX	PRES	14.0	IM7	26310.	13.0	D6AC	1408.K	176.	1756.M	11.760B	1.453B	3.241B	16.460B	72690.	1510.
1.500	1000.	10.0	LOX	PRES	14.0	IM7	24970.	13.0	D6AC	1340.K	173.	1705.M	11.450B	1.396B	3.158B	16.000B	76400.	1396.
1.500	1000.	14.0	LOX	PRES	14.0	IM7	24240.	13.0	D6AC	1309.K	173.	1682.M	11.310B	1.371B	3.120B	15.800B	75880.	1388. (2)
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	D6AC	1302.K	173.	1678.M	11.280B	1.365B	3.111B	15.760B	75590.	1390.
1.500	1000.	18.0	LOX	PRES	14.0	IM7	23870.	13.0	D6AC	1291.K	174.	1671.M	11.240B	1.355B	3.097B	15.690B	73510.	1423.
1.500	1000.	22.0	LOX	PRES	14.0	IM7	23610.	13.0	D6AC	1279.K	175.	1665.M	11.200B	1.345B	3.081B	15.630B	70230.	1484.
1.500	1000.	6.0	LOX	PRES	14.0	ALLI	140000.	13.0	D6AC	1521.K	180.	5456.M	34.890B	1.549B	3.373B	39.810B	47370.	5603.
1.500	1000.	10.0	LOX	PRES	14.0	ALLI	133200.	13.0	D6AC	1448.K	177.	5243.M	33.650B	1.487B	3.286B	38.420B	51460.	4977.
1.500	1000.	14.0	LOX	PRES	14.0	ALLI	129400.	13.0	D6AC	1414.K	176.	5129.M	32.980B	1.458B	3.246B	37.690B	51250.	4903. (6)
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	D6AC	1406.K	176.	5111.M	32.870B	1.452B	3.237B	37.560B	50980.	4912.
1.500	1000.	18.0	LOX	PRES	14.0	ALLI	127500.	13.0	D6AC	1394.K	177.	5073.M	32.660B	1.441B	3.222B	37.320B	48990.	5079.
1.500	1000.	22.0	LOX	PRES	14.0	ALLI	126200.	13.0	D6AC	1381.K	178.	5036.M	32.440B	1.430B	3.206B	37.080B	45760.	5402.
1.500	1000.	6.0	LOX	PRES	14.0	AL	183000.	13.0	D6AC	1565.K	181.	3238.M	20.930B	1.585B	3.422B	25.940B	38480.	4494.
1.500	1000.	10.0	LOX	PRES	14.0	AL	174100.	13.0	D6AC	1489.K	178.	3124.M	20.250B	1.521B	3.334B	25.110B	42700.	3920.
1.500	1000.	14.0	LOX	PRES	14.0	AL	169100.	13.0	D6AC	1454.K	177.	3065.M	19.900B	1.492B	3.293B	24.690B	42560.	3867. (4)
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	D6AC	1446.K	177.	3055.M	19.850B	1.485B	3.284B	24.610B	42300.	3879.
1.500	1000.	18.0	LOX	PRES	14.0	AL	166700.	13.0	D6AC	1433.K	178.	3037.M	19.730B	1.475B	3.268B	24.480B	40310.	4049.
1.500	1000.	22.0	LOX	PRES	14.0	AL	164900.	13.0	D6AC	1420.K	179.	3019.M	19.630B	1.463B	3.252B	24.340B	37050.	4380.

BOEING HYBRID COMPUTER DATA

CHAMBER PRESSURE TRADE ALL PRESSURIZED

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	600.	15.0	LOX	PRES	14.0	IM7	16310.	13.0	IM7	1241.K	175.	1495.M	10.070B	1.313B	3.029B	14.410B	80020.	1201.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	IM7	1253.K	173.	1631.M	10.990B	1.324B	3.050B	15.370B	88370.	1160.
1.500	1400.	15.0	LOX	PRES	14.0	IM7	32750.	13.0	IM7	1270.K	173.	1802.M	12.120B	1.338B	3.078B	16.540B	87640.	1258.
1.500	1800.	15.0	LOX	PRES	14.0	IM7	42640.	13.0	IM7	1292.K	176.	1994.M	13.380B	1.356B	3.112B	17.850B	83940.	1418.
1.500	2200.	15.0	LOX	PRES	14.0	IM7	54120.	13.0	IM7	1316.K	180.	2210.M	14.760B	1.376B	3.151B	19.290B	78760.	1633.
1.500	600.	15.0	LOX	PRES	14.0	ALLI	85760.	13.0	IM7	1309.K	177.	3874.M	25.420B	1.371B	3.114B	29.910B	61960.	3218.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	IM7	1357.K	176.	5056.M	32.580B	1.411B	3.178B	37.170B	62230.	3982.
1.500	1400.	15.0	LOX	PRES	14.0	ALLI	176800.	13.0	IM7	1414.K	178.	6360.M	40.230B	1.458B	3.252B	44.950B	53340.	5618.
1.500	1800.	15.0	LOX	PRES	14.0	ALLI	231500.	13.0	IM7	1480.K	182.	7814.M	48.560B	1.514B	3.337B	53.420B	41330.	8617.
1.500	2200.	15.0	LOX	PRES	14.0	ALLI	295300.	13.0	IM7	1557.K	188.	9471.M	57.860B	1.579B	3.434B	62.870B	27550.	15214.
1.500	600.	15.0	LOX	PRES	14.0	AL	111800.	13.0	IM7	1336.K	178.	2444.M	16.080B	1.393B	3.146B	20.620B	55430.	2480.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	IM7	1397.K	177.	3005.M	19.550B	1.444B	3.225B	24.220B	53080.	3042.
1.500	1400.	15.0	LOX	PRES	14.0	AL	231500.	13.0	IM7	1469.K	179.	3635.M	23.370B	1.505B	3.317B	28.190B	41620.	4515.
1.500	1800.	15.0	LOX	PRES	14.0	AL	303900.	13.0	IM7	1553.K	184.	4338.M	27.540B	1.576B	3.421B	32.530B	27000.	8032.
1.500	2200.	15.0	LOX	PRES	14.0	AL	388700.	13.0	IM7	1651.K	190.	5139.M	32.190B	1.659B	3.539B	37.390B	10460.	23830.
1.500	600.	15.0	LOX	PRES	14.0	IM7	16310.	13.0	D6AC	1270.K	175.	1527.M	10.270B	1.338B	3.066B	14.680B	72070.	1358.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	13.0	D6AC	1302.K	173.	1678.M	11.280B	1.365B	3.111B	15.760B	75590.	1390.
1.500	1400.	15.0	LOX	PRES	14.0	IM7	32750.	13.0	D6AC	1339.K	174.	1861.M	12.480B	1.396B	3.163B	17.040B	70440.	1613.
1.500	1800.	15.0	LOX	PRES	14.0	IM7	42640.	13.0	D6AC	1380.K	176.	2066.M	13.800B	1.430B	3.220B	18.450B	62700.	1962.
1.500	2200.	15.0	LOX	PRES	14.0	IM7	54120.	13.0	D6AC	1425.K	180.	2292.M	15.240B	1.468B	3.282B	19.990B	53830.	2476.
1.500	600.	15.0	LOX	PRES	14.0	ALLI	85760.	13.0	D6AC	1339.K	178.	3910.M	25.630B	1.395B	3.150B	30.180B	54630.	3683.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	13.0	D6AC	1406.K	176.	5111.M	32.870B	1.452B	3.237B	37.560B	50980.	4912.
1.500	1400.	15.0	LOX	PRES	14.0	ALLI	176800.	13.0	D6AC	1483.K	178.	6430.M	40.600B	1.516B	3.333B	45.450B	38790.	7811.
1.500	1800.	15.0	LOX	PRES	14.0	ALLI	231500.	13.0	D6AC	1569.K	183.	7898.M	49.000B	1.589B	3.439B	54.020B	23930.	15049.
1.500	2200.	15.0	LOX	PRES	14.0	ALLI	295300.	13.0	D6AC	1667.K	188.	9570.M	58.350B	1.672B	3.556B	63.580B	7604.	55743.
1.500	600.	15.0	LOX	PRES	14.0	AL	111800.	13.0	D6AC	1365.K	178.	2479.M	16.290B	1.417B	3.182B	20.890B	48300.	2883.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	13.0	D6AC	1446.K	177.	3055.M	19.850B	1.485B	3.284B	24.610B	42300.	3879.
1.500	1400.	15.0	LOX	PRES	14.0	AL	231500.	13.0	D6AC	1538.K	180.	3700.M	23.730B	1.563B	3.396B	28.690B	27780.	6885.
1.500	1800.	15.0	LOX	PRES	14.0	AL	303900.	13.0	D6AC	1642.K	184.	4417.M	27.960B	1.651B	3.520B	33.130B	10520.	20995.
1.500	2200.	15.0	LOX	PRES	14.0	AL	388700.	13.0	D6AC	1761.K	191.	5231.M	32.680B	1.753B	3.657B	38.090B	10.	******

BOEING HYBRID COMPUTER DATA

BODY DIAMETER TRADE ALL PRESSURIZED

M.R.	PC	ER	OX	PRESS /PUMP	OXT-DIA	OXT-MAT	OXT-WT	SOL-DIA	SOL-MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	PAY	$/PL
1.500	1000.	15.0	LOX	PRES	10.0	IM7	26140.	10.0	IM7	1252.K	282.	1662.M	11.190B	1.322B	3.048B	15.560B	90830.	1142.
1.500	1000.	15.0	LOX	PRES	12.0	IM7	24770.	12.0	IM7	1248.K	207.	1621.M	10.920B	1.319B	3.043B	15.280B	91570.	1112.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	14.0	IM7	1247.K	165.	1602.M	10.800B	1.319B	3.042B	15.160B	90310.	1119.
1.500	1000.	15.0	LOX	PRES	16.0	IM7	23440.	16.0	IM7	1247.K	140.	1589.M	10.710B	1.319B	3.043B	15.070B	90110.	1115.
1.500	1000.	15.0	LOX	PRES	18.0	IM7	22600.	18.0	IM7	1247.K	125.	1559.M	10.510B	1.318B	3.042B	14.870B	89080.	1113.
1.500	1000.	15.0	LOX	PRES	10.0	ALLI	134600.	10.0	IM7	1358.K	289.	5193.M	33.390B	1.411B	3.179B	37.980B	63790.	3969.
1.500	1000.	15.0	LOX	PRES	12.0	ALLI	131100.	12.0	IM7	1352.K	211.	5089.M	32.770B	1.407B	3.172B	37.350B	64840.	3840.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	14.0	IM7	1350.K	169.	5020.M	32.360B	1.405B	3.169B	36.930B	64040.	3844.
1.500	1000.	15.0	LOX	PRES	16.0	ALLI	127000.	16.0	IM7	1350.K	143.	4975.M	32.090B	1.404B	3.169B	36.660B	64220.	3806.
1.500	1000.	15.0	LOX	PRES	18.0	ALLI	125300.	18.0	IM7	1348.K	127.	4911.M	31.720B	1.403B	3.167B	36.290B	63550.	3807.
1.500	1000.	15.0	LOX	PRES	10.0	AL	175900.	10.0	IM7	1399.K	291.	3061.M	19.890B	1.446B	3.228B	24.560B	54190.	3021.
1.500	1000.	15.0	LOX	PRES	12.0	AL	171400.	12.0	IM7	1393.K	213.	3006.M	19.550B	1.441B	3.221B	24.210B	55460.	2910.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	14.0	IM7	1390.K	170.	2970.M	19.330B	1.438B	3.217B	23.980B	54850.	2915.
1.500	1000.	15.0	LOX	PRES	16.0	AL	165900.	16.0	IM7	1389.K	144.	2952.M	19.220B	1.437B	3.216B	23.870B	55190.	2883.
1.500	1000.	15.0	LOX	PRES	18.0	AL	163800.	18.0	IM7	1387.K	128.	2913.M	18.980B	1.435B	3.213B	23.630B	54670.	2882.
1.500	1000.	15.0	LOX	PRES	10.0	IM7	26140.	10.0	D6AC	1304.K	282.	1712.M	11.500B	1.366B	3.113B	15.980B	77160.	1381.
1.500	1000.	15.0	LOX	PRES	12.0	IM7	24770.	12.0	D6AC	1297.K	207.	1668.M	11.210B	1.360B	3.105B	15.680B	78500.	1332.
1.500	1000.	15.0	LOX	PRES	14.0	IM7	24120.	14.0	D6AC	1296.K	166.	1649.M	11.090B	1.360B	3.104B	15.550B	77360.	1340.
1.500	1000.	15.0	LOX	PRES	16.0	IM7	23440.	16.0	D6AC	1298.K	141.	1637.M	11.000B	1.361B	3.106B	15.470B	76820.	1343.
1.500	1000.	15.0	LOX	PRES	18.0	IM7	22600.	18.0	D6AC	1301.K	125.	1610.M	10.820B	1.363B	3.109B	15.300B	75110.	1358.
1.500	1000.	15.0	LOX	PRES	10.0	ALLI	134600.	10.0	D6AC	1410.K	289.	5251.M	33.700B	1.455B	3.241B	38.400B	51790.	4943.
1.500	1000.	15.0	LOX	PRES	12.0	ALLI	131100.	12.0	D6AC	1402.K	212.	5143.M	33.070B	1.448B	3.232B	37.750B	53360.	4716.
1.500	1000.	15.0	LOX	PRES	14.0	ALLI	128800.	14.0	D6AC	1399.K	169.	5074.M	32.650B	1.446B	3.229B	37.330B	52650.	4727.
1.500	1000.	15.0	LOX	PRES	16.0	ALLI	127000.	16.0	D6AC	1401.K	144.	5030.M	32.390B	1.447B	3.230B	37.070B	52500.	4707.
1.500	1000.	15.0	LOX	PRES	18.0	ALLI	125300.	18.0	D6AC	1402.K	128.	4970.M	32.040B	1.448B	3.232B	36.720B	51210.	4780.
1.500	1000.	15.0	LOX	PRES	10.0	AL	175900.	10.0	D6AC	1451.K	291.	3115.M	20.200B	1.490B	3.290B	24.980B	42700.	3900.
1.500	1000.	15.0	LOX	PRES	12.0	AL	171400.	12.0	D6AC	1442.K	213.	3057.M	19.850B	1.482B	3.279B	24.610B	44460.	3690.
1.500	1000.	15.0	LOX	PRES	14.0	AL	168300.	14.0	D6AC	1439.K	170.	3020.M	19.620B	1.479B	3.275B	24.380B	43940.	3699.
1.500	1000.	15.0	LOX	PRES	16.0	AL	165900.	16.0	D6AC	1440.K	144.	3003.M	19.520B	1.480B	3.276B	24.280B	43940.	3684.
1.500	1000.	15.0	LOX	PRES	18.0	AL	163800.	18.0	D6AC	1441.K	128.	2968.M	19.300B	1.481B	3.277B	24.060B	42830.	3745.

BOEING HYBRID COMPUTER DATA

MIXTURE RATIO TRADE QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	
1.300	1000.	15.0	LOX	PUMP	8.4	IM7	492.	8.4	IM7	303.K	109.	1235.M	8.938B	1.291B	2.985B	13.210B	
1.400	1000.	15.0	LOX	PUMP	8.4	IM7	511.	8.4	IM7	302.K	108.	1233.M	8.932B	1.286B	2.978B	13.200B	(1)
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	IM7	302.K	108.	1235.M	8.945B	1.286B	2.978B	13.210B	
1.600	1000.	15.0	LOX	PUMP	8.4	IM7	553.	8.4	IM7	303.K	108.	1236.M	8.957B	1.288B	2.982B	13.230B	
1.700	1000.	15.0	LOX	PUMP	8.4	IM7	571.	8.4	IM7	303.K	108.	1236.M	8.960B	1.288B	2.982B	13.230B	
1.300	1000.	15.0	LOX	PUMP	8.4	ALLI	2155.	8.4	IM7	305.K	109.	1513.M	10.890B	1.295B	2.991B	15.180B	
1.400	1000.	15.0	LOX	PUMP	8.4	ALLI	2228.	8.4	IM7	303.K	108.	1519.M	10.940B	1.290B	2.984B	15.220B	
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	IM7	303.K	108.	1530.M	11.020B	1.290B	2.985B	15.290B	
1.600	1000.	15.0	LOX	PUMP	8.4	ALLI	2393.	8.4	IM7	304.K	108.	1540.M	11.100B	1.293B	2.989B	15.380B	
1.700	1000.	15.0	LOX	PUMP	8.4	ALLI	2463.	8.4	IM7	304.K	108.	1548.M	11.150B	1.292B	2.989B	15.430B	
1.300	1000.	15.0	LOX	PUMP	8.4	AL	6467.	8.4	IM7	309.K	109.	1487.M	10.670B	1.309B	3.013B	14.990B	
1.400	1000.	15.0	LOX	PUMP	8.4	AL	6689.	8.4	IM7	308.K	109.	1493.M	10.720B	1.305B	3.007B	15.030B	
1.500	1000.	15.0	LOX	PUMP	8.4	AL	6933.*	8.4	IM7	308.K	109.	1503.M	10.790B	1.305B	3.008B	15.100B	
1.600	1000.	15.0	LOX	PUMP	8.4	AL	7184.	8.4	IM7	309.K	109.	1513.M	10.860B	1.308B	3.013B	15.180B	
1.700	1000.	15.0	LOX	PUMP	8.4	AL	7397.	8.4	IM7	309.K	109.	1520.M	10.910B	1.309B	3.014B	15.230B	
1.300	1000.	15.0	LOX	PUMP	8.4	IM7	492.	8.4	D6AC	318.K	109.	1269.M	9.159B	1.339B	3.058B	13.560B	
1.400	1000.	15.0	LOX	PUMP	8.4	IM7	511.	8.4	D6AC	316.K	109.	1266.M	9.142B	1.332B	3.047B	13.520B	
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	D6AC	315.K	108.	1266.M	9.144B	1.330B	3.045B	13.520B	(2)
1.600	1000.	15.0	LOX	PUMP	8.4	IM7	553.	8.4	D6AC	315.K	108.	1266.M	9.148B	1.331B	3.047B	13.530B	
1.700	1000.	15.0	LOX	PUMP	8.4	IM7	571.	8.4	D6AC	315.K	108.	1264.M	9.144B	1.329B	3.045B	13.520B	
1.300	1000.	15.0	LOX	PUMP	8.4	ALLI	2155.	8.4	D6AC	319.K	109.	1548.M	11.110B	1.343B	3.063B	15.520B	
1.400	1000.	15.0	LOX	PUMP	8.4	ALLI	2228.	8.4	D6AC	317.K	109.	1553.M	11.150B	1.336B	3.053B	15.540B	
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	D6AC	316.K	108.	1562.M	11.220B	1.334B	3.051B	15.600B	
1.600	1000.	15.0	LOX	PUMP	8.4	ALLI	2393.	8.4	D6AC	317.K	109.	1571.M	11.290B	1.335B	3.054B	15.680B	
1.700	1000.	15.0	LOX	PUMP	8.4	ALLI	2463.	8.4	D6AC	316.K	108.	1578.M	11.340B	1.334B	3.052B	15.720B	
1.300	1000.	15.0	LOX	PUMP	8.4	AL	6467.	8.4	D6AC	323.K	110.	1522.M	10.890B	1.357B	3.085B	15.330B	
1.400	1000.	15.0	LOX	PUMP	8.4	AL	6689.	8.4	D6AC	321.K	109.	1527.M	10.930B	1.351B	3.076B	15.350B	
1.500	1000.	15.0	LOX	PUMP	8.4	AL	6933.*	8.4	D6AC	321.K	109.	1535.M	10.990B	1.349B	3.074B	15.410B	
1.600	1000.	15.0	LOX	PUMP	8.4	AL	7184.	8.4	D6AC	322.K	109.	1543.M	11.050B	1.351B	3.077B	15.480B	
1.700	1000.	15.0	LOX	PUMP	8.4	AL	7397.	8.4	D6AC	321.K	109.	1550.M	11.090B	1.350B	3.076B	15.520B	

* AL tank weight incorrect.

BOEING HYBRID COMPUTER DATA

EXPANSION RATIO TRADE QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	
1.500	1000.	6.0	LOX	PUMP	8.4	IM7	600.	8.4	IM7	327.K	111.	1269.M	9.157B	1.368B	3.103B	13.630B	
1.500	1000.	10.0	LOX	PUMP	8.4	IM7	556.	8.4	IM7	311.K	109.	1246.M	9.012B	1.315B	3.023B	13.350B	
1.500	1000.	14.0	LOX	PUMP	8.4	IM7	536.	8.4	IM7	304.K	108.	1237.M	8.959B	1.291B	2.987B	13.240B	
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	IM7	302.K	108.	1235.M	8.945B	1.286B	2.978B	13.210B	
1.500	1000.	18.0	LOX	PUMP	8.4	IM7	524.	8.4	IM7	299.K	108.	1232.M	8.933B	1.277B	2.965B	13.170B	(1)
1.500	1000.	22.0	LOX	PUMP	8.4	IM7	515.	8.4	IM7	297.K	109.	1230.M	8.923B	1.268B	2.950B	13.140B	
1.500	1000.	6.0	LOX	PUMP	8.4	ALLI	2573.	8.4	IM7	328.K	111.	1595.M	11.440B	1.373B	3.110B	15.920B	
1.500	1000.	10.0	LOX	PUMP	8.4	ALLI	2404.	8.4	IM7	312.K	109.	1552.M	11.160B	1.319B	3.030B	15.510B	
1.500	1000.	14.0	LOX	PUMP	8.4	ALLI	2328.	8.4	IM7	305.K	108.	1534.M	11.050B	1.296B	2.993B	15.330B	
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	IM7	303.K	108.	1530.M	11.020B	1.290B	2.985B	15.290B	
1.500	1000.	18.0	LOX	PUMP	8.4	ALLI	2279.	8.4	IM7	301.K	108.	1524.M	10.980B	1.281B	2.971B	15.230B	
1.500	1000.	22.0	LOX	PUMP	8.4	ALLI	2245.	8.4	IM7	298.K	109.	1518.M	10.950B	1.272B	2.956B	15.170B	(5)
1.500	1000.	6.0	LOX	PUMP	8.4	AL	3339.	8.4	IM7	329.K	111.	1344.M	9.665B	1.375B	3.114B	14.150B	
1.500	1000.	10.0	LOX	PUMP	8.4	AL	3120.	8.4	IM7	313.K	109.	1315.M	9.484B	1.322B	3.033B	13.840B	
1.500	1000.	14.0	LOX	PUMP	8.4	AL	3021.	8.4	IM7	306.K	108.	1304.M	9.414B	1.298B	2.997B	13.710B	
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	IM7	304.K	108.	1301.M	9.397B	1.292B	2.988B	13.680B	
1.500	1000.	18.0	LOX	PUMP	8.4	AL	2957.	8.4	IM7	301.K	108.	1298.M	9.378B	1.283B	2.975B	13.640B	
1.500	1000.	22.0	LOX	PUMP	8.4	AL	2913.	8.4	IM7	298.K	109.	1295.M	9.362B	1.274B	2.960B	13.600B	(3)
1.500	1000.	6.0	LOX	PUMP	8.4	IM7	600.	8.4	D6AC	341.K	111.	1303.M	9.377B	1.415B	3.172B	13.960B	
1.500	1000.	10.0	LOX	PUMP	8.4	IM7	556.	8.4	D6AC	324.K	109.	1278.M	9.219B	1.360B	3.091B	13.670B	
1.500	1000.	14.0	LOX	PUMP	8.4	IM7	536.	8.4	D6AC	317.K	109.	1268.M	9.159B	1.336B	3.054B	13.550B	
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	D6AC	315.K	108.	1266.M	9.144B	1.330B	3.045B	13.520B	
1.500	1000.	18.0	LOX	PUMP	8.4	IM7	524.	8.4	D6AC	312.K	109.	1263.M	9.128B	1.321B	3.031B	13.480B	
1.500	1000.	22.0	LOX	PUMP	8.4	IM7	515.	8.4	D6AC	310.K	109.	1261.M	9.118B	1.311B	3.016B	13.450B	(2)
1.500	1000.	6.0	LOX	PUMP	8.4	ALLI	2573.	8.4	D6AC	342.K	111.	1630.M	11.660B	1.420B	3.179B	16.260B	
1.500	1000.	10.0	LOX	PUMP	8.4	ALLI	2404.	8.4	D6AC	326.K	109.	1585.M	11.370B	1.365B	3.097B	15.830B	
1.500	1000.	14.0	LOX	PUMP	8.4	ALLI	2328.	8.4	D6AC	318.K	109.	1566.M	11.250B	1.340B	3.060B	15.650B	
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	D6AC	316.K	108.	1562.M	11.220B	1.334B	3.051B	15.600B	
1.500	1000.	18.0	LOX	PUMP	8.4	ALLI	2279.	8.4	D6AC	314.K	109.	1555.M	11.180B	1.325B	3.038B	15.540B	
1.500	1000.	22.0	LOX	PUMP	8.4	ALLI	2245.	8.4	D6AC	311.K	109.	1549.M	11.140B	1.315B	3.023B	15.480B	(6)
1.500	1000.	6.0	LOX	PUMP	8.4	AL	3339.	8.4	D6AC	343.K	112.	1378.M	9.885B	1.423B	3.182B	14.490B	
1.500	1000.	10.0	LOX	PUMP	8.4	AL	3120.	8.4	D6AC	326.K	109.	1348.M	9.691B	1.367B	3.101B	14.160B	
1.500	1000.	14.0	LOX	PUMP	8.4	AL	3021.	8.4	D6AC	319.K	109.	1335.M	9.615B	1.342B	3.064B	14.020B	
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	D6AC	317.K	109.	1332.M	9.596B	1.336B	3.055B	13.990B	
1.500	1000.	18.0	LOX	PUMP	8.4	AL	2957.	8.4	D6AC	314.K	109.	1329.M	9.573B	1.327B	3.041B	13.940B	
1.500	1000.	22.0	LOX	PUMP	8.4	AL	2913.	8.4	D6AC	311.K	109.	1326.M	9.557B	1.317B	3.026B	13.900B	(7)

BOEING HYBRID COMPUTER DATA

CHAMBER PRESSURE TRADE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.500	600.	15.0	LOX	PUMP	8.4	IM7	534.	8.4	IM7	302.K	111.	1298.M	9.387B	1.287B	2.980B	13.650B
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	IM7	302.K	108.	1235.M	8.945B	1.286B	2.978B	13.210B
1.500	1400.	15.0	LOX	PUMP	8.4	IM7	529.	8.4	IM7	303.K	107.	1207.M	8.748B	1.288B	2.982B	13.020B
1.500	1800.	15.0	LOX	PUMP	8.4	IM7	528.	8.4	IM7	304.K	106.	1195.M	8.648B	1.292B	2.988B	12.930B
1.500	2200.	15.0	LOX	PUMP	8.4	IM7	527.	8.4	IM7	305.K	106.	1188.M	8.591B	1.297B	2.995B	12.880B
1.500	600.	15.0	LOX	PUMP	8.4	ALLI	2320.	8.4	IM7	303.K	111.	1596.M	11.470B	1.291B	2.986B	15.750B
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	IM7	303.K	108.	1530.M	11.020B	1.290B	2.985B	15.290B
1.500	1400.	15.0	LOX	PUMP	8.4	ALLI	2302.	8.4	IM7	304.K	107.	1501.M	10.810B	1.292B	2.988B	15.100B
1.500	1800.	15.0	LOX	PUMP	8.4	ALLI	2297.	8.4	IM7	305.K	106.	1487.M	10.710B	1.296B	2.995B	15.000B
1.500	2200.	15.0	LOX	PUMP	8.4	ALLI	2291.	8.4	IM7	306.K	106.	1479.M	10.650B	1.301B	3.001B	14.950B
1.500	600.	15.0	LOX	PUMP	8.4	AL	3010.	8.4	IM7	304.K	111.	1365.M	9.842B	1.293B	2.990B	14.120B
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	IM7	304.K	108.	1301.M	9.397B	1.292B	2.988B	13.680B
1.500	1400.	15.0	LOX	PUMP	8.4	AL	2987.	8.4	IM7	305.K	107.	1274.M	9.198B	1.295B	2.992B	13.480B
1.500	1800.	15.0	LOX	PUMP	8.4	AL	2980.	8.4	IM7	306.K	106.	1260.M	9.097B	1.299B	2.998B	13.390B
1.500	2200.	15.0	LOX	PUMP	8.4	AL	2972.	8.4	IM7	307.K	106.	1254.M	9.038B	1.303B	3.005B	13.350B
1.500	600.	15.0	LOX	PUMP	8.4	IM7	534.	8.4	D6AC	310.K	111.	1317.M	9.509B	1.313B	3.020B	13.840B
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	D6AC	315.K	108.	1266.M	9.144B	1.330B	3.045B	13.520B
1.500	1400.	15.0	LOX	PUMP	8.4	IM7	529.	8.4	D6AC	321.K	107.	1249.M	9.012B	1.350B	3.076B	13.440B
1.500	1800.	15.0	LOX	PUMP	8.4	IM7	528.	8.4	D6AC	328.K	106.	1245.M	8.971B	1.373B	3.109B	13.450B
1.500	2200.	15.0	LOX	PUMP	8.4	IM7	527.	8.4	D6AC	335.K	106.	1247.M	8.968B	1.396B	3.143B	13.510B
1.500	600.	15.0	LOX	PUMP	8.4	ALLI	2320.	8.4	D6AC	311.K	111.	1616.M	11.590B	1.317B	3.026B	15.940B
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	D6AC	316.K	108.	1562.M	11.220B	1.334B	3.051B	15.600B
1.500	1400.	15.0	LOX	PUMP	8.4	ALLI	2302.	8.4	D6AC	323.K	107.	1543.M	11.080B	1.354B	3.082B	15.520B
1.500	1800.	15.0	LOX	PUMP	8.4	ALLI	2297.	8.4	D6AC	329.K	106.	1539.M	11.030B	1.377B	3.115B	15.520B
1.500	2200.	15.0	LOX	PUMP	8.4	ALLI	2291.	8.4	D6AC	336.K	106.	1541.M	11.030B	1.400B	3.149B	15.580B
1.500	600.	15.0	LOX	PUMP	8.4	AL	3010.	8.4	D6AC	312.K	111.	1384.M	9.964B	1.319B	3.029B	14.310B
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	D6AC	317.K	109.	1332.M	9.596B	1.336B	3.055B	13.990B
1.500	1400.	15.0	LOX	PUMP	8.4	AL	2987.	8.4	D6AC	323.K	107.	1315.M	9.462B	1.357B	3.085B	13.900B
1.500	1800.	15.0	LOX	PUMP	8.4	AL	2980.	8.4	D6AC	330.K	107.	1311.M	9.419B	1.379B	3.118B	13.920B
1.500	2200.	15.0	LOX	PUMP	8.4	AL	2972.	8.4	D6AC	337.K	106.	1314.M	9.417B	1.403B	3.152B	13.970B

BOEING HYBRID COMPUTER DATA

BODY DIAMETER TRADE QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.500	1000.	15.0	LOX	PUMP	6.0	IM7	743.	6.0	IM7	303.K	194.	1269.M	9.185B	1.289B	2.983B	13.460B
1.500	1000.	15.0	LOX	PUMP	7.2	IM7	612.	7.2	IM7	302.K	137.	1245.M	9.019B	1.286B	2.979B	13.280B
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	511.	8.4	IM7	302.K	108.	1235.M	8.945B	1.286B	2.978B	13.210B
1.500	1000.	15.0	LOX	PUMP	9.6	IM7	475.	9.6	IM7	302.K	91.	1231.M	8.919B	1.287B	2.980B	13.190B
1.500	1000.	15.0	LOX	PUMP	10.8	IM7	432.	10.8	IM7	303.K	81.	1232.M	8.923B	1.289B	2.983B	13.200B
1.500	1000.	15.0	LOX	PUMP	6.0	ALLI	2900.	6.0	IM7	304.K	194.	1615.M	11.600B	1.293B	2.990B	15.890B
1.500	1000.	15.0	LOX	PUMP	7.2	ALLI	2531.	7.2	IM7	303.K	137.	1558.M	11.210B	1.290B	2.985B	15.490B
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	IM7	303.K	108.	1530.M	11.020B	1.290B	2.985B	15.290B
1.500	1000.	15.0	LOX	PUMP	9.6	ALLI	2161.	8.4	IM7	303.K	100.	1516.M	10.920B	1.290B	2.985B	15.200B
1.500	1000.	15.0	LOX	PUMP	10.8	ALLI	2053.	10.8	IM7	304.K	81.	1509.M	10.870B	1.293B	2.989B	15.150B
1.500	1000.	15.0	LOX	PUMP	6.0	AL	3764.	6.0	IM7	305.K	194.	1338.M	9.654B	1.296B	2.994B	13.940B
1.500	1000.	15.0	LOX	PUMP	7.2	AL	3285.	7.2	IM7	304.K	137.	1311.M	9.468B	1.293B	2.989B	13.750B
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	IM7	304.K	108.	1301.M	9.397B	1.292B	2.988B	13.680B
1.500	1000.	15.0	LOX	PUMP	9.6	AL	2804.	9.6	IM7	304.K	91.	1299.M	9.379B	1.293B	2.990B	13.660B
1.500	1000.	15.0	LOX	PUMP	10.8	AL	2664.	10.8	IM7	305.K	81.	1301.M	9.397B	1.295B	2.993B	13.680B
1.500	1000.	15.0	LOX	PUMP	6.0	IM7	743.	6.0	D6AC	319.K	194.	1308.M	9.435B	1.341B	3.062B	13.840B
1.500	1000.	15.0	LOX	PUMP	7.2	IM7	612.	7.2	D6AC	316.K	137.	1278.M	9.231B	1.332B	3.048B	13.610B
1.500	1000.	15.0	LOX	PUMP	8.4	IM7	531.	8.4	D6AC	315.K	108.	1266.M	9.144B	1.330B	3.045B	13.520B
1.500	1000.	15.0	LOX	PUMP	9.6	IM7	475.	9.6	D6AC	316.K	92.	1262.M	9.116B	1.331B	3.047B	13.490B
1.500	1000.	15.0	LOX	PUMP	10.8	IM7	432.	10.8	D6AC	317.K	81.	1264.M	9.130B	1.335B	3.053B	13.520B
1.500	1000.	15.0	LOX	PUMP	6.0	ALLI	2900.	6.0	D6AC	320.K	194.	1655.M	11.860B	1.346B	3.069B	16.270B
1.500	1000.	15.0	LOX	PUMP	7.2	ALLI	2531.	7.2	D6AC	317.K	137.	1592.M	11.430B	1.336B	3.055B	15.820B
1.500	1000.	15.0	LOX	PUMP	8.4	ALLI	2309.	8.4	D6AC	316.K	108.	1562.M	11.220B	1.334B	3.051B	15.600B
1.500	1000.	15.0	LOX	PUMP	9.6	ALLI	2161.	9.6	D6AC	317.K	92.	1547.M	11.120B	1.335B	3.053B	15.500B
1.500	1000.	15.0	LOX	PUMP	10.8	ALLI	2053.	10.8	D6AC	318.K	81.	1542.M	11.080B	1.339B	3.059B	15.480B
1.500	1000.	15.0	LOX	PUMP	6.0	AL	3764.	6.0	D6AC	321.K	194.	1377.M	9.904B	1.349B	3.073B	14.330B
1.500	1000.	15.0	LOX	PUMP	7.2	AL	3285.	7.2	D6AC	318.K	138.	1345.M	9.681B	1.339B	3.059B	14.080B
1.500	1000.	15.0	LOX	PUMP	8.4	AL	2997.	8.4	D6AC	317.K	109.	1332.M	9.596B	1.336B	3.055B	13.990B
1.500	1000.	15.0	LOX	PUMP	9.6	AL	2804.	9.6	D6AC	317.K	92.	1330.M	9.578B	1.337B	3.056B	13.970B
1.500	1000.	15.0	LOX	PUMP	10.8	AL	2664.	10.8	D6AC	318.K	81.	1333.M	9.604B	1.341B	3.062B	14.010B

BOEING HYBRID COMPUTER DATA

MIXTURE RATIO TRADE ALL PRESSURIZED QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.300	1000.	15.0	LOX	PRES	8.4	IM7	6587.	8.4	IM7	321.K	120.	2283.M	16.120B	1.350B	3.107B	20.580B
1.400	1000.	15.0	LOX	PRES	8.4	IM7	6761.	8.4	IM7	320.K	119.	2301.M	16.250B	1.346B	3.103B	20.700B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	IM7	321.K	120.	2325.M	16.410B	1.348B	3.107B	20.870B
1.600	1000.	15.0	LOX	PRES	8.4	IM7	7164.	8.4	IM7	322.K	120.	2343.M	16.540B	1.352B	3.114B	21.000B
1.700	1000.	15.0	LOX	PRES	8.4	IM7	7341.	8.4	IM7	322.K	120.	2363.M	16.680B	1.353B	3.117B	21.150B
1.300	1000.	15.0	LOX	PRES	8.4	ALLI	34920.	8.4	IM7	349.K	122.	6198.M	40.310B	1.442B	3.241B	45.000B
1.400	1000.	15.0	LOX	PRES	8.4	ALLI	35820.	8.4	IM7	348.K	122.	6307.M	40.960B	1.441B	3.241B	45.640B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	IM7	350.K	122.	6426.M	41.670B	1.445B	3.248B	46.360B
1.600	1000.	15.0	LOX	PRES	8.4	ALLI	37920.	8.4	IM7	352.K	123.	6555.M	42.420B	1.453B	3.260B	47.140B
1.700	1000.	15.0	LOX	PRES	8.4	ALLI	38830.	8.4	IM7	353.K	123.	6665.M	43.080B	1.456B	3.266B	47.800B
1.300	1000.	15.0	LOX	PRES	8.4	AL	45610.	8.4	IM7	359.K	123.	3765.M	25.460B	1.478B	3.292B	30.230B
1.400	1000.	15.0	LOX	PRES	8.4	AL	46800.	8.4	IM7	359.K	123.	3818.M	25.790B	1.478B	3.293B	30.560B
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	IM7	361.K	123.	3877.M	26.160B	1.483B	3.301B	30.940B
1.600	1000.	15.0	LOX	PRES	8.4	AL	49530.	8.4	IM7	363.K	124.	3936.M	26.530B	1.492B	3.315B	31.330B
1.700	1000.	15.0	LOX	PRES	8.4	AL	50730.	8.4	IM7	365.K	124.	3990.M	26.870B	1.496B	3.322B	31.680B
1.300	1000.	15.0	LOX	PRES	8.4	IM7	6587.	8.4	D6AC	335.K	120.	2319.M	16.340B	1.397B	3.177B	20.920B
1.400	1000.	15.0	LOX	PRES	8.4	IM7	6761.	8.4	D6AC	334.K	120.	2336.M	16.460B	1.392B	3.170B	21.030B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	D6AC	334.K	120.	2358.M	16.610B	1.391B	3.171B	21.170B
1.600	1000.	15.0	LOX	PRES	8.4	IM7	7164.	8.4	D6AC	334.K	120.	2375.M	16.730B	1.394B	3.176B	21.300B
1.700	1000.	15.0	LOX	PRES	8.4	IM7	7341.	8.4	D6AC	334.K	121.	2394.M	16.860B	1.394B	3.177B	21.430B
1.300	1000.	15.0	LOX	PRES	8.4	ALLI	34920.	8.4	D6AC	363.K	122.	6239.M	40.540B	1.490B	3.309B	45.340B
1.400	1000.	15.0	LOX	PRES	8.4	ALLI	35820.	8.4	D6AC	362.K	122.	6346.M	41.170B	1.487B	3.305B	45.970B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	D6AC	363.K	123.	6464.M	41.870B	1.489B	3.310B	46.670B
1.600	1000.	15.0	LOX	PRES	8.4	ALLI	37920.	8.4	D6AC	364.K	123.	6591.M	42.620B	1.495B	3.319B	47.430B
1.700	1000.	15.0	LOX	PRES	8.4	ALLI	38830.	8.4	D6AC	365.K	123.	6700.M	43.260B	1.497B	3.324B	48.080B
1.300	1000.	15.0	LOX	PRES	8.4	AL	45610.	8.4	D6AC	374.K	123.	3804.M	25.680B	1.526B	3.358B	30.560B
1.400	1000.	15.0	LOX	PRES	8.4	AL	46800.	8.4	D6AC	373.K	123.	3855.M	26.000B	1.524B	3.356B	30.880B
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	D6AC	374.K	123.	3912.M	26.360B	1.527B	3.362B	31.250B
1.600	1000.	15.0	LOX	PRES	8.4	AL	49530.	8.4	D6AC	376.K	124.	3971.M	26.720B	1.534B	3.373B	31.630B
1.700	1000.	15.0	LOX	PRES	8.4	AL	50730.	8.4	D6AC	377.K	124.	4023.M	27.050B	1.538B	3.379B	31.970B

BOEING HYBRID COMPUTER DATA

EXPANSION RATIO TRADE ALL PRESSURIZED QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.500	1000.	6.0	LOX	PRES	8.4	IM7	7557.	8.4	IM7	347.K	124.	2423.M	17.040B	1.435B	3.238B	21.710B
1.500	1000.	10.0	LOX	PRES	8.4	IM7	7188.	8.4	IM7	330.K	121.	2356.M	16.610B	1.379B	3.154B	21.150B
1.500	1000.	14.0	LOX	PRES	8.4	IM7	6981.	8.4	IM7	322.K	120.	2330.M	16.450B	1.354B	3.116B	20.920B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	IM7	321.K	120.	2325.M	16.410B	1.348B	3.107B	20.870B
1.500	1000.	18.0	LOX	PRES	8.4	IM7	6881.	8.4	IM7	318.K	120.	2314.M	16.350B	1.338B	3.092B	20.780B
1.500	1000.	22.0	LOX	PRES	8.4	IM7	6810.	8.4	IM7	315.K	120.	2304.M	16.280B	1.328B	3.077B	20.690B
1.500	1000.	6.0	LOX	PRES	8.4	ALLI	39940.	8.4	IM7	378.K	127.	6838.M	44.050B	1.542B	3.386B	48.980B
1.500	1000.	10.0	LOX	PRES	8.4	ALLI	38040.	8.4	IM7	360.K	123.	6581.M	42.570B	1.480B	3.299B	47.350B
1.500	1000.	14.0	LOX	PRES	8.4	ALLI	36960.	8.4	IM7	352.K	123.	6448.M	41.800B	1.452B	3.257B	46.500B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	IM7	350.K	122.	6426.M	41.670B	1.445B	3.248B	46.360B
1.500	1000.	18.0	LOX	PRES	8.4	ALLI	36440.	8.4	IM7	347.K	122.	6380.M	41.400B	1.435B	3.233B	46.070B
1.500	1000.	22.0	LOX	PRES	8.4	ALLI	36080.	8.4	IM7	343.K	122.	6334.M	41.130B	1.424B	3.217B	45.770B
1.500	1000.	6.0	LOX	PRES	8.4	AL	127500.	8.4	IM7	466.K	135.	6749.M	43.180B	1.842B	3.767B	48.790B
1.500	1000.	10.0	LOX	PRES	8.4	AL	121500. *	8.4	IM7	443.K	131.	6497.M	41.730B	1.764B	3.670B	47.160B
1.500	1000.	14.0	LOX	PRES	8.4	AL	118000.	8.4	IM7	433.K	130.	6365.M	40.970B	1.727B	3.623B	46.320B
1.500	1000.	15.0	LOX	PRES	8.4	AL	117500.	8.4	IM7	430.K	130.	6344.M	40.850B	1.719B	3.613B	46.180B
1.500	1000.	18.0	LOX	PRES	8.4	AL	116400.	8.4	IM7	426.K	130.	6299.M	40.590B	1.706B	3.596B	45.890B
1.500	1000.	22.0	LOX	PRES	8.4	AL	115200.	8.4	IM7	422.K	130.	6253.M	40.330B	1.692B	3.578B	45.590B
1.500	1000.	6.0	LOX	PRES	8.4	IM7	7557.	8.4	D6AC	361.K	124.	2460.M	17.260B	1.482B	3.305B	22.040B
1.500	1000.	10.0	LOX	PRES	8.4	IM7	7188.	8.4	D6AC	343.K	121.	2391.M	16.820B	1.424B	3.219B	21.460B
1.500	1000.	14.0	LOX	PRES	8.4	IM7	6981.	8.4	D6AC	335.K	120.	2364.M	16.650B	1.397B	3.180B	21.230B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	D6AC	334.K	120.	2358.M	16.610B	1.391B	3.171B	21.170B
1.500	1000.	18.0	LOX	PRES	8.4	IM7	6881.	8.4	D6AC	331.K	120.	2347.M	16.540B	1.382B	3.156B	21.080B
1.500	1000.	22.0	LOX	PRES	8.4	IM7	6810.	8.4	D6AC	328.K	120.	2336.M	16.480B	1.371B	3.140B	20.990B
1.500	1000.	6.0	LOX	PRES	8.4	ALLI	39940.	8.4	D6AC	392.K	127.	6879.M	44.280B	1.589B	3.450B	49.320B
1.500	1000.	10.0	LOX	PRES	8.4	ALLI	38040.	8.4	D6AC	373.K	124.	6620.M	42.780B	1.525B	3.361B	47.670B
1.500	1000.	14.0	LOX	PRES	8.4	ALLI	36960.	8.4	D6AC	365.K	123.	6486.M	42.000B	1.496B	3.319B	46.810B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	D6AC	363.K	123.	6464.M	41.870B	1.489B	3.310B	46.670B
1.500	1000.	18.0	LOX	PRES	8.4	ALLI	36440.	8.4	D6AC	360.K	123.	6418.M	41.600B	1.479B	3.294B	46.380B
1.500	1000.	22.0	LOX	PRES	8.4	ALLI	36080.	8.4	D6AC	356.K	123.	6371.M	41.330B	1.467B	3.278B	46.080B
1.500	1000.	6.0	LOX	PRES	8.4	AL	127500.	8.4	D6AC	480.K	135.	6790.M	43.400B	1.890B	3.825B	49.120B
1.500	1000.	10.0	LOX	PRES	8.4	AL	121500. *	8.4	D6AC	457.K	132.	6536.M	41.940B	1.810B	3.727B	47.480B
1.500	1000.	14.0	LOX	PRES	8.4	AL	118000.	8.4	D6AC	446.K	131.	6403.M	41.170B	1.772B	3.679B	46.620B
1.500	1000.	15.0	LOX	PRES	8.4	AL	117500.	8.4	D6AC	443.K	130.	6381.M	41.050B	1.764B	3.669B	46.480B
1.500	1000.	18.0	LOX	PRES	8.4	AL	116400.	8.4	D6AC	439.K	130.	6336.M	40.790B	1.750B	3.652B	46.190B
1.500	1000.	22.0	LOX	PRES	8.4	AL	115200.	8.4	D6AC	435.K	130.	6290.M	40.520B	1.736B	3.633B	45.890B

* wrong tank weight

BOEING HYBRID COMPUTER DATA

CHAMBER PRESSURE TRADE ALL PRESSURIZED QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC	
1.500	600.	15.0	LOX	PRES	8.4	IM7	4473.	8.4	IM7	314.K	118.	2061.M	14.670B	1.327B	3.063B	19.060B	(1)
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	IM7	321.K	120.	2325.M	16.410B	1.348B	3.107B	20.870B	
1.500	1400.	15.0	LOX	PRES	8.4	IM7	9859.	8.4	IM7	328.K	124.	2619.M	18.320B	1.374B	3.159B	22.860B	
1.500	1800.	15.0	LOX	PRES	8.4	IM7	13330.	8.4	IM7	337.K	129.	2936.M	20.350B	1.404B	3.219B	24.970B	
1.500	2200.	15.0	LOX	PRES	8.4	IM7	17550.	8.4	IM7	348.K	136.	3303.M	22.660B	1.438B	3.287B	27.390B	
1.500	600.	15.0	LOX	PRES	8.4	ALLI	23380.	8.4	IM7	333.K	119.	4784.M	31.880B	1.388B	3.153B	36.420B	
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	IM7	350.K	122.	6426.M	41.670B	1.445B	3.248B	46.360B	
1.500	1400.	15.0	LOX	PRES	8.4	ALLI	52690.	8.4	IM7	370.K	128.	8296.M	52.460B	1.515B	3.359B	57.340B	
1.500	1800.	15.0	LOX	PRES	8.4	ALLI	71800.	8.4	IM7	395.K	134.	10450.M	64.580B	1.598B	3.488B	69.660B	
1.500	2200.	15.0	LOX	PRES	8.4	ALLI	95130.	8.4	IM7	424.K	143.	13010.M	78.610B	1.698B	3.636B	83.940B	
1.500	600.	15.0	LOX	PRES	8.4	AL	30470.	8.4	IM7	340.K	120.	3077.M	21.180B	1.412B	3.187B	25.780B	
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	IM7	361.K	123.	3877.M	26.160B	1.483B	3.301B	30.940B	
1.500	1400.	15.0	LOX	PRES	8.4	AL	68990.	8.4	IM7	387.K	129.	4783.M	31.660B	1.570B	3.434B	36.660B	
1.500	1800.	15.0	LOX	PRES	8.4	AL	94230.	8.4	IM7	417.K	136.	5815.M	37.780B	1.675B	3.587B	43.040B	
1.500	2200.	15.0	LOX	PRES	8.4	AL	125100.	8.4	IM7	454.K	145.	7038.M	44.880B	1.801B	3.764B	50.440B	
1.500	600.	15.0	LOX	PRES	8.4	IM7	4473.	8.4	D6AC	322.K	118.	2081.M	14.790B	1.353B	3.102B	19.240B	(2)
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	D6AC	334.K	120.	2358.M	16.610B	1.391B	3.171B	21.170B	
1.500	1400.	15.0	LOX	PRES	8.4	IM7	9859.	8.4	D6AC	347.K	124.	2664.M	18.590B	1.435B	3.248B	23.270B	
1.500	1800.	15.0	LOX	PRES	8.4	IM7	13330.	8.4	D6AC	361.K	129.	2991.M	20.670B	1.483B	3.332B	25.490B	
1.500	2200.	15.0	LOX	PRES	8.4	IM7	17550.	8.4	D6AC	377.K	136.	3368.M	23.040B	1.536B	3.423B	28.000B	
1.500	600.	15.0	LOX	PRES	8.4	ALLI	23380.	8.4	D6AC	340.K	120.	4806.M	32.010B	1.414B	3.191B	36.610B	
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	D6AC	363.K	123.	6464.M	41.870B	1.489B	3.310B	46.670B	
1.500	1400.	15.0	LOX	PRES	8.4	ALLI	52690.	8.4	D6AC	389.K	128.	8347.M	52.730B	1.577B	3.443B	57.750B	
1.500	1800.	15.0	LOX	PRES	8.4	ALLI	71800.	8.4	D6AC	419.K	135.	10520.M	64.910B	1.679B	3.592B	70.180B	
1.500	2200.	15.0	LOX	PRES	8.4	ALLI	95130.	8.4	D6AC	453.K	143.	13090.M	79.000B	1.798B	3.759B	84.550B	
1.500	600.	15.0	LOX	PRES	8.4	AL	30470.	8.4	D6AC	347.K	120.	3098.M	21.300B	1.438B	3.225B	25.960B	
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	D6AC	374.K	123.	3912.M	26.360B	1.527B	3.362B	31.250B	
1.500	1400.	15.0	LOX	PRES	8.4	AL	68990.	8.4	D6AC	405.K	129.	4831.M	31.930B	1.632B	3.516B	37.070B	
1.500	1800.	15.0	LOX	PRES	8.4	AL	94230.	8.4	D6AC	441.K	136.	5876.M	38.110B	1.755B	3.689B	43.550B	
1.500	2200.	15.0	LOX	PRES	8.4	AL	125100.	8.4	D6AC	483.K	145.	7110.M	45.260B	1.902B	3.884B	51.040B	

BOEING HYBRID COMPUTER DATA

BODY DIAMETER TRADE ALL PRESSURIZED QUARTER SIZE

M.R.	PC	ER	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.500	1000.	15.0	LOX	PRES	6.0	IM7	7298.	6.0	IM7	321.K	219.	2374.M	16.740B	1.351B	3.111B	21.200B
1.500	1000.	15.0	LOX	PRES	7.2	IM7	7096.	7.2	IM7	320.K	154.	2336.M	16.490B	1.348B	3.106B	20.950B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	IM7	321.K	120.	2325.M	16.410B	1.348B	3.107B	20.870B
1.500	1000.	15.0	LOX	PRES	9.6	IM7	6798.	9.6	IM7	321.K	99.	2307.M	16.300B	1.348B	3.107B	20.750B
1.500	1000.	15.0	LOX	PRES	10.8	IM7	6613.	10.8	IM7	321.K	86.	2277.M	16.100B	1.348B	3.108B	20.560B
1.500	1000.	15.0	LOX	PRES	6.0	ALLI	38110.	6.0	IM7	351.K	224.	6577.M	42.550B	1.451B	3.256B	47.260B
1.500	1000.	15.0	LOX	PRES	7.2	ALLI	37330.	7.2	IM7	350.K	158.	6474.M	41.950B	1.446B	3.249B	46.640B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	IM7	350.K	122.	6426.M	41.670B	1.445B	3.248B	46.360B
1.500	1000.	15.0	LOX	PRES	9.6	ALLI	36380.	9.6	IM7	350.K	101.	6384.M	41.420B	1.445B	3.248B	46.110B
1.500	1000.	15.0	LOX	PRES	10.8	ALLI	36010.	10.8	IM7	350.K	88.	6333.M	41.120B	1.445B	3.248B	45.810B
1.500	1000.	15.0	LOX	PRES	6.0	AL	49810.	6.0	IM7	363.K	226.	3948.M	26.590B	1.490B	3.311B	31.390B
1.500	1000.	15.0	LOX	PRES	7.2	AL	48770.	7.2	IM7	361.K	159.	3892.M	26.250B	1.484B	3.303B	31.040B
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	IM7	361.K	123.	3877.M	26.160B	1.483B	3.301B	30.940B
1.500	1000.	15.0	LOX	PRES	9.6	AL	47530.	9.6	IM7	361.K	102.	3860.M	26.060B	1.483B	3.301B	30.840B
1.500	1000.	15.0	LOX	PRES	10.8	AL	47050.	10.8	IM7	361.K	88.	3834.M	25.900B	1.483B	3.301B	30.680B
1.500	1000.	15.0	LOX	PRES	6.0	IM7	7298.	6.0	D6AC	337.K	219.	2416.M	16.990B	1.403B	3.187B	21.580B
1.500	1000.	15.0	LOX	PRES	7.2	IM7	7096.	7.2	D6AC	334.K	154.	2372.M	16.700B	1.393B	3.173B	21.270B
1.500	1000.	15.0	LOX	PRES	8.4	IM7	6949.	8.4	D6AC	334.K	120.	2358.M	16.610B	1.391B	3.171B	21.170B
1.500	1000.	15.0	LOX	PRES	9.6	IM7	6798.	9.6	D6AC	334.K	99.	2340.M	16.500B	1.392B	3.172B	21.060B
1.500	1000.	15.0	LOX	PRES	10.8	IM7	6613.	10.8	D6AC	334.K	87.	2311.M	16.310B	1.394B	3.174B	20.880B
1.500	1000.	15.0	LOX	PRES	6.0	ALLI	38110.	6.0	D6AC	367.K	225.	6624.M	42.800B	1.503B	3.329B	47.630B
1.500	1000.	15.0	LOX	PRES	7.2	ALLI	37330.	7.2	D6AC	363.K	158.	6513.M	42.160B	1.492B	3.313B	46.970B
1.500	1000.	15.0	LOX	PRES	8.4	ALLI	36800.	8.4	D6AC	363.K	123.	6464.M	41.870B	1.489B	3.310B	46.670B
1.500	1000.	15.0	LOX	PRES	9.6	ALLI	36380.	9.6	D6AC	363.K	102.	6421.M	41.620B	1.489B	3.310B	46.420B
1.500	1000.	15.0	LOX	PRES	10.8	ALLI	36010.	10.8	D6AC	363.K	88.	6372.M	41.330B	1.491B	3.312B	46.130B
1.500	1000.	15.0	LOX	PRES	6.0	AL	49810.	6.0	D6AC	379.K	226.	3992.M	26.840B	1.542B	3.383B	31.770B
1.500	1000.	15.0	LOX	PRES	7.2	AL	48770.	7.2	D6AC	375.K	159.	3930.M	26.460B	1.530B	3.366B	31.360B
1.500	1000.	15.0	LOX	PRES	8.4	AL	48070.	8.4	D6AC	374.K	123.	3912.M	26.360B	1.527B	3.362B	31.250B
1.500	1000.	15.0	LOX	PRES	9.6	AL	47530.	9.6	D6AC	374.K	102.	3896.M	26.260B	1.527B	3.362B	31.150B
1.500	1000.	15.0	LOX	PRES	10.8	AL	47050.	10.8	D6AC	374.K	89.	3871.M	26.110B	1.528B	3.364B	31.000B

BOEING HYBRID COMPUTER DATA

FLUID INJECTION TVC IMPACTS - OXIDIZER INJECTION

DCYC	TDA	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDTLE	ACQ	NON-OP	OPS	LCC	PAY	$/PL
20.	1.	LOX	PUMP	14.0	IM7	2466.	13.0	IM7	1212.K	167.	955.M	6.390B	1.289B	2.984B	10.660B	102000.	697.
20.	3.	LOX	PUMP	14.0	IM7	2528.	13.0	IM7	1226.K	168.	975.M	6.500B	1.301B	3.002B	10.800B	99820.	721.
20.	5.	LOX	PUMP	14.0	IM7	2633.	13.0	IM7	1248.K	170.	995.M	6.615B	1.320B	3.030B	10.960B	96780.	755.
40.	1.	LOX	PUMP	14.0	IM7	2488.	13.0	IM7	1216.K	167.	957.M	6.399B	1.293B	2.989B	10.680B	101500.	701.
40.	3.	LOX	PUMP	14.0	IM7	2613.	13.0	IM7	1242.K	170.	981.M	6.537B	1.314B	3.022B	10.870B	98020.	739.
40.	5.	LOX	PUMP	14.0	IM7	2823.	13.0	IM7	1282.K	173.	1008.M	6.696B	1.348B	3.073B	11.120B	92950.	798.
60.	1.	LOX	PUMP	14.0	IM7	2511.	13.0	IM7	1220.K	168.	958.M	6.409B	1.296B	2.995B	10.700B	101000.	706.
60.	3.	LOX	PUMP	14.0	IM7	2698.	13.0	IM7	1257.K	171.	987.M	6.573B	1.327B	3.041B	10.940B	96270.	758.
60.	5.	LOX	PUMP	14.0	IM7	3013.	13.0	IM7	1314.K	176.	1020.M	6.775B	1.375B	3.114B	11.260B	89320.	840.

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1

BOEING HYBRID COMPUTER DATA

QUARTER SIZE FLUID INJECTION TVC IMPACTS - OXIDIZER INJECTION

DCYC	TDA	OX	PRESS /PUMP	OXT-DIA	OXT-MAT	OXT-WT	SOL-DIA	SOL-MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
20.	1.	LOX	PUMP	8.4	IM7	536.	8.4	IM7	301.K	108.	975.M	7.206B	1.283B	2.974B	11.460B
20.	3.	LOX	PUMP	8.4	IM7	549.	8.4	IM7	304.K	109.	991.M	7.314B	1.294B	2.991B	11.600B
20.	5.	LOX	PUMP	8.4	IM7	570.	8.4	IM7	309.K	110.	1008.M	7.423B	1.311B	3.017B	11.750B
40.	1.	LOX	PUMP	8.4	IM7	540.	8.4	IM7	302.K	109.	976.M	7.215B	1.287B	2.980B	11.480B
40.	3.	LOX	PUMP	8.4	IM7	566.	8.4	IM7	308.K	110.	997.M	7.348B	1.307B	3.011B	11.670B
40.	5.	LOX	PUMP	8.4	IM7	608.	8.4	IM7	318.K	112.	1020.M	7.500B	1.339B	3.059B	11.900B
60.	1.	LOX	PUMP	8.4	IM7	545.	8.4	IM7	303.K	109.	977.M	7.223B	1.290B	2.985B	11.500B
60.	3.	LOX	PUMP	8.4	IM7	583.	8.4	IM7	312.K	111.	1001.M	7.379B	1.319B	3.030B	11.730B
60.	5.	LOX	PUMP	8.4	IM7	645.	8.4	IM7	326.K	115.	1031.M	7.570B	1.366B	3.100B	12.040B

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1

BOEING HYBRID COMPUTER DATA

QUARTER SIZE FLUID INJECTION TVC IMPACTS - GG INJECTION

DCYC	TDA	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
20.	1.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	301.K	108.	974.M	7.202B	1.283B	2.974B	11.460B
20.	3.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	305.K	109.	991.M	7.308B	1.295B	2.992B	11.590B
20.	5.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	310.K	111.	1008.M	7.414B	1.313B	3.020B	11.750B
40.	1.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	302.K	109.	975.M	7.209B	1.287B	2.980B	11.480B
40.	3.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	309.K	110.	995.M	7.331B	1.308B	3.012B	11.650B
40.	5.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	319.K	113.	1016.M	7.465B	1.342B	3.064B	11.870B
60.	1.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	303.K	109.	976.M	7.215B	1.290B	2.985B	11.490B
60.	3.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	312.K	111.	999.M	7.354B	1.321B	3.031B	11.710B
60.	5.	LOX	PUMP	8.4	IM7	531.	8.4	IM7	327.K	115.	1025.M	7.513B	1.370B	3.104B	11.990B

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1

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OF POOR QUALITY

BOEING HYBRID COMPUTER DATA

SAFETY FACTOR IMPACT

SF	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.60	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B
1.70	LOX	PUMP	14.0	IM7	2585.	13.0	IM7	1214.K	167.	1090.M	7.188B	1.291B	2.987B	11.470B
1.80	LOX	PUMP	14.0	IM7	2728.	13.0	IM7	1215.K	167.	1095.M	7.218B	1.292B	2.988B	11.500B
1.90	LOX	PUMP	14.0	IM7	2870.	13.0	IM7	1216.K	167.	1099.M	7.247B	1.293B	2.989B	11.530B
2.00	LOX	PUMP	14.0	IM7	3013.	13.0	IM7	1217.K	167.	1103.M	7.276B	1.294B	2.990B	11.560B

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1

BOEING HYBRID COMPUTER DATA

PROPELLANT RESERVE IMPACT

RES	OX	PRESS /PUMP	OXT-DIA	OXT-MAT	OXT-WT	SOL-DIA	SOL-MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
2.	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B
3.	LOX	PUMP	14.0	IM7	2480.	13.0	IM7	1225.K	168.	1089.M	7.179B	1.300B	3.000B	11.480B
4.	LOX	PUMP	14.0	IM7	2516.	13.0	IM7	1236.K	169.	1093.M	7.200B	1.309B	3.014B	11.520B
5.	LOX	PUMP	14.0	IM7	2553.	13.0	IM7	1247.K	170.	1096.M	7.222B	1.319B	3.029B	11.570B

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1

BOEING HYBRID COMPUTER DATA

IMPACT OF INTERNAL GRAIN RADIUS

IGR	OX	PRESS /PUMP	OXT- DIA	OXT- MAT	OXT- WT	SOL- DIA	SOL- MAT	TOT-WT	LEN	DDT&E	ACQ	NON-OP	OPS	LCC
1.5	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1212.K	162.	1082.M	7.128B	1.290B	2.984B	11.400B
1.9	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	164.	1084.M	7.141B	1.290B	2.985B	11.420B
2.3	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1213.K	167.	1086.M	7.158B	1.291B	2.986B	11.430B
2.8	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1214.K	170.	1089.M	7.184B	1.291B	2.987B	11.460B
3.3	LOX	PUMP	14.0	IM7	2443.	13.0	IM7	1216.K	175.	1094.M	7.216B	1.292B	2.988B	11.500B
1.5	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1258.K	163.	1123.M	7.393B	1.328B	3.042B	11.760B
1.9	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1260.K	165.	1126.M	7.416B	1.330B	3.045B	11.790B
2.3	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1263.K	167.	1131.M	7.448B	1.332B	3.048B	11.830B
2.8	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1267.K	171.	1137.M	7.493B	1.336B	3.054B	11.880B
3.3	LOX	PUMP	14.0	IM7	2443.	13.0	D6AC	1273.K	176.	1146.M	7.553B	1.340B	3.061B	11.950B

FOR ALL CASES: MIXTURE RATIO = 1.5
CHAMBER PRESSURE = 1000 psi
EXPANSION RATIO = 15:1