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HYBRID PROPULSION TECHNOLOGY PROGRAM
PHASE I

FINAL REPORT

VOLUME II

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George C. Marshall Space Flight Center
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Marshall Space Flight Center, Alabama 35812

Prepared by
GENERAL DYNAMICS SPACE SYSTEMS DIVISION
P.O. Box 85990
San Diego, California 92138

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PREFACE

This volume is part of a four-volume set that describes the work performed from 6 March to 30 November 1989 under contract NAS8-37777 entitled, "The Hybrid Propulsion Technology Program--Phase I." The study was directed by Mr. Ben Shackelford of the NASA/Marshall Space Flight Center. Listed below are major sections from the four volumes that comprise this Final Report.

Volume I • Executive Summary

Volume II • General Dynamics Final Report

- Concept Definition
- Technology Acquisition Plans
- Large Subscale Motor System Technology Demonstration Plan

Volume III • Thiokol Corporation Final Report

- Trade Studies and Analyses
- Technology Acquisition
- Large Subscale Motor Demonstration

Volume IV • Rockwell International Corporation Final Report

- Concept Evaluation
- Technology Identification
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LIST OF ACRONYMS AND ABBREVIATIONS

ALS	Advanced Launch System
ASRM	Advanced Solid Rocket Motor
COTR	Contracting Officer's Technical Representative
DDT&E	Design, Development, Test and Evaluation
ERB	Engineering Review Board
GD	General Dynamics
GDSS	General Dynamics Space Systems Division
GR/EP	Graphite/Epoxy
GSE	Ground Support Equipment
HP	Hybrid Propulsion
HPB	Hybrid Propellant Booster
HPT	Hybrid Propulsion Technology
HRB	Hybrid Rocket Booster
HRM	Hybrid Rocket Motor
HTPB	Hydroxy - Terminated Poly Butadiene
IRAD	Internal Research and Development
ISP	Specific Impulse
LBM	Pound Mass
LCC	Life Cycle Costs
LO2	Liquid Oxygen
LRB	Liquid Rocket Booster
LBF	Pounds Force
MSFC	Marshall Space Flight Center
N2H2	Hydrazine
NASA	National Aeronautics and Space Administration
O/F	Oxygen-to-Fuel Ratio
ODE	One Dimensional Equilibrium
RD	Rocketdyne
RP-1	Rocket Propellant
SDV	Shuttle Derived Vehicles
SOW	Statement of Work
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSME	Space Shuttle Main Engine
STS	Space Transportation System
TC	Thiokol Corporation

1.0 INTRODUCTION

We strongly endorse the program objectives of developing hybrid propulsion technology (HPT) to enable its application for manned and unmanned high-thrust, high-performance space launch vehicles. Our studies indicate that hybrid propulsion (HP) is very attractive, especially when applied to large boosters for programs such as the Advanced Launch System (ALS) and the second-generation Space Shuttle.

Figure 1-1 identifies some of the advantages of HP. Space launch vehicles using HP are less costly than those flying today because their propellant and insulation costs are substantially less and there are fewer operational restraints due to reduced safety requirements. Propellant costs for a booster using HP are about $1.05 per kilogram loaded versus about $7.72 for existing solid rocket boosters (SRBs). The cost savings of $6.67 per kilogram is significant. For example, the Space Transportation System would yield a cost savings of $9.7 million per flight in propellant and insulation costs if hybrid boosters were used instead of SRBs.

Boosters using HP have safety features that are highly desirable, particularly for manned flights. They can terminate thrust at any time by closing liquid propellant valves. Their LO2 tanks are empty until just prior to launch, which provides a safe environment for ground operations. During flight, accidental burning or detonation of the propellants is impossible, because the fuel is contained in a separate pressure chamber.

HP systems will have a clean exhaust and high performance. The fuel grain can incorporate inert, clean exhausting ingredients which, when combined with an energetic oxidizer such as LO2, will provide specific impulses approaching those of LO2/hydrocarbon rocket engines.

Boosters using HP readily integrate with launch vehicles and their launch operations,

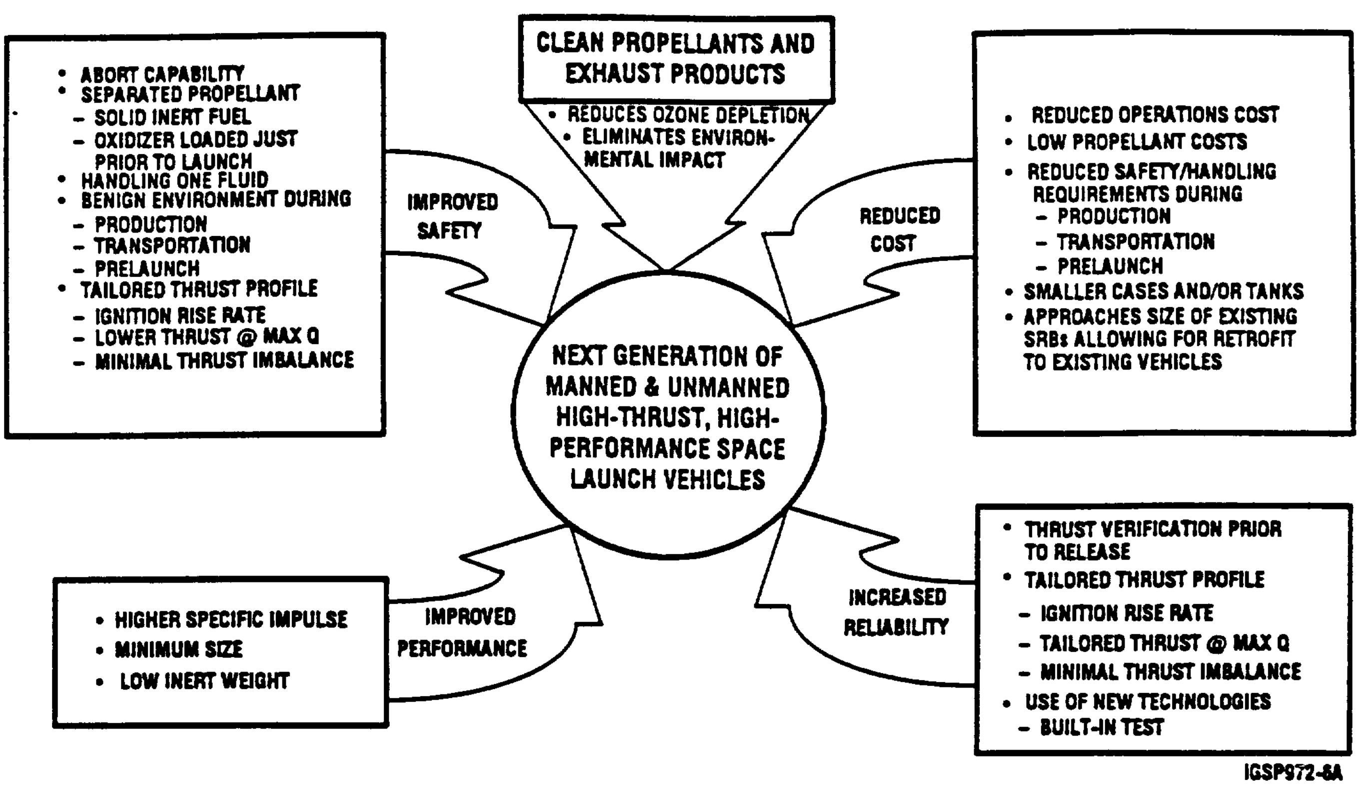


Figure 1-1. Hybrid propulsion offers many advantages over convential strap-on boosters.

because they are very compact for the amount of energy contained. While their propellant bulk density is less than that of SRBs, their specific impulse is higher.

Hybrid propulsion will increase the probability of mission success. Boosters using HPT allow for thrust verification prior to release, soft ignition, controllable thrust, soft shutdown, and may have lower emission of acoustic and vibratory energy.

For HPT to reach the objective, it is necessary to show that it best satisfies a space launch vehicle need. It is very important to focus the technology on that need when it is in its infancy to ensure that the technology will have an application when it matures. Our team is dedicated to this goal, and will ensure that HPT matures sufficiently to be a candidate for the next generation of manned and unmanned space launch vehicles.

2.0 CONCEPT DEFINITION

In order to properly develop the technologies of hybrid propulsion, we evaluated preliminary HP concepts. Our first step was to fully understand and define the requirements. Next, we defined the ranking criteria, Section 2.3, used to refine the preliminary concepts. Those preliminary HP concepts that satisfied the program requirements are defined in Section 2.4.

During the ranking of the concepts, there were details that had to be analyzed to ensure that each concept was given a fair evaluation. We performed system analyses and trade studies to identify technologies applicable to HP. Section 2.5 describes our system analyses trade studies.

In support of our trade studies and evaluations, we conducted a test program. Labscale tests were performed by our subcontractor, Thiokol Corp. During the Phase I effort, we also initiated a subscale test program. Due to our team's strong interest and enthusiasm, both test programs being funded with discretionary monies.

As a result of the system analyses trade studies and labscale testing, we recommended a preferred concept for the ASRM-size and quarter-ASRM size boosters. Section 2.6 further describes these selections.

2.1 METHODOLOGY

HPT concepts were identified, optimized, evaluated, and refined through the iterative process shown in Figure 2-1. This process continually forced improvement of the concepts with respect to the criteria against which they were measured and the requirements they were to satisfy. For each of the ASRM-size and quarter-ASRM size HP systems, one of the refined hybrid propulsion concepts was recommended and further defined with conceptual design and technology identification packages.

Planning for Phases II and III was also accomplished during Phase I. Phase II includes technology acquisition through design, laboratory/subscale testing, and verification of analyses and scaling. Phase III includes large subscale technology demonstration and verification of analyses and scaling.

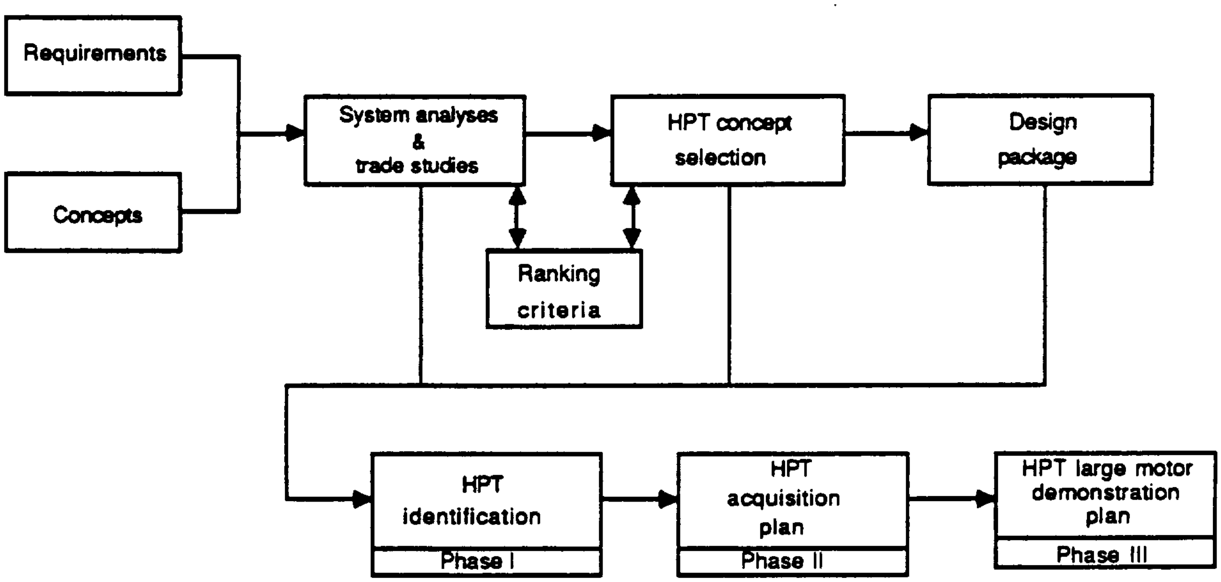


Figure 2-1. A systematic approach identified HPTs and planned their acquisition and demonstration.

Figure 2-2 shows that there were frequent team meetings at rotating locations to assure a meaningful interchange of information among all the program participants. There were four program reviews with the Contracting Officer's Technical Representative (COTR) at the Marshall Space Flight Center (MSFC). In addition there have been six technical interchange meetings, two each at GD, TC, and RD. On Friday, September 15, the Phase I study was reviewed at a meeting of our HPT Review Board. This board is composed of nine senior members with relevant experience from GD, TC, and RD.

2.2 REQUIREMENTS

The requirements listed below provided the basis for the HPT program. They are included in this report to provide a readily available reference.

2.2.1 RFP REQUIREMENTS. The following 10 requirements were contained within requisition/purchase number 1-8-EP-98621. They were reviewed and accepted as pertinent to a successful HPT study.

1. The Advanced Solid Rocket Motor (ASRM) baseline performance requirements are described in Table 2-1. Figure 2-3 gives a performance summary. To encompass a range of possible vehicle system requirements, two hybrid rocket motors shall be conceptualized: a single unit which meets the performance requirements: and a single unit, four of which in combination meet the performance requirements. Two of the large motors or eight of the small motors would be required for one launch vehicle.
2. Design concepts shall use thrust vector control.
3. Design concepts shall not use asbestos or asbestos-containing materials.
4. Design concepts shall use active control system for performance, thrust imbalance, propellant usage, and all transients.
5. Design concepts shall minimize environmentally degrading exhaust .

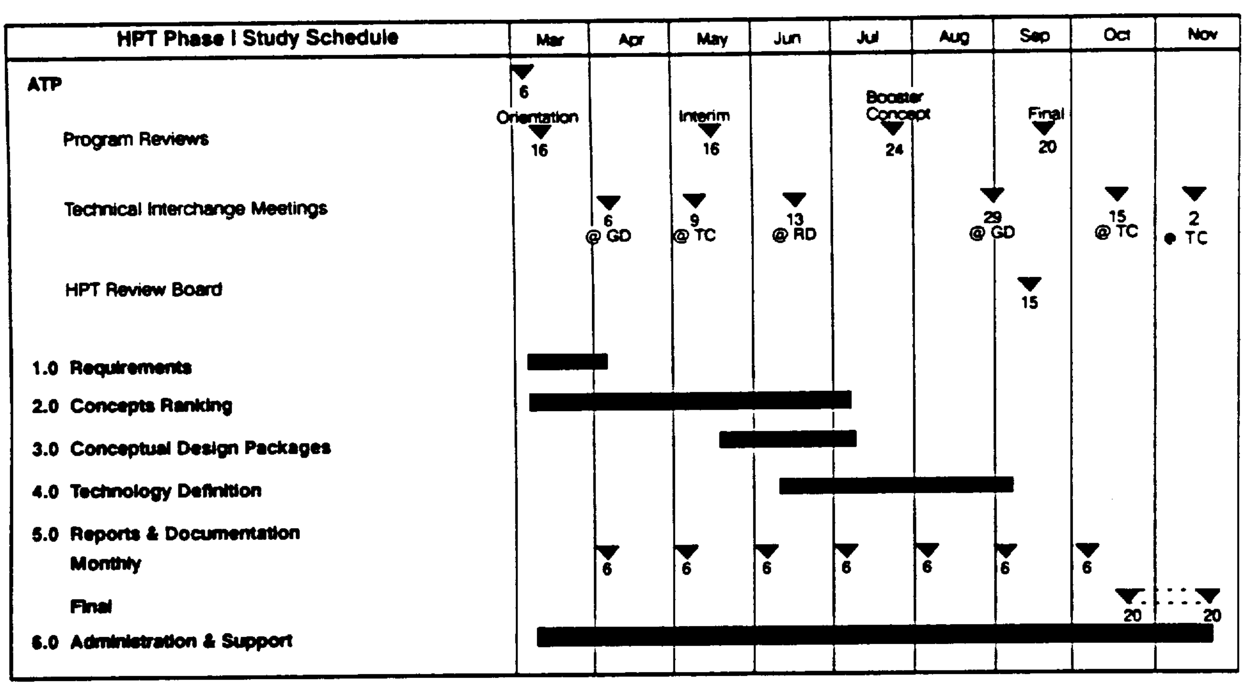


Figure 2-2. There were frequent meetings to assure meaningful interchange of information.

Table 2-1. ASRM reference thrust-time history and associated upper and lower bounds at 60F.

TIME (sec.)	MINIMUM (klbs)	REFERENCE (klbs)	MAXIMUM (klbs)
0.9	3053.2	3147.7	3242.1
1.2	3066.2	3161.1	3255.9
2.0	3081.7	3177.1	3272.4
2.9	3096.7	3192.5	3288.3
3.9	3111.9	3208.2	3304.4
4.9	3138.2	3235.3	3332.4
5.9	3176.6	3274.8	3373.1
6.9	3215.1	3314.6	3414.0
7.8	3248.6	3349.1	3449.5
8.8	3277.0	3378.4	3479.7
9.8	3305.0	3407.2	3509.4
10.7	3314.6	3417.1	3519.7
11.7	3310.1	3412.5	3514.9
12.7	3297.4	3399.4	3501.3
13.6	3287.0	3388.6	3490.3
14.6	3276.3	3377.7	3479.0
15.6	3273.9	3375.2	3476.4
16.6	3279.9	3381.3	3482.8
17.5	3287.4	3389.1	3490.8
18.5	3296.6	3398.5	3500.5
19.0	3300.5	3402.6	3504.7
19.5	3304.0	3406.2	3508.4
19.7	3305.7	3407.9	3510.1
19.9	3306.3	3408.6	3510.8
20.1	3306.0	3408.3	3510.5
20.5	3306.2	3408.5	3510.7
20.9	3303.3	3405.5	3507.6
21.1	3301.8	3403.9	3506.0
21.5	3296.1	3398.1	3500.0
21.8	3280.9	3382.4	3483.8
22.2	3251.3	3351.8	3452.4
22.6	3221.9	3321.6	3421.2
23.0	3199.4	3298.4	3397.3
23.4	3177.4	3275.6	3373.9
24.0	3145.7	3243.0	3340.3
25.9	3063.7	3158.5	3253.2
34.5	2716.0	2800.0	2884.0
47.0	2400.8	2475.0	2549.3
56.0	2172.8	2240.0	2307.2
82.3	2546.1	2624.8	2703.5
115.5	1950.4	2015.2	2084.1
119.1	1140.5	1950.0	2017.5
123.0	436.6	1090.3	1878.1
126.1	198.6	519.5	1218.4
127.1	147.3	403.5	1018.6
128.1	104.6	315.8	839.3
129.1	72.8	251.2	659.9
130.1	49.7	192.7	495.1
131.1	32.3	143.9	387.6
132.1	----	103.0	306.0
133.1	----	72.6	244.0
134.1	----	50.1	188.1
135.1	----	32.9	141.2
136.1	----	----	101.9
137.1	----	----	72.5
138.1	----	----	50.6
139.1	----	----	33.8

6. Design concepts shall maximize shelf life.
7. The solid propellant grain shall extinguish when the fluid propellant flow is stopped. ("Extinguish" is deemed to have occurred when the thrust-to-weight of a booster by itself is less than 70%.)
8. The safety and reliability requirements shall be identical for manned and unmanned systems.
9. To encompass a range of possible mission models, life cycle cost (LCC) shall be based upon a 14-year operational phase including linear growth for four years, then constant flight rate for 10 years. LCC shall be determined based on two flight rates during the 10-year interval: one flight per month and one flight per week.
10 Recoverable and reusable concepts versus expendable concepts shall be evaluated.

2.2.2 ADDITIONAL REQUIREMENTS. The following three requirements were recommended and included in the HPT program.

1. HPT shall be usable for boosters for the next generation of vehicles. HPT for near-term application is preferred over technology for research only with nebulous application in the future.
2. The motor thrust shall be throttleable to 50% of maximum thrust. Thrust throttling is a desirable feature for a boost propulsion system. Space launch vehicles may fly various trajectories depending upon the mass and configuration of their payloads. Different trajectories as well as the overall vehicle configuration usually call for thrust shaping to relieve vehicle loads during maximum Q.

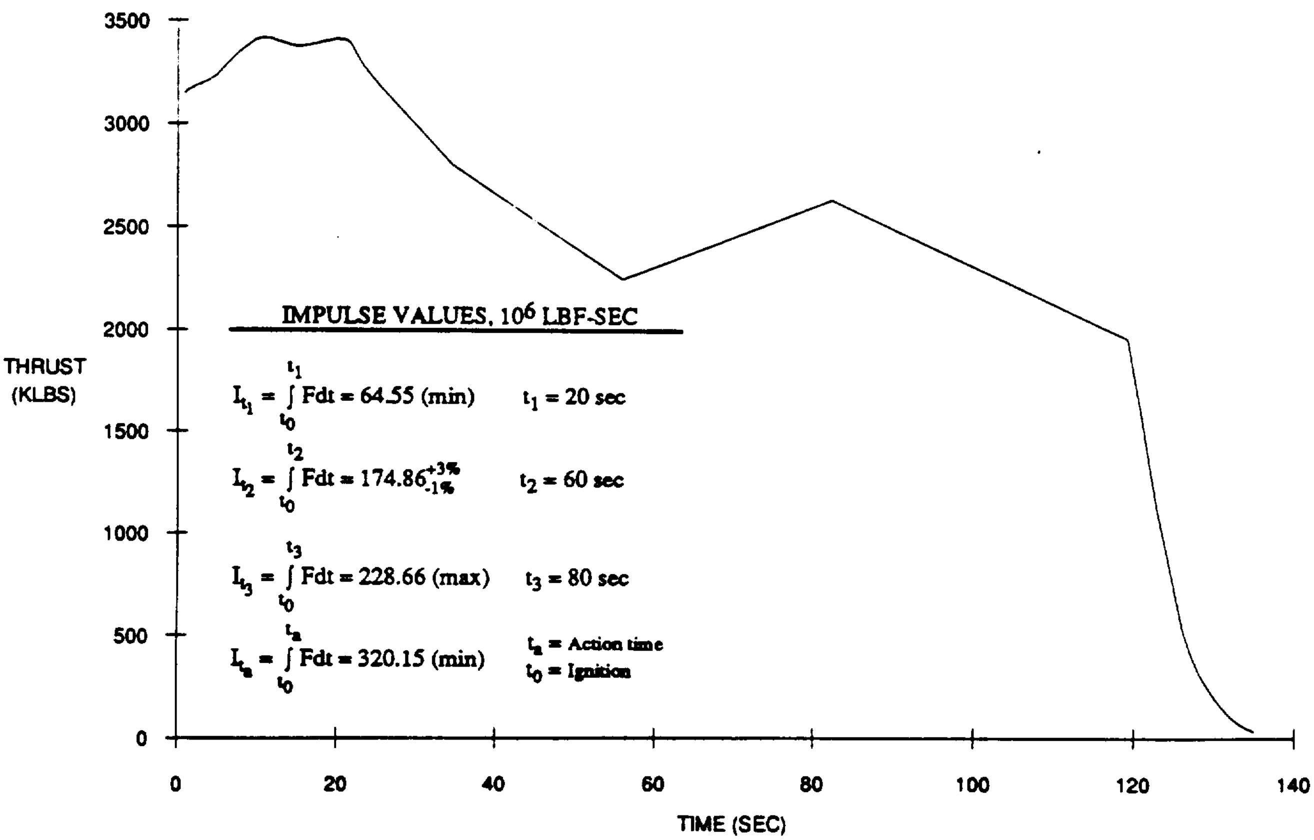


Figure 2-3. Performance summary.

3. HPT shall have predictable scaleability to all booster sizes applicable for the next generation of manned and unmanned space launch vehicles. Predictable scaleability is a necessary measure of the maturity of a technology. It provides assurance that the technology will successfully transition through full-scale development into a viable product.

2.3 RANKING CRITERIA

The ranking process provides insight into the attributes and weaknesses of alternate HPT concepts. When comparing concepts, it provides a relative measure that allows the concepts to be ranked in order of desirability. A numerical rating factor is assigned to each of the ranking criteria. Numerical rating factors are given to reduce the effect of evaluator bias on the analyses of the alternatives and to facilitate comparison among criteria. They are applied to each criterion by assessing its relative importance.

Concepts ranking is based on the following criteria, in order of priority:

- Flight safety and reliability
- Life cycle cost
- Performance
- Operational considerations

2.3.1 FLIGHT SAFETY, RELIABILITY. Flight safety and reliability was allocated the highest priority, because in a manned system assuring crew safety is the most important consideration. In our society human life is most precious. The HPT concepts selected must provide the best opportunity for human survival at all times.

Launch vehicle users require demonstrated reliability that gives them confidence that their payloads will be placed safely into their requested orbit as scheduled. Their provided payloads are sophisticated, requiring significant time and materials to design, develop and manufacture. Some are unique, being one of a kind, very valuable, and almost irreplaceable. It is not unusual that the payload costs more than the launch services.

Flight safety and reliability applies to the extent which alternative HPT concepts minimize hazards at the launch facility and to the launch vehicle from booster arrival until booster separation. Ranking criteria elements include thrust termination, propellant toxicity/explosive hazard, operational contingency modes, failure detection, susceptibility to induced failures, and least critical failure modes.

2.3.2 LIFE CYCLE COST. Life cycle cost was allocated second priority. There is a national need for low-cost transportation of payloads into space. Last year Congress mandated the pursuit of a recurring cost per pound of vehicle capability placed in or near low-earth orbit of $300 or less at all flight rates of 25 flights per year or greater, achievable by the year 2005.

Current launch vehicles were designed for performance, and incorporate the technology from their design era. They typically cost about $3,600 (Titan IV) per payload pound to orbit. We can significantly reduce this cost by $2,000/lbm by using the design-to-cost process, by the economy of large payload capability, higher launch rates, more producible design, carefully managing design margins to maximize cost-effectiveness, improved quality, and standardized procedures and interfaces.

Further cost reductions (an additional $1,300/lbm) must come from incorporating appropriate new and essential technologies to reduce costs to manufacture and launch space launch systems. A HP system for launch vehicle boosters is one of the more promising of the appropriate new and essential technologies. It has a high probability of obtaining the allocated cost goal of $78/lbm for booster costs (Figure 2-4).

Current launch boosters entered full-scale development long before the ongoing exponential advances in material improvements, microelectronics, and digital technology. By using these commercially and government-developed applications for information processing, systems simulations, built-in-test, health diagnosis, and control, we can achieve a major reduction in the operations cost of placing payloads into orbit.

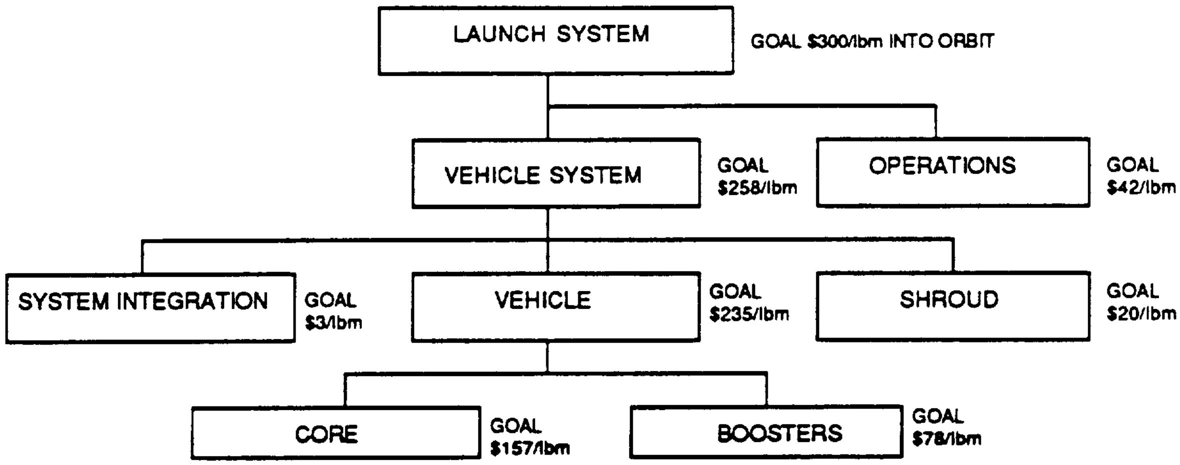


Figure 2-4. Cost goals for the hybrid propulsion booster.

When they are combined with the additional cost-reducing HP attributes such as the combustion of low-cost and inherently safe propellants, the probability of achieving the allocated cost goals for vehicle boosters becomes even more realistic.

Non-recurring LCC includes all costs (undiscounted) incurred for full-scale development. It assumes all technology has been identified, acquired, and demonstrated. It includes the design, development, test and evaluation of full-size or quarter-size ASRM boosters for a specific application. It excludes production of all flight hardware. Ranking criteria elements include full-scale development of the boosters and activation costs for the launch facility.

2.3.3 PERFORMANCE. Performance was allocated third priority. It is an indicator of booster "goodness." It directly reflects changes in propellant mass fraction and specific impulse, which in turn affects booster size and weights. It is readily evaluated and provides definitized ranking criteria.

Performance is meeasured by the relative effect of candidate concepts on the vehicle velocity at booster burnout, or on the mass of payload placed into low-Earth orbit. The current STS was baselined as the considered vehicle.

2.3.4 OPERATIONAL ASPECTS. Operational considerations at the launch facility was allocated fourth priority. There is a desire to ease the complexity of launch operations and thus reduce launch cycle times and cost. Launch facilities and their operation provide unique services for one particular launch vehicle. As a result, it is possible to temper the launch vehicle and thereby reduce the magnitude of the operational considerations.

Ranking criteria elements considered included rapid component turnaround; insensitivity to faults; non-toxic, inert propellants; inexpensive or existing facilities and equipment built-in test and check-out.

2.3.5 RANKING METHODOLOGY. Figure 2-5 presents the assigned rating factors for each of the four rating criterion and their further breakdown into sub-criterion. The relationship is shown by the rating factors presented in parentheses. The assigned rating factors total 1.0, which is divided according to the importance of the factors whereby progressively lower values have less importance.

Figure 2-6 presents the final ranking of candidate HPT concepts. Each criteria is scored from zero to one hundred where one hundred is the best score. The scoring emphasizes the relation of the scores between

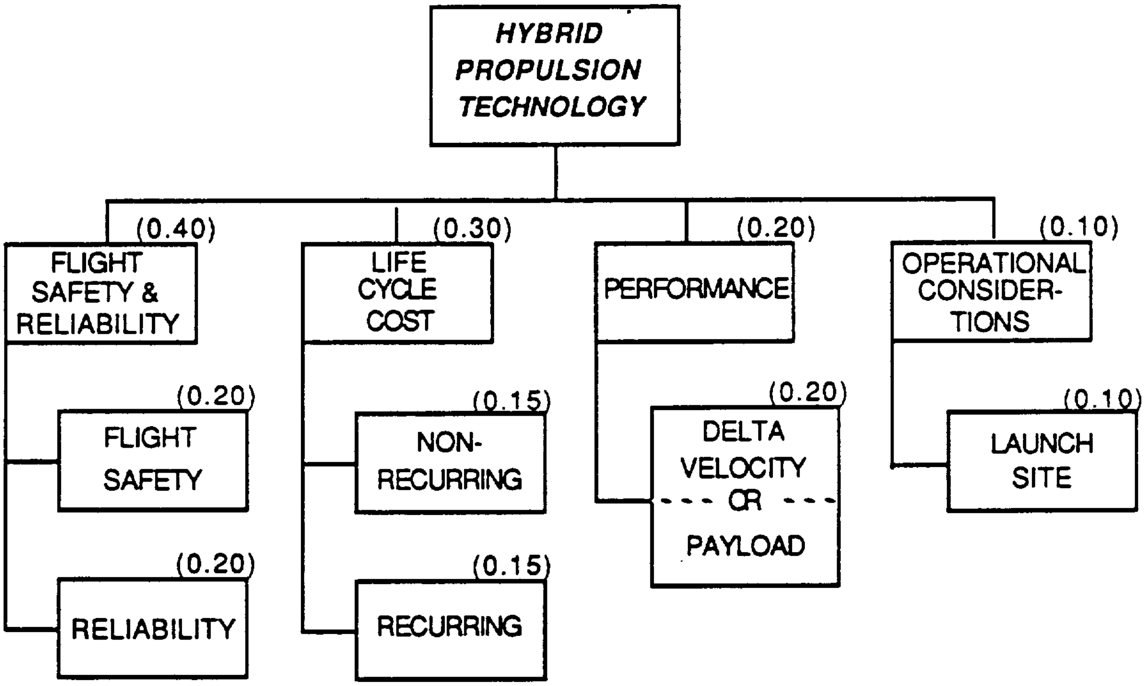


Figure 2-5. The rating factors show the relative importance of each criterion.

CRITERIA (RATING FACTOR) / CONCEPT	#1		#2		#3	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
<u>*FLIGHT SAFETY & RELIABILITY*</u> • FLIGHT SAFETY (0.20)						
• RELIABILITY (0.20)						
<u>*LIFE CYCLE COST*</u> • NON RECURRING (0.15)						
• RECURRING (0.15)						
<u>*PERFORMANCE*</u> • DELTA VELOCITY (0.20)						
<u>*OPERATIONAL CONSIDERATIONS*</u> • LAUNCH SITE (0.10)						
TOTALS						
RANK						

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST

Figure 2-6. Ranking of HPT concepts.

each of the concepts to be ranked. "Comparison" scoring provides a direct insight into perceived advantages and disadvantages of each with relative importance. The scores are multiplied by the respective rating factors to obtain the "weighted score." These are then added to provide a total for comparison and insertion of the rank of each concept evaluated.

2.4 HYBRID PROPULSION CONCEPTS. A number of booster propulsion system concepts are being considered for the next generation of manned and unmanned space launch vehicles. The one concept that has potential for reducing costs with increased safety, reliability, and performance is hybrid propulsion.

A hybrid propulsion system may be thought of as a liquid propulsion system with solid fuel or as a solid propulsion system with a liquid oxidizer. As shown in Figure 2-7, the hybrid propulsion system extracts the best features of both the liquid and solid propulsion systems, and supplements them with additional features that neither currently incorporate.

The liquid propulsion features that are most attractive are the higher specific impulse, clean exhaust, separated propellants, and oxidizer loading just prior to launch. The higher specific impulse requires less propellants to reach the specified delta velocity. The clean exhaust is in keeping with current environmental concerns. Maintaining separated propellants increases safety during manufacturing, processings, and flight. With the oxidizer loaded just prior to launch, less weight needs to be transported and erected, and safety is increased by the

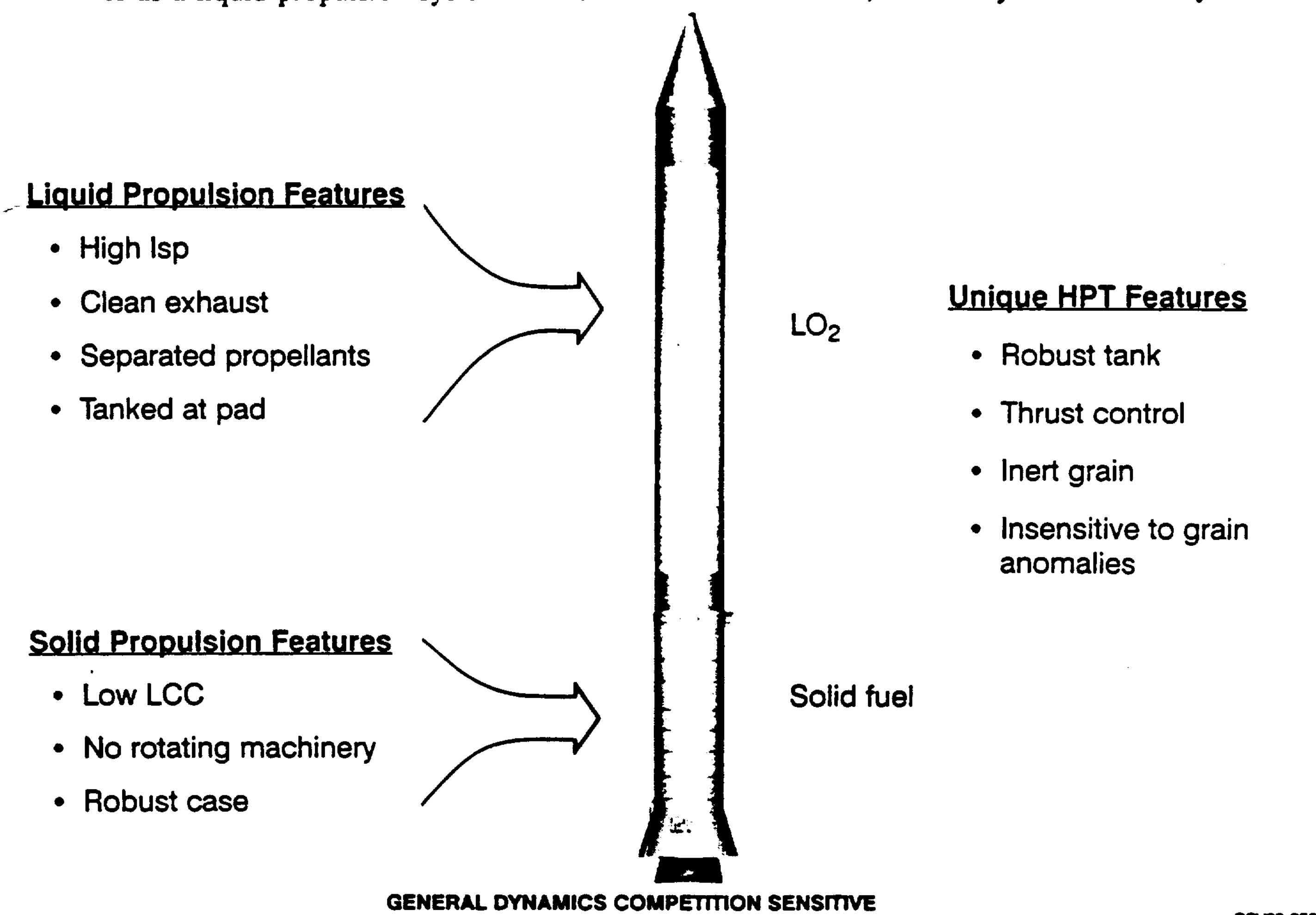


Figure 2-7. HPT integrates the best features of liquid and solid propulsion systems.

absence of oxidizer until the area is vacated for launch.

The most attractive solid propulsion features include low life cycle costs, no rotating machinery, compact size, and a robust case. In addition, a hybrid propulsion system has a robust LO2 tank; provides thrust control for ignition, to alleviate flight loads, and for thrust termination; and uses an inert grain that is not sensitive to anomalies such as cracks, voids, and separations.

Nine hybrid propulsion concepts were studied as candidates for the selected hybrid propulsion concept. The three most promising concepts are shown in Figure 2-8. All use a liquid oxidizer and an inert or live grain. The other six hybrid propulsion concepts do not have any liquid oxidizer. They burn a solid oxidizer with a liquid or solid fuel.

2.4.1 CLASSICAL HYBRID PROPULSION CONCEPT. The classical HP concept was subsequently selected as the preferred concept for acquisition and demonstration. Its main features are the injection of oxidizer into the forward volume only of a combustion chamber containing an inert fuel grain. It was most attractive because it uses an inert grain that is non-hazardous in the absence of an oxidizer; is forgiving as to anomalies such as cracks, voids, and separations; requires only short oxidizer feed lines with single flow controls; and uses only low-cost energetic liquid oxygen for the oxidizer.

Approximately one-half percent of performance is sacrificed to obtain these desirable features. The performance penalties are inherent in a classical concept which throttles. During motor operation optimum mixture ratio will be attainable only at a specific point during the grain regression. Specific throttling demands to meet a thrust/time requirement will contribute to degradations in combustion C* performance. Specific impulse degradation may be minimized by shaping the thrust time curve.

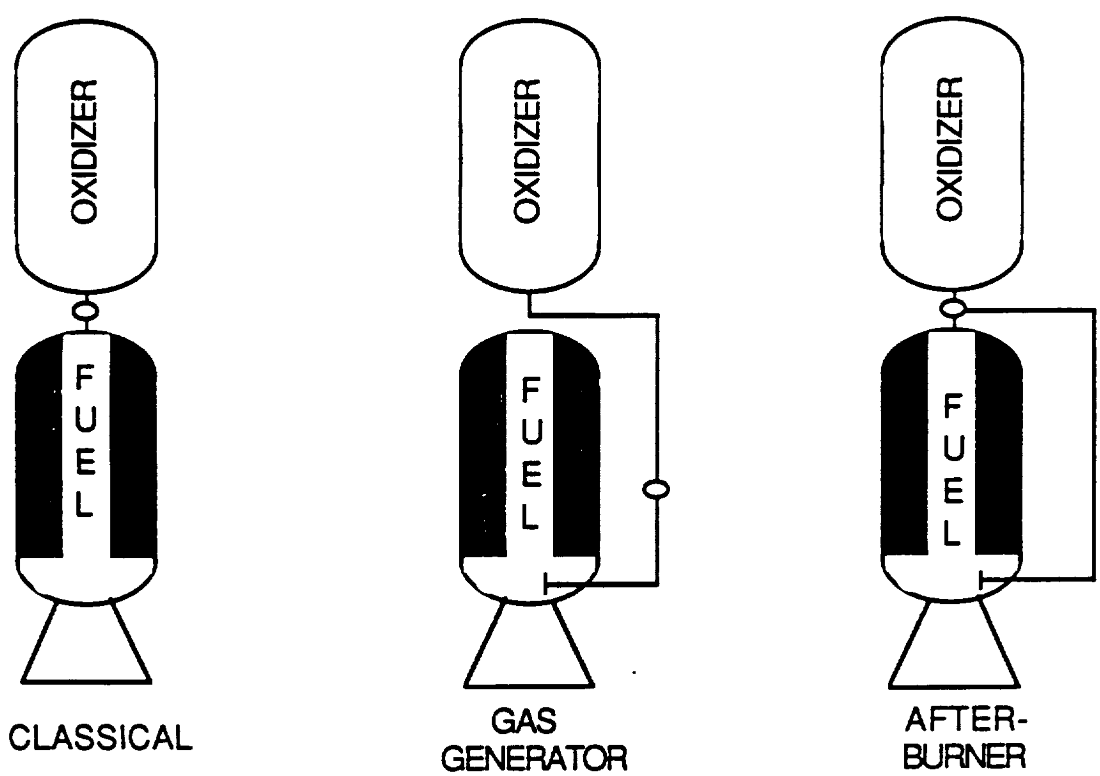


Figure 2-8. The three promising hybrid propulsion concepts.

2.4.2 GAS GENERATOR HYBRID PROPULSION CONCEPT. Development difficulties associated with grain regression in classical hybrid motors can be corrected by the approach illustrated as the gas generator concept. The fuel grain contains sufficient oxidizer to sustain combustion and exhibit the well-known solid rocket burn rate pressure dependence. Fuel-rich gases from this grain are burned at optimum mixture ratio by secondary injection of oxidizer in an afterburner chamber.

Throttling is achieved by varying the flow rate of the liquid oxidizer into the afterburner. Injection of additional mass into the combustion chamber increases pressure and thrust. The increased chamber pressure results in a higher fuel production rate from the gas generator to balance the mixture ratio.

Performance losses attributable to this concept may be expected from off-optimum mixture ratio excursions during throttle transients and from the inherently less energetic result of adding oxidizer to the fuel grain. The self-sustaining nature of the fuel grain also raises the normal issues of concern in solid rocket motors such as grain defects and debonding. However, studies have shown that a reduction in the chamber pressure does cause the gas generator grain to self-extinguish.

2.4.3 AFTERBURNER HYBRID PROPULSION CONCEPT. The afterburner hybrid propulsion system is identical to the classical with an added aft liquid oxygen system to maintain the combustion mixture ratio near optimum for the entire burn.

The HP system's fuel grain is operated in a fuel-rich combustion mode at all times during the motor burn. This is accomplished by injecting only a portion of the oxidizer required for proper mixture ratio operation at the head end of the fuel grain. The remainder of the oxidizer is bypassed and injected into a chamber at the aft end of the grain.

Mixing of oxidizer and fuel gases is enhanced by the injection process, and a nearly constant mixture ratio can be maintained at all times during motor operation. This motor combustion cycle theoretically delivers the highest specific impulse performance of any concept currently identified but at the cost of increased complexity in oxidizer bypass plumbing, injection and flow control. This technique is particularly suited to throttling applications and has the added advantage of an inert fuel grain.

2.4.4 OTHER HYBRID PROPULSION CONCEPTS. The six hybrid propulsion concepts presented in Figure 2-9 do not fully satisfy the HPT Program requirements. They will not mature for application as boosters for the next generation of manned and unmanned space launch vehicles.

Concepts 1 through 4 burn self-sustaining grains that exhaust oxidizer-rich gases into a fuel-rich grain, or vice versa. These self-sustaining grains do not satisfy the requirement that "the solid propellant grain shall extinguish when the fluid propellant is stopped."

Concepts 5 and 6 flow a fuel into an oxidizer rich grain for combustion. The fuel flow can be interrupted, but the oxidizer and binder are in the same grain and will continue to burn. Also, the exhaust will contain "environmentally degrading exhaust products" in order to obtain acceptable performance. A high-performance oxidizer such as ammonium perchlorate is required to increase the combustion specific impulse to an acceptable level. Such a grain composition would exhaust hydrogen chloride which is environmentally degrading.

2.5 SYSTEM ANALYSES AND TRADE STUDIES

The system analyses and trade studies shown in Figure 2-10 refined the preliminary hybrid propulsion concepts to their likely configuration at maturity. These analyses and trades also identified the required technology and provided a merit of benefit when applied to HPT. The initial analyses and trades center on the ballistic characterization of the motor. These include propellant selection, motor performance,

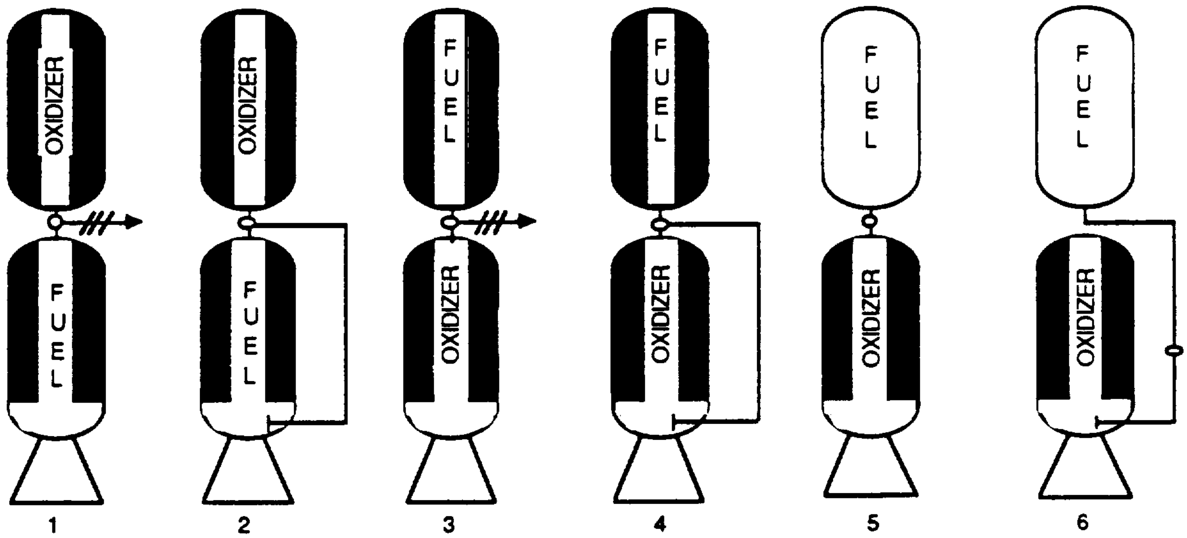


Figure 2-9. Six hybrid propulsion concepts lacking promise.

pressure/area ratio, oxidizer injection, ignition system, thrust control, and combustion stability.

The second set of analyses and trades relate to efficiently providing the oxidizer for combustion. They recommended a particular pressurization system and whether the oxidizer should be pump- or pressure-fed to the injector(s). The remaining analyses and trades concern reusable versus expendable, structure and insulation, oxidizer supply, and thrust vector control. The study leader for each of the system analyses and trade studies was assigned as shown. The study leader

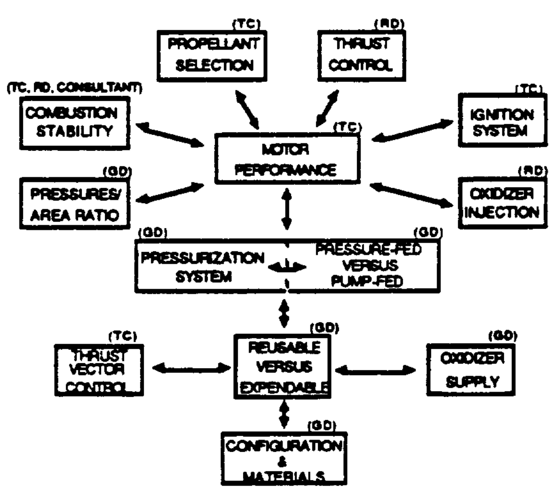


Figure 2-10. System analyses and tradestudies selected the preferred approach.

was selected because of his having the best available information and having the most involvement in the pertinent subject. The study leader was supported by the other two companies, and each trade study drew on the assets of all three teams.

System analyses and trade studies were performed with a formal systems engineering approach as depicted in the flow diagram, Figure 2-11. The initial Engineering Review Board (ERB) meeting was scheduled after the trade team generated alternatives and screened out those that obviously did not meet the requirements. The ERB approved the initial work, sometimes with revisions to the objectives, requirements, assumptions, and trade tree of alternatives.

The alternatives were evaluated by first defining them for a particular hybrid propulsion concept, then applying the ranking criteria and ranking factors. During the definition and evaluation process meetings occurred with related trade study teams and peer reviews to ensure coherency. Next, sensitivity analyses and adverse consequences analyses were performed to provide confidence that the concepts selected were the best solution. The final ERB met to review and approve the results of the study and/or suggest further work.

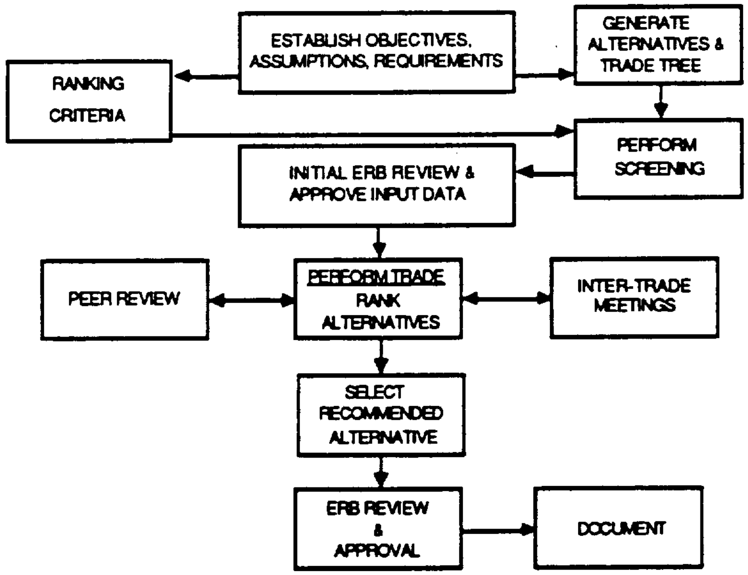


Figure 2-11. System analyses and trade studies followed a formal systems engineering methodology.

2.5.1 OXIDIZER SUPPLY. The objectives of the oxidizer supply system analyses and trade studies were to provide information on the recommended structure and configuration of liquid oxidizer supply system and to provide parametric data to support other studies. The new technology areas were to be highlighted for development.

2.5.1.1 Requirements. The applicable requirements from the Statement Of Work (SOW) were imposed on every system analyzed. The requirement of active control was met where applicable. Booster performance was based on the thrust/time profile required of the ASRM for the full-size booster, and 25% of that for the quarter-size unit. Safety and reliability was maintained at a man-rated level. Life cycle cost was be based on production quantities and schedule requirements as given in the SOW.

General configuration was determined for each of the two booster sizes based on an afterburner-type booster with 1200-psia LO2 tank pressure. Propellant conditioning, flow rates and transients, residuals, movement, and ground support considerations were among the parameters evaluated. For the flow rate calculation, several motor performance parameters were baselined. The propellant combination of LO2 and HTPB at a mixture ratio of 2.33:1 with an Isp of 306 lbf-sec/lbm set the necessary design requirements.

LO2 supply hardware was provided for both the primary and afterburner injection systems. These two systems were sized for 100% LO2 flow to the primary ports, and 10% LO2 flow to the afterburner ports at peak thrust.

The result of this system analysis and trade studie was a recommended oxidizer supply system for the hybrid rocket booster (HRB). Due to its dependency on other system studies, this analysis was performed for both the pressure-fed and the pump-fed systems in each of the two candidate booster sizes. The results of the evaluation established optimal aspects of injector plumbing including number and length of injector feed ducts, movement compensation/isolation, and LO2 conditioning schemes. The baseline configuration is highlighted in the trade tree in Figure 2-12. This baseline served as the point of departure in the analysis.

2.5.1.2 Required LO2 Flow Rates. In a hybrid motor the size of the full-scale booster, it is necessary to supply a large quantity of LO2 to the grain case for combustion. Given the thrust requirements

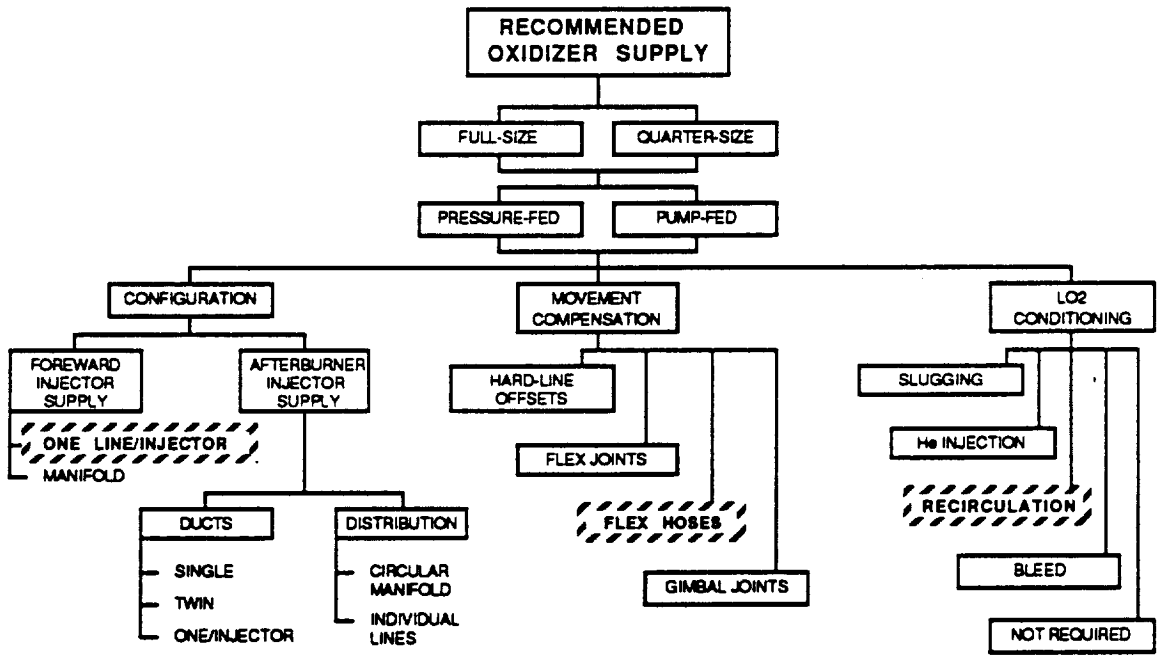


Figure 2-12. Oxidizer supply trade tree with point of departure highlighted.

from the SOW, a diagram depicting the required LO2 flow rates can be generated.

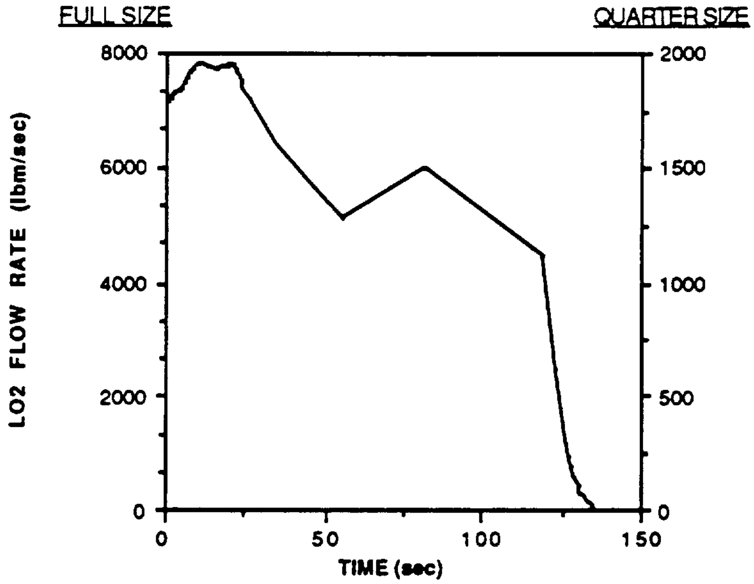
The plot shown in Figure 2-13 is based on an Isp value of 306 lbf-sec/lbm and an oxidizer-to-fuel ratio of 2.33 to 1. The full-size HRB has a peak oxidizer flow rate of nearly 8000 lbm/sec, and the quarter-size HRB exactly one-fourth that, as read from the axis to the right of the curve in the figure. For comparison, the peak flow rates of some other large liquid propellant engines are shown.

2.5.1.3 Diameter vs. Number of Ducts. Although a flow rate of 3/4 of the peak level is satisfactory for 70% of the burn time, the LO2 feed system must be sized for the maximum expected flow rate. As can be seen in Figure 2-14, the more feed ducts used to deliver the LO2, the smaller the individual diameters can be. If a single feed duct were used to supply all of the LO2 to a full-size motor at 25 ft/sec, it would have to be almost 30 inches in diameter. This would not present any technical problems; however, it would require a 30-inch diameter control valve and enough inter-tank separation distance to accommodate it.

On the other hand, using four 15-inch ducts allows much smaller control valves, more refined control, and by distributing the ducts in a circumferential pattern, the tank bulkheads can be moved closer together to take advantage of their convex shape. The afterburner ducts show a similar trend and a maximum of four afterburner ducts was considered since the size benefit for additional ducts diminishes rapidly.

With all of the duct configurations presented, the total mass of the ducts themselves is nearly constant, independent of the number selected. The total weight of the valves used does increase slightly with an increase in the number of feed paths, but the difference is very slight. A real distinction in system configuration can be seen when assessing system reliability, cost, and operations impacts.

2.5.1.4 LO2 Supply Mass vs. Pressure and Flow. The mass of the supply system including ducts, valves, movement compensation, flanges, and supports is depicted in Figure 2-15 for both the full- and quarter-size boosters. It is seen to be both pressure and mass flow-rate dependent. At the two desired flow rates the



LO2 FLOW RATES (lbm/sec)
(FOR COMPARISON)

F-1 ENGINE	= 3,931
STEP	= 1,145
SSME	= 989
RS 27	= 539

Figure 2-13. Oxidizer flow rates required for the ASRM-size and quarter-ASRM-size motors. Also shown are required oxidizer flow rates for existing engine systems.

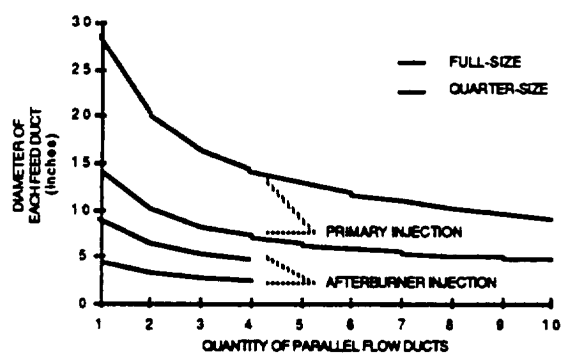


Figure 2-14. LO2 duct diameter-reduction trend flattens out quickly above four primary and two afterburner paths.

system mass can be approximated as linear functions of pressure. This relationship was supplied as an input to the pressures/area ratio and motor performance analyses and studies.

2.5.1.5 Primary Feed System Movements. The ducting required for the LO2 supply system is composed of two separate systems: primary injection and afterburner. The primary injection system for the pressure-fed system consists of four injector valves mounted directly to the LO2 tank supply flanges (Figure 2-16). The valves meter the flow of oxidizer to the grain and allow for throttling and thrust termination at staging or abort. Four 15-inch vertical flex lines run from the control valves to the case-mounted injectors.

The use of individual control valves and feed lines for each injector provides several advantages over a system based on a single LO2 tank supply flange with a distribution manifold between the tank and case. The minimum practical radius of curvature of the manifold ducts mandates a greater separation distance between the tank and case. By installing the distribution ducting within the LO2 tank, the amount of hardware between the tank and case is minimized, and several potential external leak paths are eliminated.

Since the grain will be designed with multiple sectors (parallel combustion tunnels), it may become necessary to trim the performance of the individual ports. Grain regression rate and combustion efficiency can be specifically tailored in each sector. Through the adjustment of the supply valves, control of the LO2 flow is possible, allowing trimming of the individual burn rates. Additionally, in the event of valve failure or inadequate sector performance, the remaining control valves can provide compensation through an increase in their nominal flow rate.

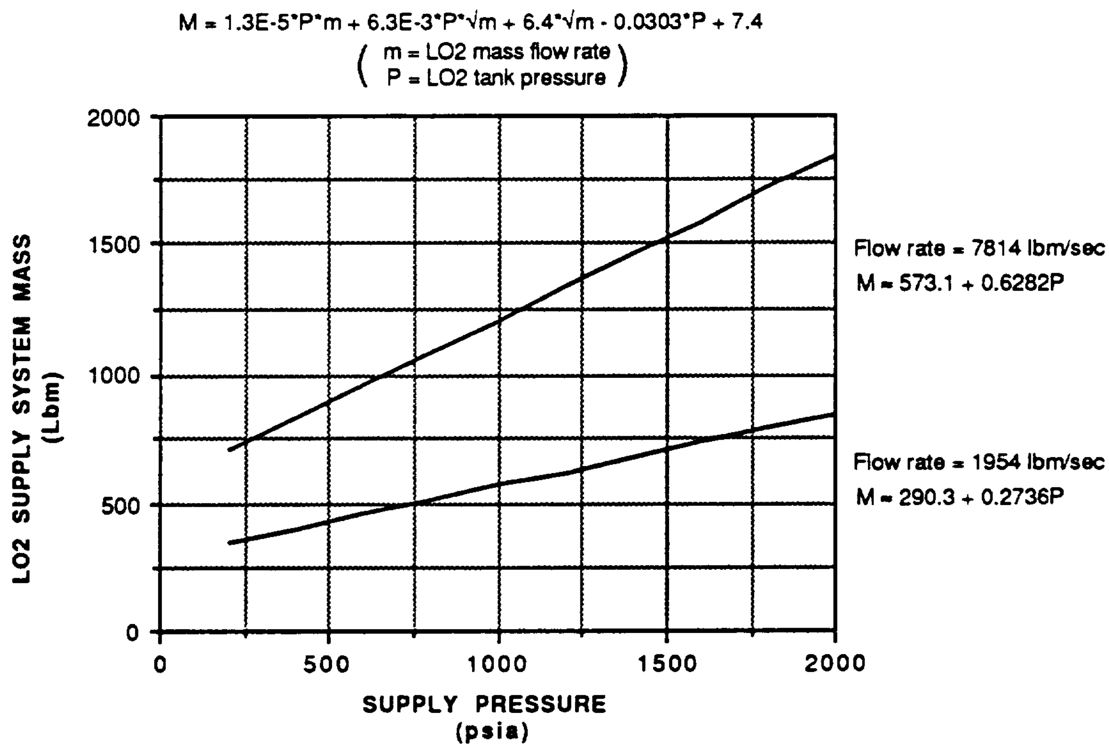


Figure 2-15. The total LO2 feed system mass is a roughly linear function of pressure.

THE CONNECTION COMPONENT EXPERIENCES
THE FOLLOWING AXIAL MOTIONS (INCHES):

DURING:	TENSION	COMPRESSION
ASSEMBLY	1.0	1.0
TANK COOL DOWN	0.2	-
DUCT COOL DOWN	0.1	-
PRESSURIZATION	-	0.5
FLIGHT	0.1	0.1
TOTAL AXIAL MOTION	1.4	1.6

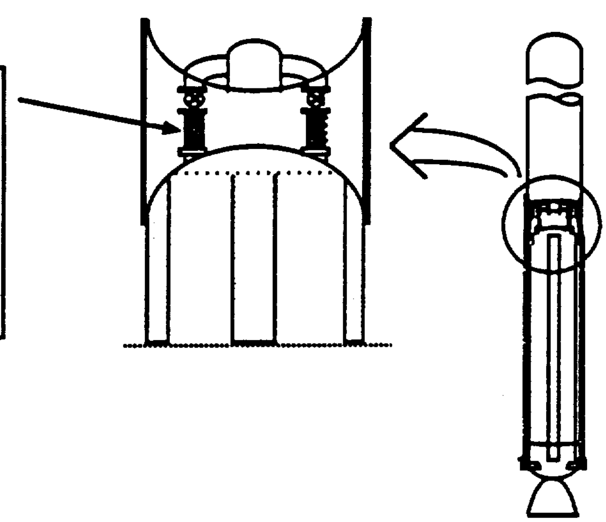


Figure 2-16. Primary feed system and the movements which must be accommodated by the motion compensation hardware.

To accommodate the range of movement required by the feed system, the ducting must include flexible elements. Large-diameter flex hose was selected for the primary injection system. The use of flex hose as the only conduit type in the primary lines allows maximum flexibility as well as the minimum number of interfaces and potential failure points. Flex hose also allows greater ease of assembly and allows less stringent tank, case, and adapter manufacturing tolerances. The effects of tank elongation under pressure, bulkhead and duct thermal contraction, and movement required for assembly are all assessed to determine flex hose suitability (Figure 2-16).

Each control valve is positioned immediately aft of the LO2 tank exit flanges, and bolted directly to the flex hoses. This placement provides a safe LO2 shut-off capability in case of hose failure. Additionally, the valves are better isolated from the effects of the combustion process, including the thermal gradient which may be difficult to accommodate.

2.5.1.6 Afterburner Feed System Movements. The afterburner portion of the supply system is composed of two 6-inch descending ducts that terminate at control valves near the booster nozzle (Figure 2-17). Flow through these valves is expelled from aft injector nozzles into the fuel-rich exhaust stream. Although both control valves will be commanded to the same opening, having two valves simplifies the ducting, increases redundancy, and provides a method for LO2 conditioning.

Movement compensation will be handled by a section of flex hose and compensator joints. The radial and lateral strains at the bottom LO2 tank bulkhead due to LO2 thermal effects will be accommodated by the flex hose mounted aft of the tank flanges. The bellows joints contained in the descending supply lines will accommodate both the 3.6-inch thermal strain, and the 2-inch case length change due to vehicle stacking and case pressurization at ignition. These requirements are listed in Figure 2-17. Two ducts running the length of the case will have different thermal absorption characteristics due to sun, shade, wind, and other inevitable effects. As indicated in Figure 2-18, thermally induced LO2 density variations produce a recirculation action of the LO2 within the ducts and through the connecting duct across the bottom of the case. This constant flow will maintain propellant condition in a passive manner, and the process tends to be self-regulating.

THE CONNECTION COMPONENTS EXPERIENCE THE FOLLOWING AXIAL MOTIONS (INCHES):

DURING:	TENSION	COMPRESSION
ASSEMBLY	2.0	2.0
TANK COOL DOWN	0.2	-
DUCT COOL DOWN	3.6	-
PRESSURIZATION	2.0	-
FLIGHT	0.4	0.4
TOTAL AXIAL MOTION	8.2	2.4

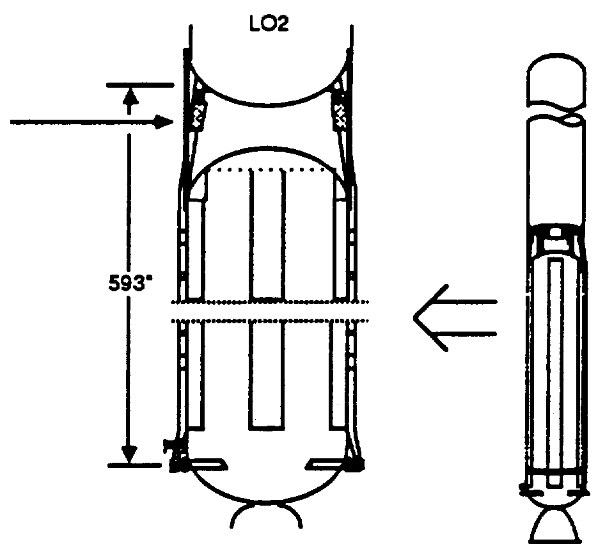


Figure 2-17. Afterburner feed system and the movements that must be accommodated.

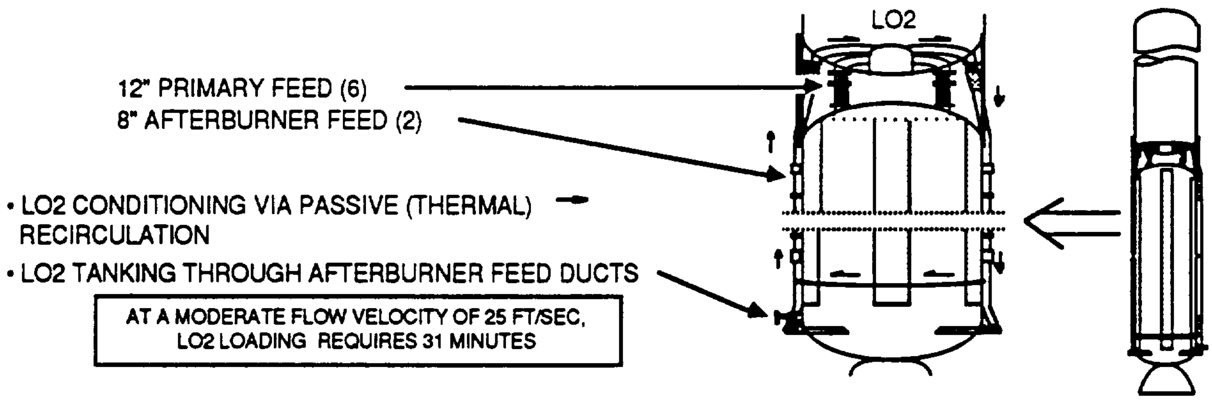


Figure 2-18. The recommended baseline feed system which was used for evaluation of required new technologies.

With the afterburner supply lines in place near the booster nozzle, an additional advantage is provided for booster LO2 tanking. One of the descending ducts contains a three-way valve upstream of the injector control valve to provide a flow path from a ground support equipment (GSE) disconnect to the LO2 tank. Tanking LO2 through a single 6-inch branch of the afterburner supply system would require only 33 minutes at a moderate flow velocity of 25 ft/sec (5 cubic feet/sec).

The recommended oxidizer supply system baseline suitable for technology review has been presented in the three previous figures. At this level of review, selection of an exact supply system configuration yields no additional insight into the new technology required or the ability to meet the requirements of the study.

Several concepts of the system will be required in any configuration chosen. Regardless of the booster concept used there are several components that will necessarily be incorporated. These include high-pressure motion compensators; high-pressure cryogenic, throttleable valves; and high-pressure ducting and joint components, all of which being LO2-compatible.

This analysis found no enabling oxidizer supply system technologies required for HRBs. All of the recommended components are either currently available, or are within the industry's current production capability.

2.5.2 PRESSURES/AREA RATIO. The objectives of the pressures/area ratio system analysis and trade study were to recommend desirable case and LO2 tank operating pressures, and to determine an optimum nozzle-exit/throat-area ratio. Parametric data for this study was taken as input from the pressurization system and configuration and materials system analyses and trade studies. Results from this study are presented as performance measured by ΔV (approximate velocity at burnout).

Several SOW requirements were found to be applicable to this analysis and adhered to throughout the work. Booster performance was based on the thrust/time profile required of the ASRM for the full-size booster, and 25% of that for the quarter-size unit. The baseline level of safety and reliability was that of a man-rated system. Life cycle cost was based on the production quantities and the scheduling requirements listed in the Statement of Work.

2.5.2.1 Assumptions. An afterburning motor concept was used as the baseline concept for analysis. With a propellant combination of LO2 and HTPB, a mixture ratio (O/F) of 2.33 was reasonable and was used for all related calculations. It was assumed that there would be a 20% LO2 pressure drop across the oxidizer injectors at the flow rates required.

This provided the pressure requirements of the pressurized portions of the oxidizer supply system, including the tank, in the pressure-fed designs. The tank, case, and nozzle diameters were assumed to be 150 inches, for the ASRM-size HRB and 90 inches for the quarter-ASRM size.

The volumetric efficiency of the grain case was set to be 76% based on total case volume. Delivered Isp was assumed to be 92% of the computed theoretical vacuum specific impulse. Calculations regarding the pressure-fed motor used a cascaded Tridyne system to charge the LO2 tank. The pump-fed system analyses were based on an LO2/RP-1 gas generator turbopump configuration. Both of these systems are explained in detail in Section 2.5.3.

In all of the ΔV calculations a pair of ASRM-size HRBs and a group of eight quarter-ASRM-size HRBs comprised the booster system for the ASRM-size and quarter-ASRM-size analyses, respectively. Orbiter/external tank weights of 1.9 Mlbf at lift-off, and 1.5 Mlbf at booster staging were assumed. Additionally, an Isp of 452 lbf-sec/lbm was used for the Space Shuttle Main Engine performance.

The trade tree in Figure 2-19 outlines the parameters of interest in the Pressures/Area Ratio trade study and analysis. This investigation was designed to determine the optimum LO2 tank and grain case pressures and optimum nozzle size and area ratio. In this analysis, the Isp increase and the weight penalty resulting from an increased operating pressure were evaluated as they related to vehicle performance. A separate evaluation was performed for the ASRM-size and quarter-ASRM-size HRBs, each considering pressure-fed and pump-fed configurations.

Both the tank and the case of the ASRM-size HRB were required to withstand the on-pad static load (support STS stack) and dynamic load (just prior to booster ignition) as required of the current STS SRB motors. Although the case and tank were sized to support a baseline vehicle weight, their pressure-carrying capability was not fully exploited in every configuration. This was especially true for the pump-fed systems which had a tank strength capable of withstanding up to 570 psia, but were run with an oxidizer pressure of only 80 psia.

Because an increase in pump-head yields a decrease in the pumping system weight, one would expect an optimum pump inlet pressure above 80 psia. However, the weight of the additional pressurization gas required for the greater pump-head overcomes any benefits in decreased pumping system size.

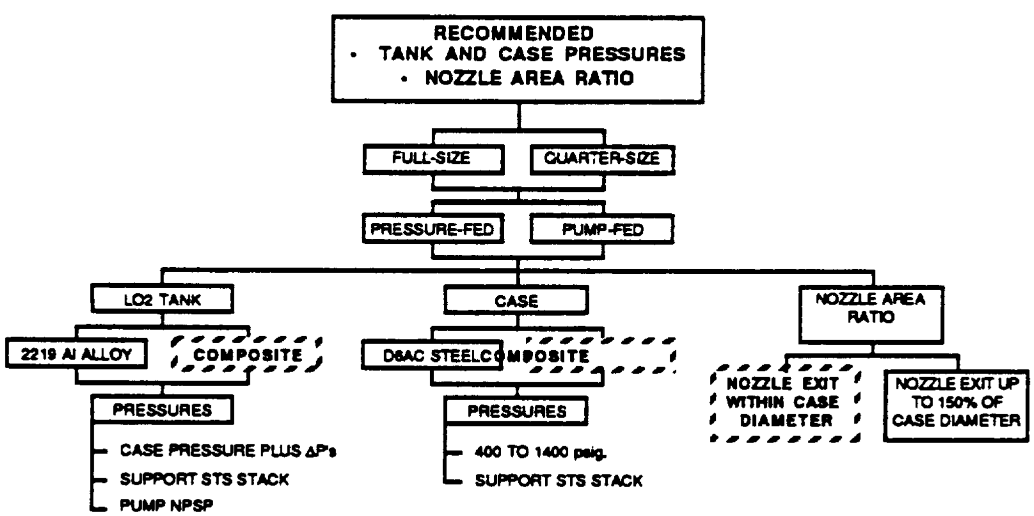


Figure 2-19. The trade tree for the pressures/area ratio analysis and trade study. The point of departure for this study is highlighted.

The nozzle area ratio was also studied as a trade-off between thrust and weight. Ranges of exit diameters from 75% to 150% of case diameter were investigated. After some preliminary analyses it was felt that the increase in performance available from a nozzle greater than the case outer diameter (150 inches) was slight, and optimizing this parameter would not drive out any new technologies. For the studies presented here, a nozzle exit diameter of 150 inches was used. The length of the nozzle however, was allowed to vary as dictated by standard nozzle design practices. These lengths become relevant in the overall booster length and weight calculations.

A composite tank/case combination was the point of departure in this study and is highlighted in the trade tree of Figure 2-19.

2.5.2.2 ISP vs. Pressure. The performance available from a propellant combination is dependent on several conditions present within the combustion chamber, the chamber pressure being of major concern. As the combustion pressure is increased, the specific impulse in lbf thrust per lbm/sec of propellant rises. As stated in the requirements, the propellants used in this evaluation were oxygen and HTPB at an O/F of 2.33. With a combustion efficiency of 92% and a nozzle exit of 150 inches, several Isp data points were calculated using the NASA standard program One Dimensional Equilibrium (ODE).

As shown in Figure 2-20, the computed Isp values can be approximated as a third-order function of pressure. This relationship is necessary for subsequent performance analyses.

2.5.2.3 Boss-to-Boss Lengths. The mass of propellant required has been determined from the total booster impulse given as a design requirement in the SOW. Due to the pressure dependence of the Isp, the propellant mass (and consequently the lengths and volumes of the tank and case) can be presented as functions of pressure. These relationships are depicted in Figures 2-21 and 2-22. Figure 2-23 shows the pressure effects on the entire hybrid motor, including changes in size of the nozzle. The height of the motor from the nozzle exit to the top of the LO2 tank is seen to decrease almost 200 inches (17 feet) as chamber pressure rises from 400 to 1400 psia. in the pressure-fed booster.

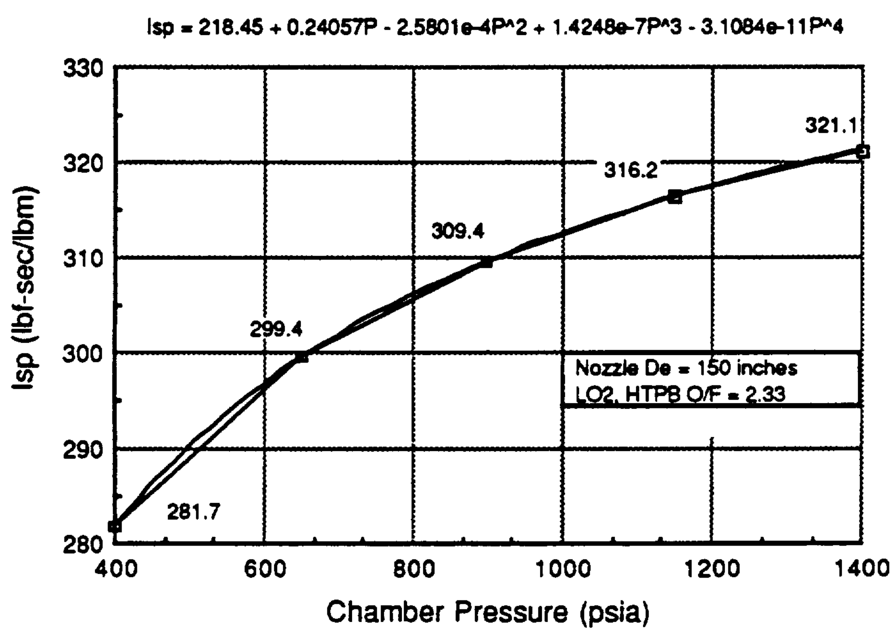


Figure 2-20. The theoretical vacuum specific impulse for the hybrid motor can be characterized as a function of combustion pressure.

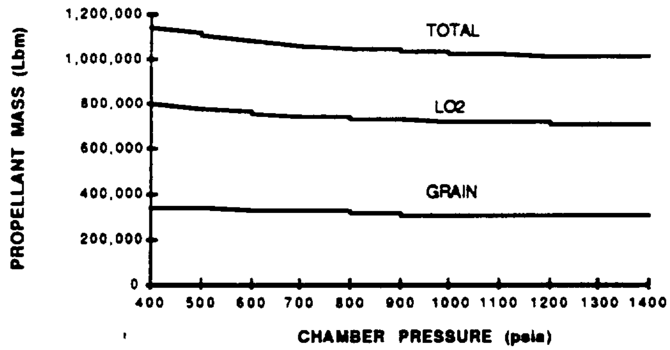


Figure 2-21. The net effect of combustion chamber pressure on propellant weight.

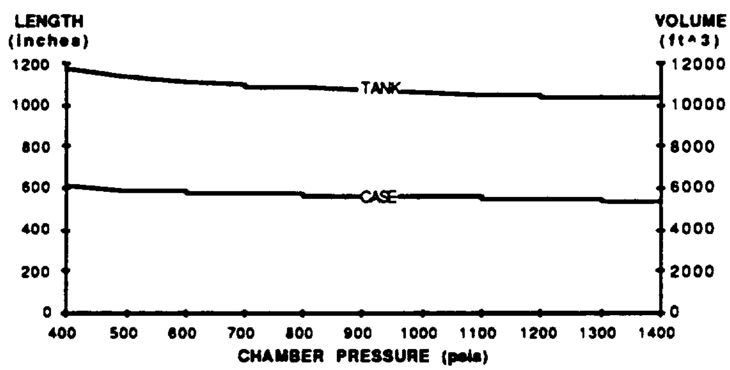


Figure 2-22. As the required amount of propellant decreases (a function of chamber pressure), the total required vehicle size is reduced.

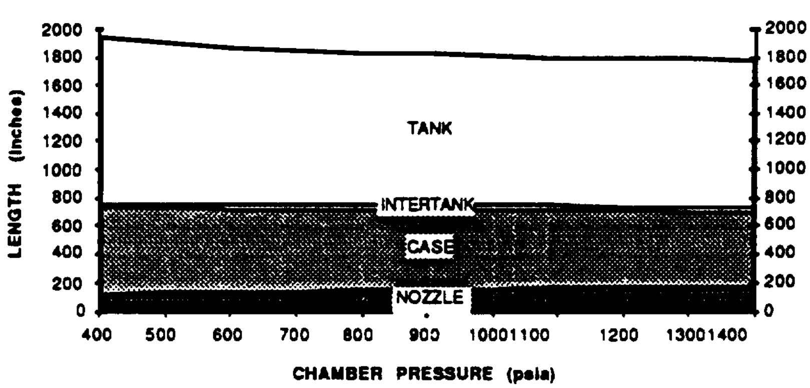


Figure 2-23. Effect of chamber pressure on vehicle length from nozzle exit to top of LO2 tank using HTPB/LO2 propellants at an average O/F of 2.33.

2.5.2.4 Weight, ΔV vs. Pressure. Components such as the inter-tank adaptor and the recovery system are assumed to have constant weight, independent of motor operating pressure. However, for items required to contain the system pressure, the weight rises with the chamber pressure. Since the Isp also tends to increase with pressure, the effects yield a peak performance at some optimum chamber pressure.

As shown in the shaded areas of Figure 2-24, the accumulated booster weight increases from 116,700 lbm to over 208,900 lbm over the pressure range, as read from the axis on the left. Over the same pressure range the ΔV is seen to peak at between 500-600 psia with a value of 9050 ft/sec. This diagram assumes filament-wound graphite/epoxy (GR/EP) for both the LO2 tank and the grain case.

Sizing for the minimum thicknesses of both the tank and case at low chamber pressures is determined by the axial and bending loads on the booster present before and during vehicle launch. The ΔV calculated includes the weights of the existing orbiter and external tank, and assumes a burn time and thrust profile consistent with the ASRM (132 seconds action time, 320 Mlbf total impulse).

Figures 2-25 through 2-27 show the weights and ΔV values for the same vehicle as abovewith the substitution of alternate tank and case materials. The first of these uses a GR/EP tank and a D6AC steel case, the second adds 2219 aluminum for the tank with a GR/EP case, and the third uses 2219 and D6AC. The corresponding ΔVs are plotted using different scales which are appropriate for each configuration.

The last of the weight and ΔV versus pressure charts (Figure 2-28) shows an entirely composite booster running in a pump-fed system configuration. The main

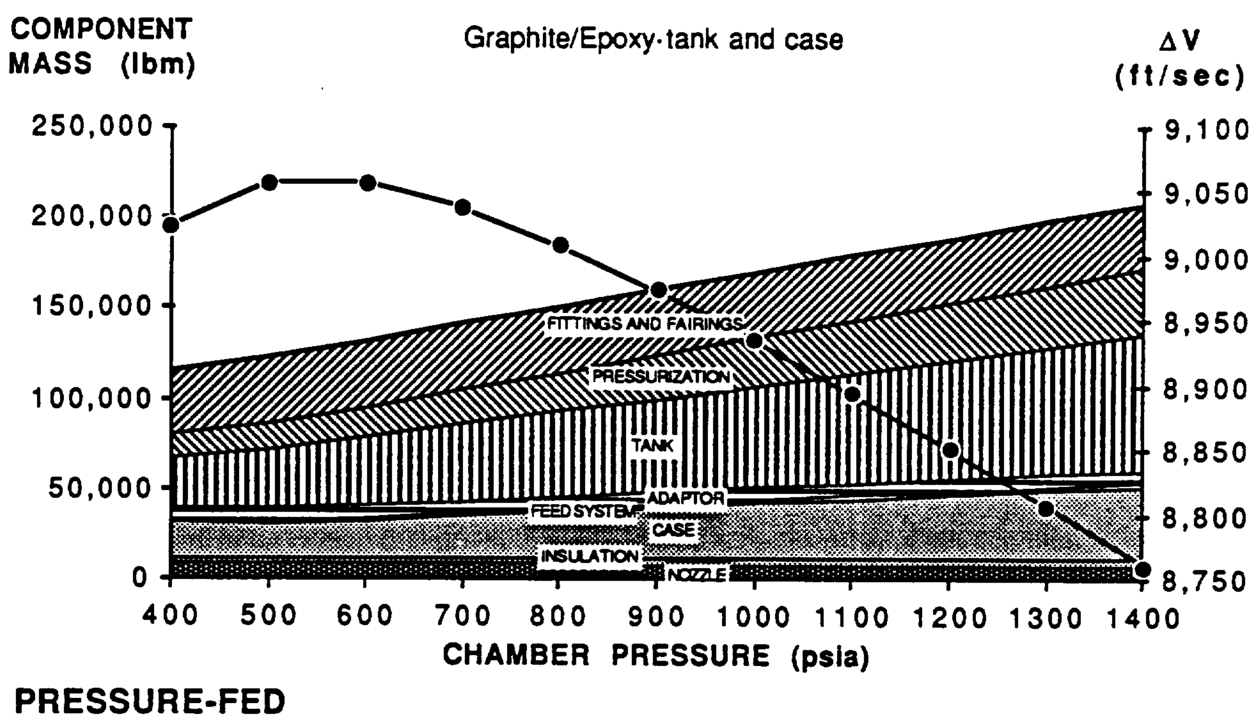


Figure 2-24. Booster total inert weight and performance as functions of pressure. Assumes the use of a composite oxidizer tank and grain case in a pressure-fed configuration.

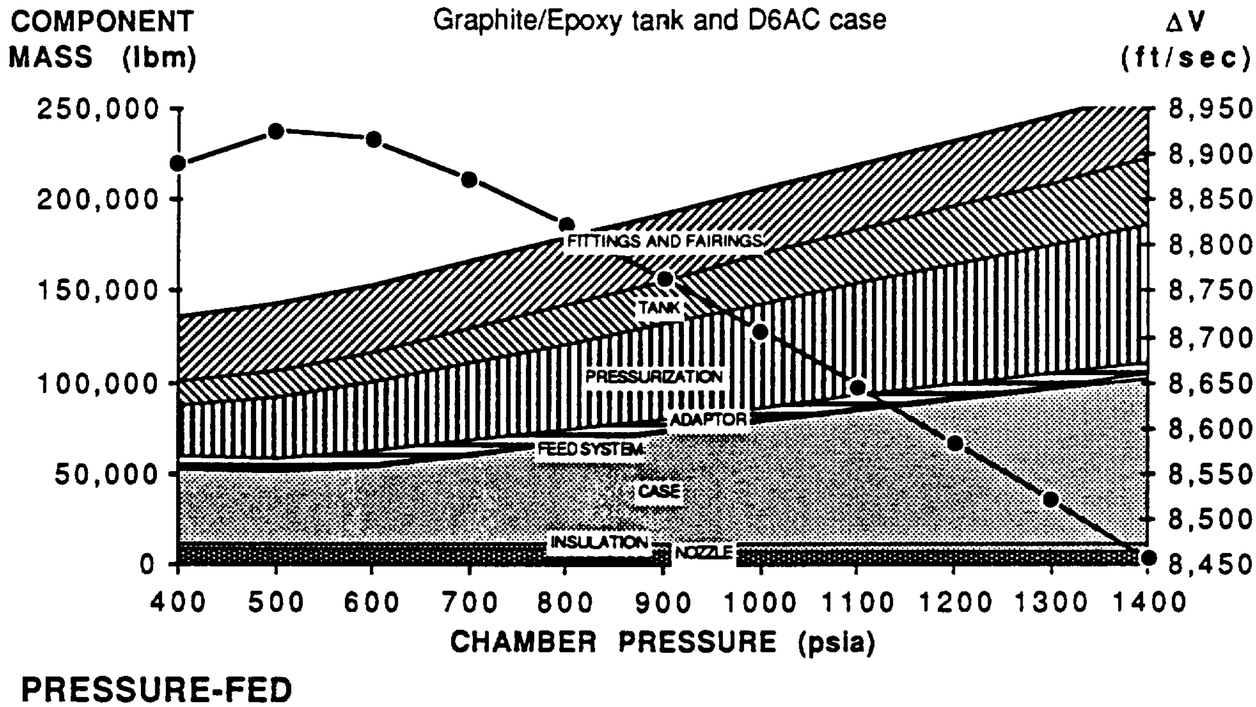


Figure 2-25. Booster total inert weight and performance as functions of pressure. Assumes the use of a composite oxidizer tank and a D6AC steel grain case in a pressure-fed configuration.

difference between this concept and the pressure-fed ones is that the oxidizer tank can be designed for a minimum thickness and is no longer pressure-dependent. The kink in the weight plot at about 600 psia is a result of the minimum case thickness constraint required for structural stability.

2.5.2.5 Comparison of ΔVs vs. Pressure. The ΔV values for each of the tank and case material combinations are plotted for both the pressure- and pump-fed systems in Figure 2-29. The pump-fed systems have consistently higher peak performance at greater pressures than their pressure-fed counterparts. For each of the pump-fed systems, it is interesting to note that the ΔV curves are relatively flat above 800 psia, at which point the weight continues to increase along with the Isp and no net performance is gained.

Figure 2-30 shows the performance of the quarter-ASRM-size boosters. Both boosters are composed of GR/EPP case and tank and are approximately 1,250-inches in length and 90 inches in diameter.

Because the quarter-ASRM-size HRBs are not required to support the weight of a large stack, the case and tank weights are significantly reduced at lower pressures. This effect is more pronounced in the pump-fed configuration since the LO2 tank wall only needs to be thick enough to contain the required 80-psia pump inlet requirements. While the pressure-fed quarter-ASRM-size booster shows only a slight performance difference from the ASRM-size HRB, the pump-fed booster achieves a significantly higher level of performance at lower operating pressures.

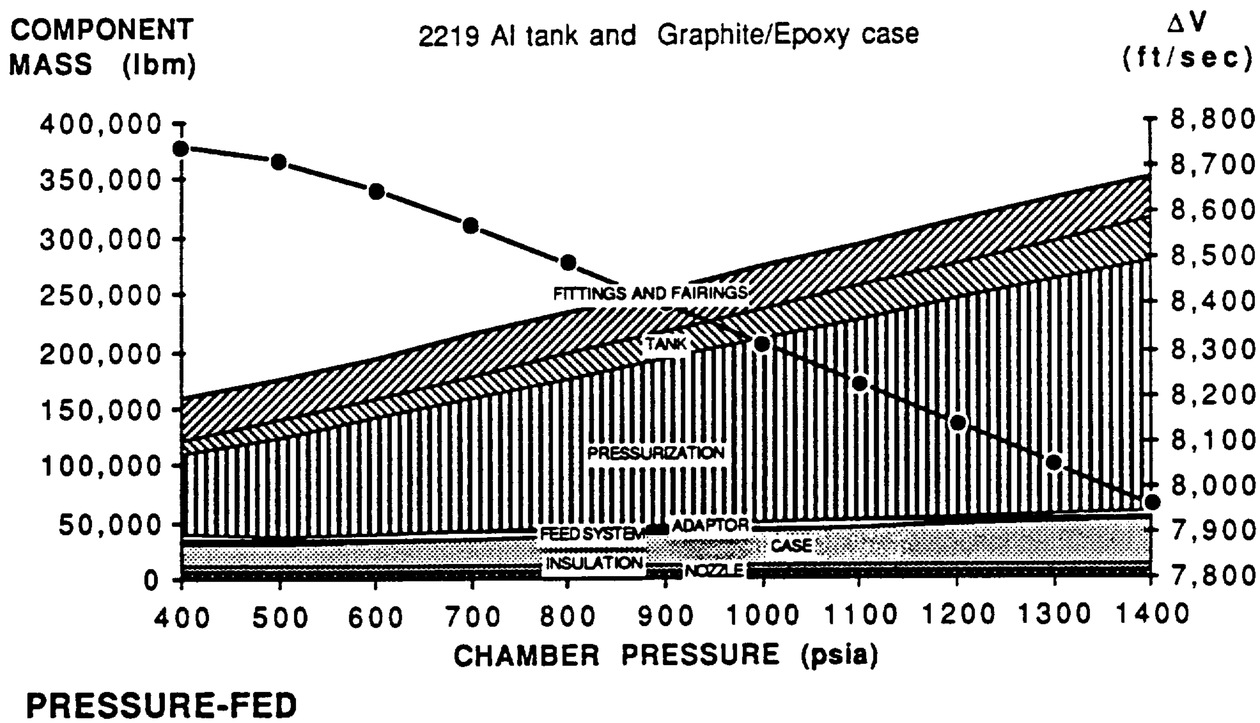


Figure 2-26. Booster total inert weight and performance as functions of pressure. Assumes the use of a 2219 aluminum oxidizer tank and a composite grain case in a pressure-fed configuration.

Although ΔV is not a true measure of actual booster performance, it does provide a comparable baseline value. As a check, a comprehensive computer program was run to find the maximum payload weight orbitable for various booster configurations. When compared to the corresponding ΔV values at staging, a linear relationship was observed. For the preliminary trade studies and analyses, ΔV was deemed sufficient for ranking purposes. In the final booster concept review, the performance was measured with a pounds-to-orbit approach.

Only the graphite/epoxy tank and case booster composition was evaluated in the ranking. The pattern of performance as a function of pressure was similar for each of the composition choices, and it was felt that the GR/EP-GR/EP motor would be representative of the trend. In addition, since the 2219 aluminum tank and the D6AC case are proven technologies, there would be less benefit in exploring the implementation aspects associated with them.

The ranking of the pressures/area ratio study was broken into four separate evaluations: pump-fed, and pressure-fed in both the ASRM and quarter-ASRM-size motors. The two were evaluated using the same criteria but treated as independent concepts and evaluated against each other in the pump-fed versus pressure-fed trade study and analysis. Within the ranking matrix weighted scores were calculated for systems operating at combustion chamber pressures ranging from 400 to 1400 psia.

Scores were based on various appropriate relationships in each of the ranking categories. For normalization of the process, the total of the scores in each row was set to equal 100. Each score was then multiplied

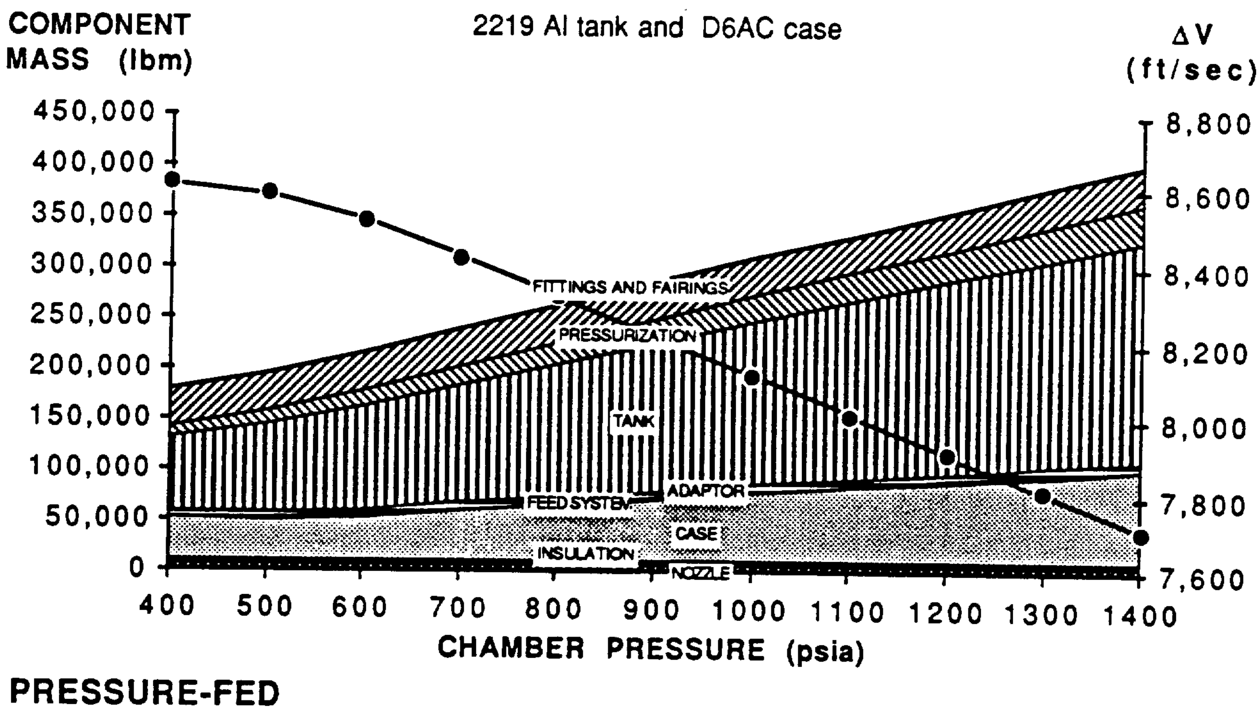


Figure 2-27. Booster total inert weight and performance as functions of pressure. Assumes the use of a 2219 aluminum oxidizer tank and a D6AC steel grain case in a pressure-fed configuration.

by the weighting factor given to each category. The sum of the weighted score columns determined the optimum operating pressure based on the requirements.

Figure 2-31 is a ranking matrix for the pressure-fed, ASRM-size HRB, rating each of the criteria versus average motor chamber pressure. The scores in the flight safety criteria were driven by potential vehicle damage in the case of tank rupture. It was assumed that at higher operating pressures, tank structural failure would impart greater damage to the core vehicle and remaining propulsion systems. Thus lower working pressures would result in a safer system and, consequently, received higher scores.

Booster system reliability was calculated as the sum of the reliabilities of the components of which it is made. Within the range of pressures studied in this analysis, component similarity and scaling relationships provided identical reliability scores.

The analysis of non-recurring cost yielded a desirable trend toward lower chamber pressures. The major driver in this study proved to be the extreme weight of a higher pressure oxidizer tank and case. The required tooling for manufacture and transportation of the heavier components was significant. Additionally, the investment required to develop high-pressure fluid components affected the resulting scores.

The assessment of recurring cost was greatly affected by the initial program assumption of booster recovery and refurbishment. A recovery system identical to that employed on the SRB was analyzed for use with the HRB. Considering the water impact loading of the somewhat brittle filament-wound motor, more durable case and tank construction

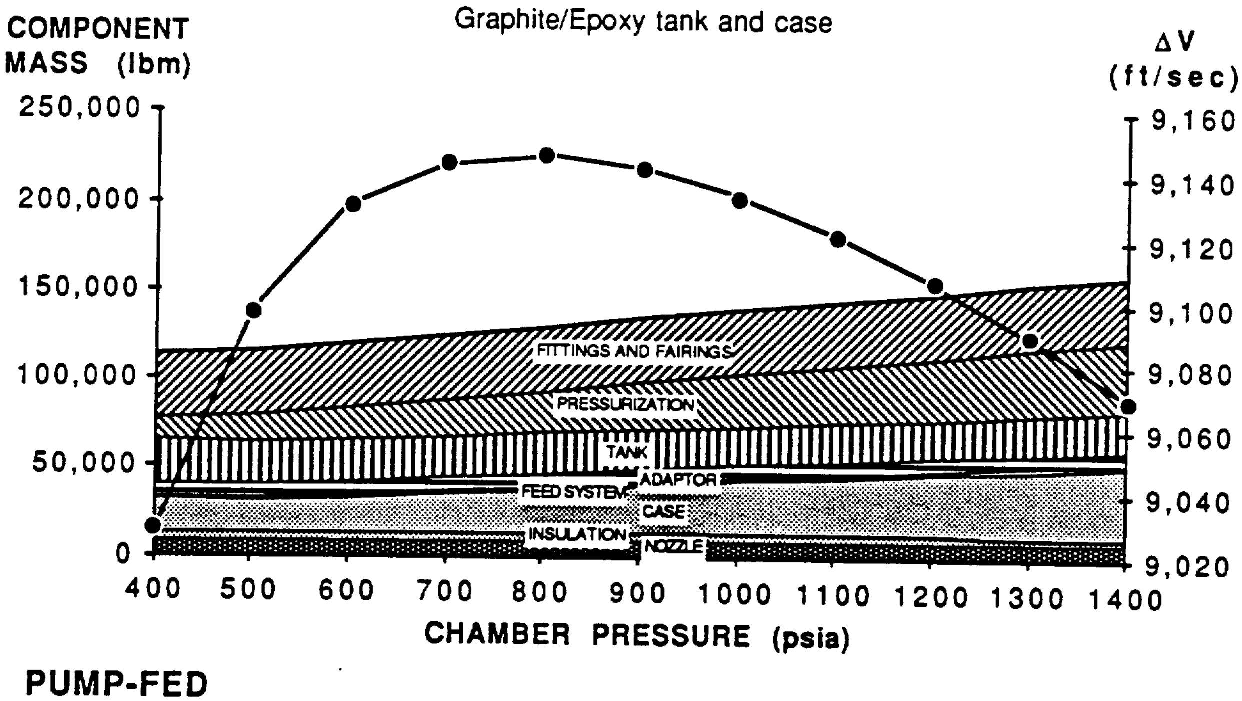


Figure 2-28. Booster total inert weight and performance as functions of pressure. Assumes the use of a composite oxidizer tank and grain case in a pump-fed configuration.

would allow greater resistance to damage. The increased reusability achieved quickly overshadows the relatively low cost of additional composite wrapping and the higher pressure fluid components.

In the performance category, the sole measure of evaluation was the vehicle velocity at booster staging (ΔV). Although the numbers were very close in value, a large emphasis was placed on the differences observed: it was calculated that for each foot-per-second of ΔV over 8500, an additional 34 lbm of payload could be put into orbit aboard the STS vehicle. The scores recorded were numerically scaled from the ΔV versus pressure plot from Figure 2-29.

In the determination of operational merit as a function of pressure, several maintenance, logistic, and ground system issues were considered. In every case, a lower chamber pressure provided a more favorable system. Consequently, the low scores at the higher pressures reflect the increased overhead and equipment for booster preparation.

The results of the ranking of average operating chamber pressure for the pressure-fed ASRM-size HRB was a tie between 400 and 600 psia. It was assumed that the optimum would be between these at 500 psia.

The ranking of the ASRM-size HRB with a pump-fed oxidizer supply system is shown in Figure 2-32. Much of the scoring versus pressure came out the same as in the pressure-fed ranking. The major differences were in performance and operational considerations. The ΔV performance in Figure 2-29 peaks at higher pressures than in a similar pressure-fed system

Operationally, the pump-fed system was found to benefit greatly from lower operating pressures. The increased calibration and testing required for higher pressure, high-performance pumping machinery was the driving consideration. The resulting totals of the pump-fed, ASRM-size HRB chamber pressure was a tie between 600 and 800 psia. Seven hundred was assumed to be the optimum chamber pressure.

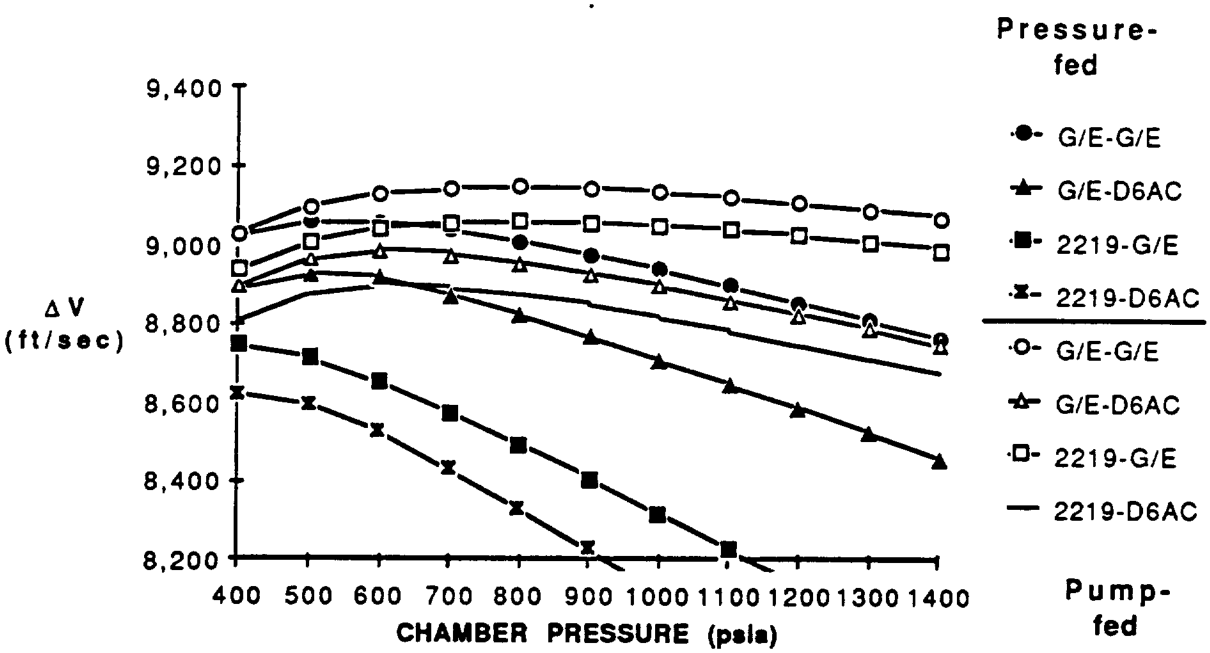


Figure 2-29. A summary of booster performance for the various material combinations and feed configurations.

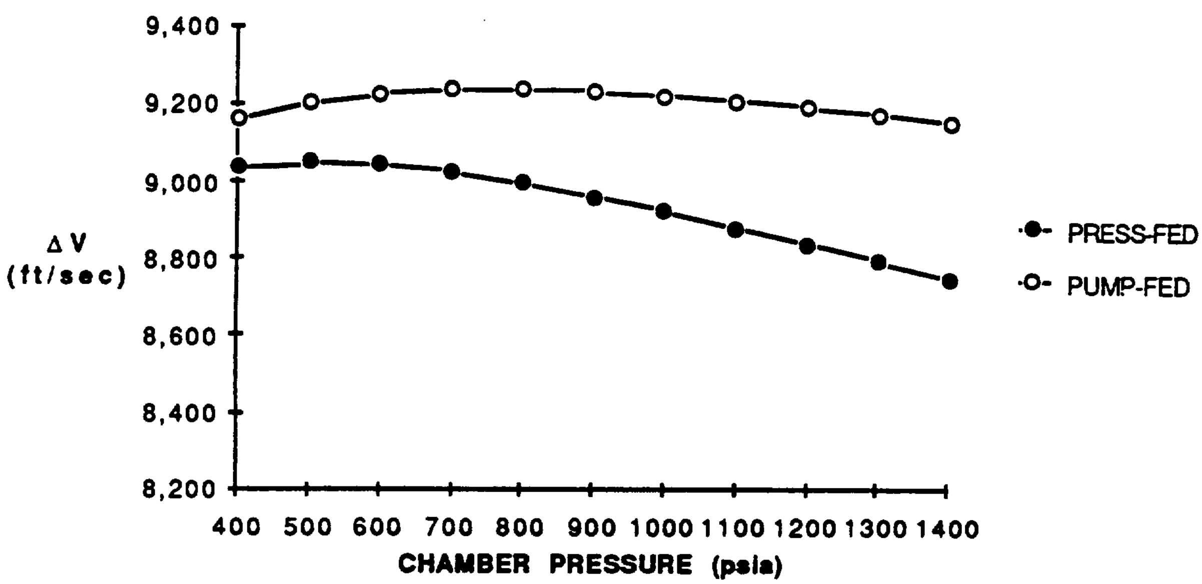


Figure 2-30. The performance of the quarter-ASRM-size booster in the pump-fed and pressure-fed configurations. Both instances use composite tanks and cases.

CRITERIA (RATING FACTOR) \ CONCEPT	CHAMBER PRESSURE							
	400 psia		600 psia		800 psia		1000 psia	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
<u>FLIGHT SAFETY & RELIABILITY</u> • FLIGHT SAFETY (0.20)	31	6	27	5	23	5	19	4
• RELIABILITY (0.20)	25	5	25	5	25	5	25	5
<u>LIFE CYCLE COST</u> • NON RECURRING (0.15)	34	5	28	4	22	3	16	3
• RECURRING (0.15)	17	3	23	3	27	4	31	5
<u>PERFORMANCE</u> • DELTA VELOCITY (0.20)	30	6	45	9	20	4	5	1
<u>OPERATIONAL CONSIDERATIONS</u> • LAUNCH SITE (0.10)	40	4	30	3	20	2	10	1
TOTALS		29		29		23		19
RANK		1t		1t		3		4

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST (Σ CONCEPTS = 100)

Figure 2-31. Ranking of the ASRM-size pressure-fed booster average chamber pressures; 500 psia was judged optimum.

CRITERIA (RATING FACTOR) \ CONCEPT	CHAMBER PRESSURE							
	400 psia		600 psia		800 psia		1000 psia	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
<u>FLIGHT SAFETY & RELIABILITY</u> • FLIGHT SAFETY (0.20)	31	6	27	5	23	5	19	4
• RELIABILITY (0.20)	25	5	25	5	25	5	25	5
<u>LIFE CYCLE COST</u> • NON RECURRING (0.15)	34	5	28	4	22	3	16	3
• RECURRING (0.15)	17	3	23	3	27	4	31	5
<u>PERFORMANCE</u> • DELTA VELOCITY (0.20)	4	1	30	6	34	7	32	6
<u>OPERATIONAL CONSIDERATIONS</u> • LAUNCH SITE (0.10)	31	3	27	3	23	2	19	2
TOTALS		23		26		26		25
RANK		4		1t		1t		3

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST (Σ CONCEPTS = 100)

Figure 2-32. Ranking of the ASRM-size pump-fed booster average chamber pressures; 700 psia was judged optimum.

The analyses of the quarter-ASRM-size HRB yielded results similar to those of the ASRM-size HRB studies (see Figure 2-33). The major difference between the two sizes of pressure-fed boosters was seen in the operational considerations. For the smaller-sized booster, the additional material required for structural integrity at higher operating pressures had less of an impact on launch site processing. The trend of increasing ranking scores was still towards lower pressures.

The scores in the performance criteria had the same values as those for the pressure-fed ASRM-size booster, although the ΔV values were significantly different. As was shown in Figures 2-29 and 2-30, the ΔV curves were unique for each of the different sizes and pressurization schemes. The scaled scores, nonetheless, turned out the same. The highest-ranked pressure level for the pressure-fed HRB was 600 psia.

Figure 2-34 shows how the ranking process resulted for the pump-fed version of the quarter-ASRM-size HRB. As with the pressure-fed systems, the ranking matrix was similar to that of the analogous ASRM-size HRB. The difference is in the performance criteria, which, as presented previously in Figure 2-32, displays a much different ΔV versus pressure progression than the ASRM-size booster. The desired chamber pressure for this HRB was found to lie between 600 and 800 psia (700 was assumed to be optimum).

In summary, lower average operating pressure has the advantages of greater safety in case of rupture, and the reduced cost of tooling, transportation, and processing. Higher pressures offer increased booster ruggedness (increased reusability), and higher performance.

CRITERIA (RATING FACTOR) \ CONCEPT	CHAMBER PRESSURE							
	400 psia		600 psia		800 psia		1000 psia	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
FLIGHT SAFETY & RELIABILITY								
• FLIGHT SAFETY (0.20)	31	6	27	5	23	5	19	4
• RELIABILITY (0.20)	25	5	25	5	25	5	25	5
LIFE CYCLE COST								
• NON RECURRING (0.15)	34	5	28	4	22	3	16	3
• RECURRING (0.15)	17	3	23	3	27	4	31	5
PERFORMANCE								
• DELTA VELOCITY (0.20)	30	6	45	9	20	4	5	1
OPERATIONAL CONSIDERATIONS								
• LAUNCH SITE (0.10)	32	3	26	3	24	2	18	2
TOTALS		28		29		23		20
RANK		2		1		3		4

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST (Σ CONCEPTS = 100)

Figure 2-33. Ranking of the quarter-ASRM-size pressure-fed booster average chamber pressures: 600 psia was judged optimum.

CRITERIA (RATING FACTOR) \ CONCEPT	CHAMBER PRESSURE							
	400 psia		600 psia		800 psia		1000 psia	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
<u>FLIGHT SAFETY & RELIABILITY</u> • FLIGHT SAFETY (0.20)	31	6	27	5	23	5	19	4
• RELIABILITY (0.20)	25	5	25	5	25	5	25	5
<u>LIFE CYCLE COST</u> • NON RECURRING (0.15)	34	5	28	4	22	3	16	3
• RECURRING (0.15)	17	3	23	3	27	4	31	5
<u>PERFORMANCE</u> • DELTA VELOCITY (0.20)	10	2	30	6	35	7	25	5
<u>OPERATIONAL CONSIDERATIONS</u> • LAUNCH SITE (0.10)	31	3	27	3	23	2	19	2
TOTALS		24		26		26		24
RANK		2		1t		1t		2

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST (Σ CONCEPTS = 100)

Figure 2-34. Ranking of the quarter-ASRM-size pump-fed booster average chamber pressure; 700 psia was judged optimum.

2.5.3 PRESSURIZATION SYSTEM

2.5.3.1 Objectives. The objective of this system analysis and trade study was to recommend a tank pressurization concept to pressure-feed LO2 for hybrid propulsion combustion. The information was also created in parametric form to support other system analyses and trade studies.

2.5.3.2 Assumptions. It was assumed that the liquid oxygen tank had an empty volume of 10,000 cubic feet, which is typical of the ASRM-size booster. The pressurant would be non-reactant with gaseous and liquid oxygen. It would flow into the tank at a temperature of 800 degrees rankine to provide reasonable allowable stress values for the aluminum or composite tanks. The maximum required pressure was set at 1,000 psia with pressure versus time varying to match the required thrust curve.

Pumps and turbines were not considered the most attractive pressure-fed oxidizer systems because they include rotating machinery. A factor of safety of 1.5 was applied to the pressurant storable bottles.

Figure 2-35 shows the tank ullage versus burn time. Figure 2-36 shows the flow characteristics for pressurized tank systems. It uses the isentropic expansion of the pressurant to provide pressure during the final portion of LO2 flow.

The critical time to determine the pressurant cutoff point occurs at 119.1 seconds during the isentropic expansion. The resultant cutoff point occurs at 74.9 sec at a tank volume of 6250 cubic feet.

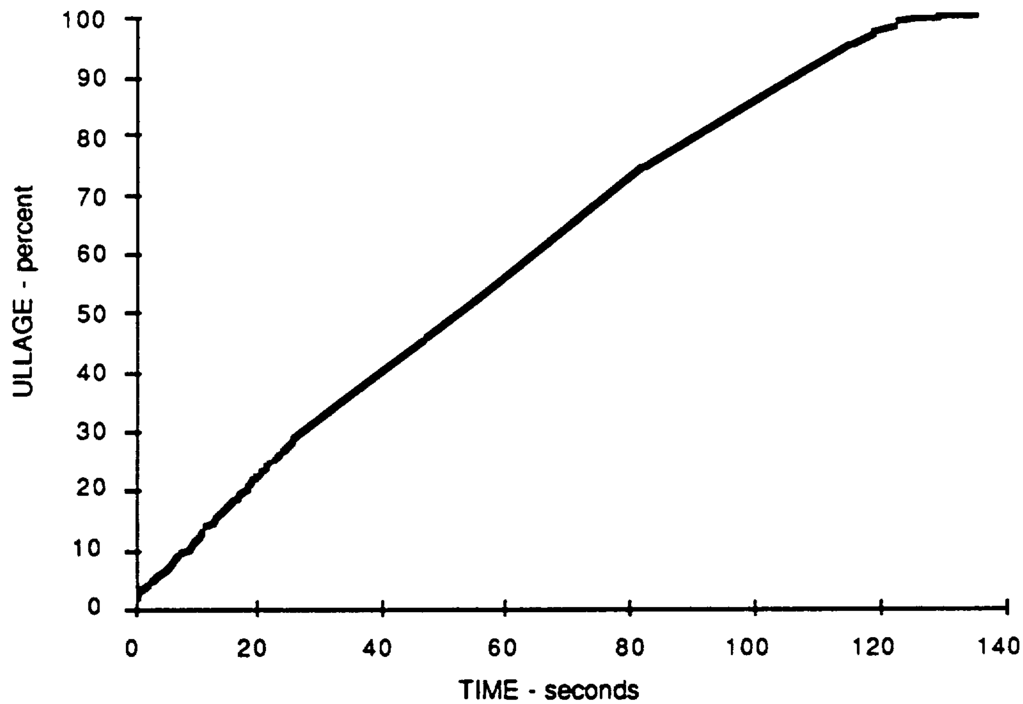


Figure 2-35. Ullage vs. time for specified duty cycle.

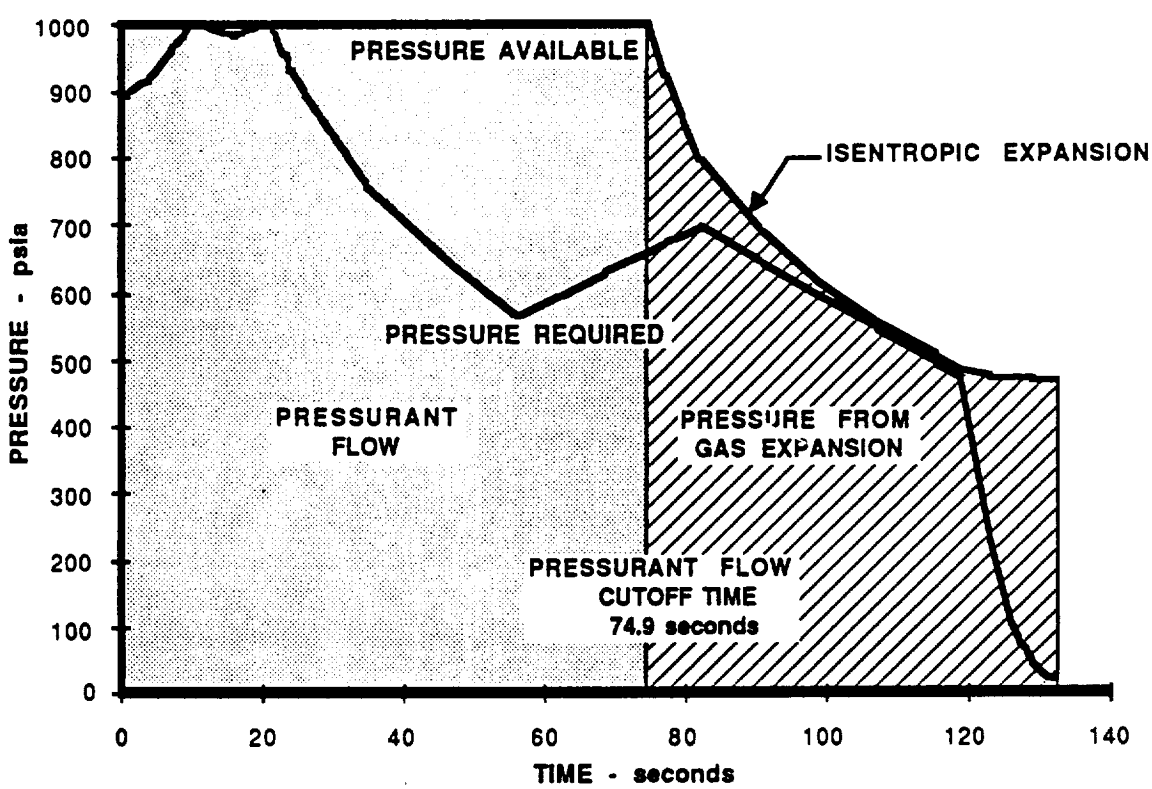


Figure 2-36. Pressurant flows for 58% of the total burn time.

2.5.3.3 Trade Tree. The trade tree shown in Figure 2-37 portrays the pressurization systems considered. Crossed-off entries on the tree indicate concepts which, after preliminary review, did not merit further study based on previous pressurization studies funded by MSFC.

A previous GDSS contracted study for the pressure-fed liquid rocket booster selected high-pressure storage of Tridyne at low temperature as the preferred pressurization system. This concept is highlighted to denote its acceptance as the initial baseline.

2.5.3.4 Analyses and Studies. The systems studied in detail were the Tridyne cascade, the motor/case heated helium cascade and the gas generator heated helium cascade pressurization systems.

The Trydine cascade system is shown in Figure 2-38. It contains cool Tridyne gas in a secondary bottle, which reacts when flowing through a catalyst bed and discharges as heated gas into the primary bottle.

The Tridyne in the primary bottles functions similarly to discharge heated gas into the LO2 tank ullage. By preloading a precalculated percentage of oxygen and hydrogen mixed in the helium, the resultant temperature of the gas flowing into the LO2 remains constant.

A parallel flow of gas around the catalyst bed is not required for temperature control. Pressure control is required to reduce the bottle storage pressure to the desired ullage pressure.

Moisture inherent in the gas discharge into the LO2 tank is acceptable. The amount discharged is between 6 and 12% of the allowable when purchasing liquid oxygen for the Atlas Space Launch Vehicle. Of the total mass of moisture entering the tank, only a small percentage would float/dissolve in the liquid oxygen as fine granules.

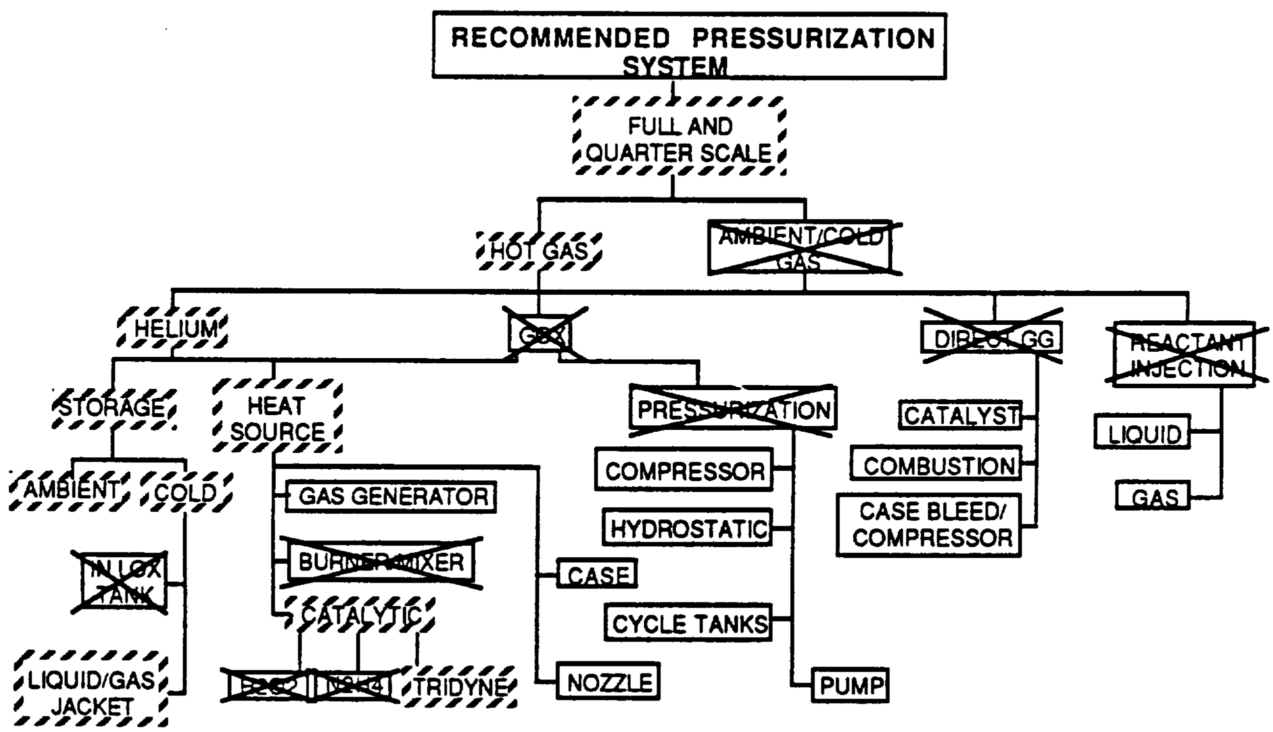


Figure 2-37. All viable pressurization systems were considered.

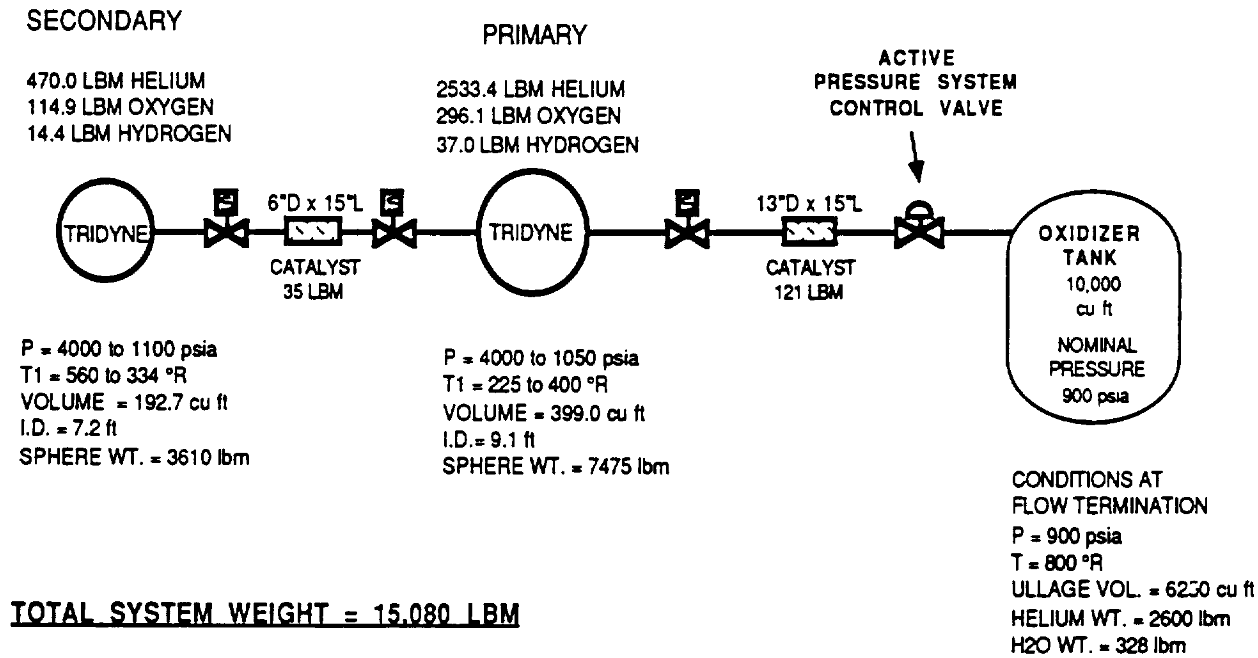


Figure 2-38. Tridyne cascade pressurization system.

The temperature rise occurring when Tridyne flows through the catalyst bed increases as the concentration of the reactant increases. This effect is shown in Figure 2-39, which also notes the presented point design. The ratio of hydrogen-to-oxygen concentration is 2:1 by volume. Experimental results with concentrations at or near 2:1 were obtained using mass spectrometry and temperature measurement have been shown to verify the computations.

There was good agreement between the stability test results presented by NASA, TRW, and Rocketdyne. These results as well as the planned operating range are shown in Figure 2-40. The location of the operating range is well separated from the ignitable and uncertain regions, indicating a very good margin of safety.

Stratification of the mixed gasses is not possible according to computations based on the diffusivity rates of the gasses. The rates completely overpower the tendency of stratification under gravitation and acceleration forces.

Figure 2-41 presents the test results from NASA, Rocketdyne, and National Bureau of Standards defining the detonation zone boundaries. The proposed operating area is well separated from the detonatability zone.

The motor case/nozzle helium cascade pressurization system is shown in Figure 2-42. Helium stored at ambient temperature is heated as it flows through a case/nozzle heat exchanger prior to discharging into the primary sphere. The primary helium is also heated in a like manner prior to pressurizing the LO2 tank ullage. The cascade flow reduces system size and weight.

The gas generator heated helium cascade pressurization system in Figure 2-43 combusts LO2 from the main tank and RP-1 from an auxiliary tank. The gas generator exhaust heats the cascading helium to reduce its system weight and size, and the exhaust is discharged overboard.

2.5.3.5 Ranking. The three cascade systems were ranked as shown in Figure 2-44. The Tridyne system was preferred.
The nozzle heated helium system is a totally inert system. It was preferred when evaluated for flight safety. The gas generator was the least preferred as it requires combustion and another fuel system.

The Tridyne system was the most reliable as documented by Figures 2-45 through 2-48. It also had the lowest life cycle costs due to the simplistic nature of the system with short lines, no additional support systems, and the ability to develop and manufacture the system independent of the other booster systems.

The booster performance is best when installing the Tridyne system. Its weight is 16% lower, shown by Figure 2-49, than the other two considered systems.

The Tridyne system is also preferred based on operational considerations, again due to a minimum number of components to install and check-out prior to launch.

2.5.4 PRESSURE-FED VERSUS PUMP-FED LIQUID OXYGEN

2.5.4.1 Objectives. The objectives of this system analysis and trade study are to: 1) recommend the preferred pump-fed system to supply pressurized oxygen to the grain, and 2) compare the preferred pump-fed system with the previously selected pressure-fed system and select either as the baseline hybrid propulsion technology oxygen pressurization system.

2.5.4.2 Assumptions. The principal assumptions implemented for these analyses and trades were previously assumed for the pressurization system analyses and trades. They are that the liquid oxygen tank had an empty volume of 10,000 cubic feet, the maximum pressure required is 1,000 psia, and the ullage pressurant to provide net pump suction pressure has an inlet temperature of 800 degrees rankine.

The pump-fed liquid oxygen system is active when motor thrust is desired as is shown in Figure 2-50. It must operate to discharge the oxygen at the desired pressure. The tailoff at the end of the required thrust-time curve,

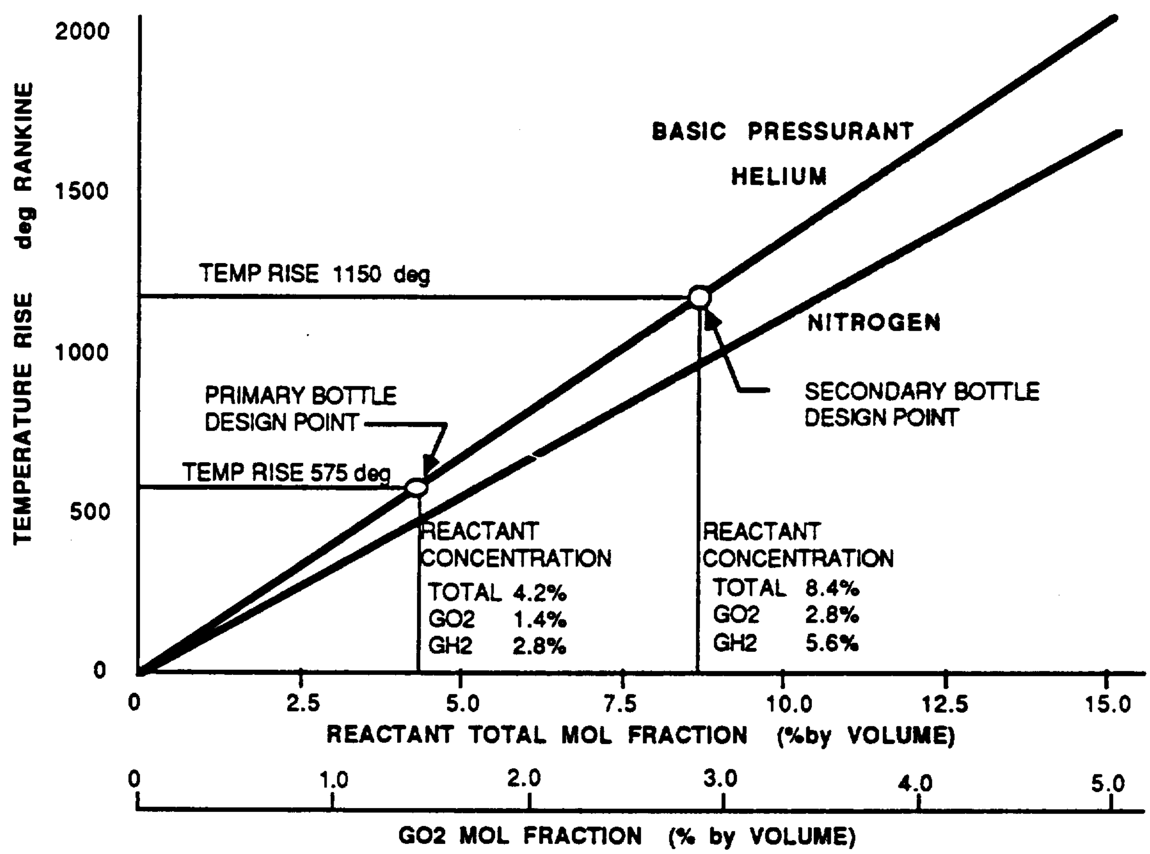


Figure 2-39. Tridyne performance.

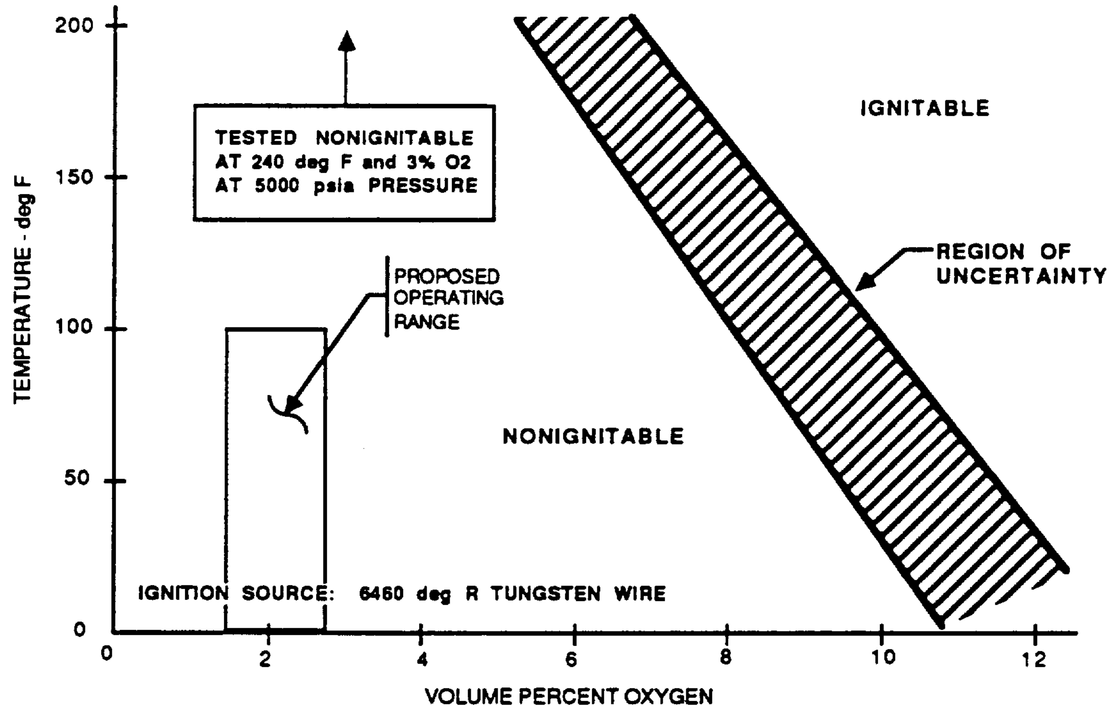


Figure 2-40. Concentration/temperature stability for helium-based Tridyne at 2000 psia.

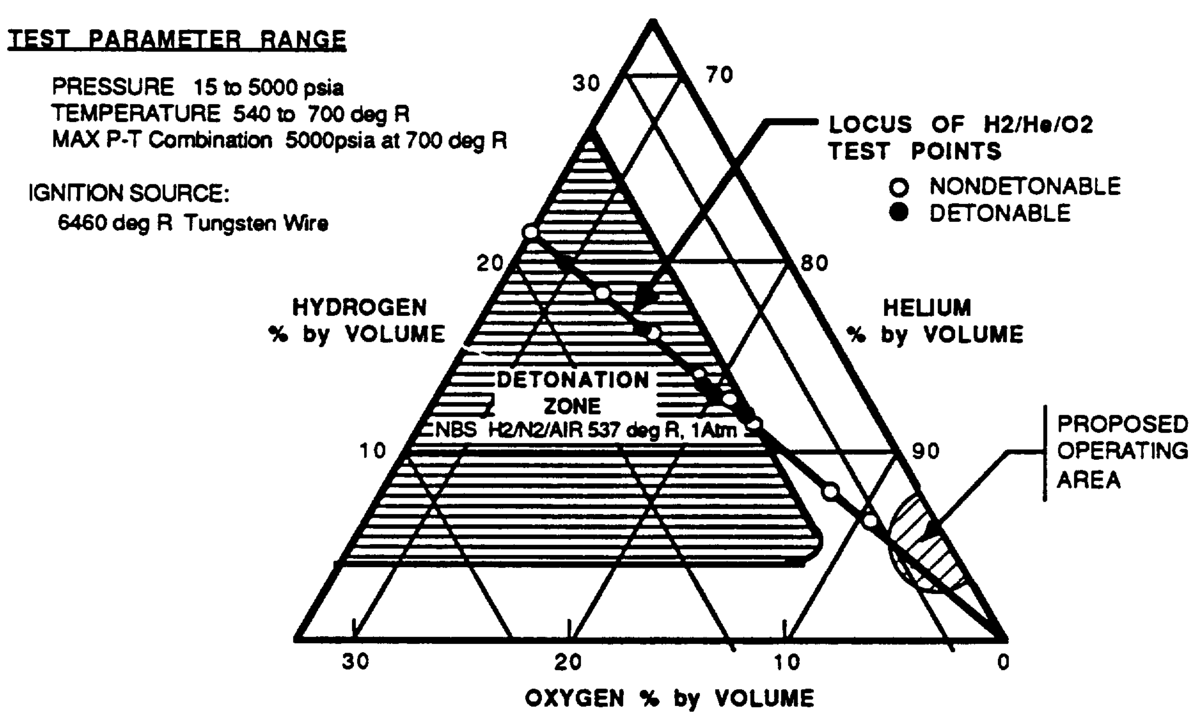


Figure 2-41. Helium-based Tridyne detonatability range.

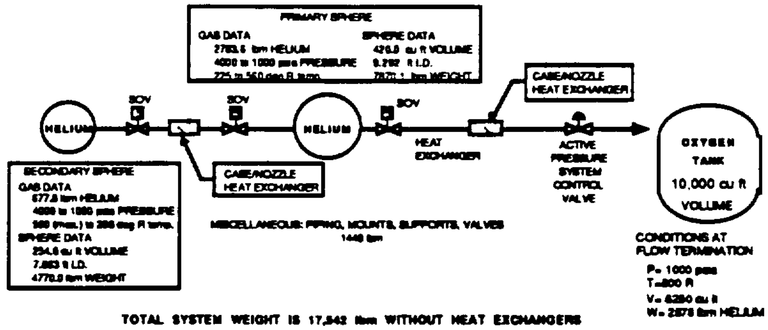


Figure 2-42. Motor case/nozzle heated helium cascade pressurization system.

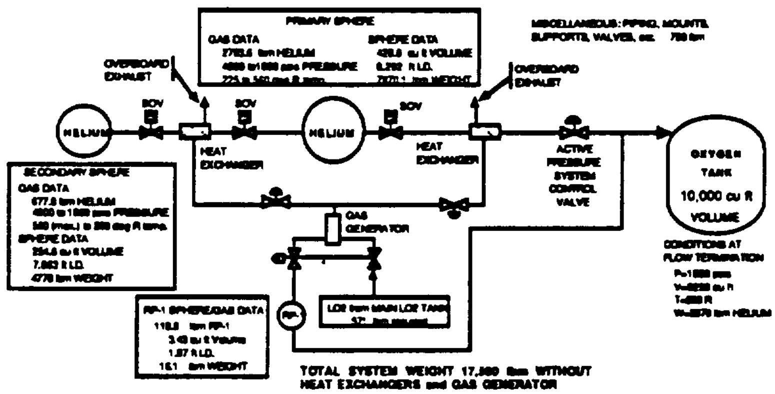


Figure 2-43. Gas generator heated helium cascade pressurization system.

CRITERIA (RATING FACTOR) \ CONCEPT	TRIDYNE SCORE*	TRIDYNE WEIGHTED SCORE	NOZZLE HEATED HELIUM SCORE*	NOZZLE HEATED HELIUM WEIGHTED SCORE	GAS GENERATOR HEATED HELIUM SCORE*	GAS GENERATOR HEATED HELIUM WEIGHTED SCORE
FLIGHT SAFETY & RELIABILITY						
• FLIGHT SAFETY (0.20)	40	8	60	12	0	0
• RELIABILITY (0.20)	54	11	46	9	0	0
LIFE CYCLE COST						
• NON RECURRING (0.15)	55	8	45	7	0	0
• RECURRING (0.15)	70	10	30	5	0	0
PERFORMANCE						
• DELTA VELOCITY (0.20)	50	10	25	5	25	5
OPERATIONAL CONSIDERATIONS						
• LAUNCH SITE (0.10)	70	7	30	3	0	0
TOTALS		54		41		5
RANK		1		2		3

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST

Figure 2-44. The Tridyne cascade pressurization system is preferred.

typical of a solid rocket motor termination, is replaced with a tailoff more typical of a liquid rocket motor. Liquid oxygen flow is maintained at 57% flow to 122.9 seconds instead of the requested flow decrease to 134.1 seconds. This acceptable mode of operation reduced the throttling requirements of the turbo-pump to a manageable value while providing the identical total impulse.

2.5.4.3 Trade Tree. Figure 2-51 presents the turbo-pump options evaluated to select the preferred concept, which was then compared with the selected pressure-fed concept. The five concepts evaluated to drive the turbo-pump were LO2/RP-1, LO2/hybrid, motor heat, N2H4, and Tridyne. Those concepts crossed out in the trade tree were considered but not studied because they lacked promise based on information generated by programs exploring similar systems.

Figure 2-52 presents the LO2/RP-1 gas generator-driven turbo-pump system. RP-1 fuel is expelled from its tank by stored helium at ambient temperature. The RP-1 flows to the gas generator where it combusts with a partial flow of LO2 from the LO2 tank. After

TRIDYNE CASCADE PRESSURIZATION SYSTEM

UNCOMMON COMPONENTS

ITEM 1 CATALYST (2)

ITEM 2 TRIDYNE TANKS (2)

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	CATALYST INTERNAL/EXTERNAL LEAKAGE DUE TO MECHANICAL FAILURE RESULTS IN LOSS OF TRIDYNE PRESSURE APPROX 15 FT OF TUBING	LOSS OF TRIDYNE PRESSURE LOST OR DEGRADED MISSION	3	1/2	6
2	EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF TRIDYNE	LOSS OF TRIDYNE PRESSURE LOST OR DEGRADED MISSION	3	1/2	6
				TOTAL =	12

Figure 2-45. Tridyne cascade pressurization system criticality.

MOTOR CASE/NOZZLE HEATED HELIUM CASCADE PRESSURIZATION SYSTEM

UNCOMMON COMPONENTS

ITEM 1 HEAT EXCHANGER (2)

ITEM 2 HELIUM TANKS (2)

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	HEAT EXCHANGER EXTERNAL/INTERNAL LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN LOSS OF HELIUM PRESSURE APPROX 200 FT OF TUBING	LOSS OF HEAT TRANSFER AND HELIUM PRESSURE LOST OR DEGRADED MISSION	3	[illegible]	12
2	EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF HELIUM PRESSURE	LOSS OF HELIUM PRESSURE LOSS OF HELIUM LOST OR DEGRADED MISSION	3	1/2	6
				TOTAL =	18

Figure 2-46. Motor case/nozzle heated helium cascade pressurization system criticality.

GAS GENERATOR HEATED HELIUM CASCADE PRESSURIZATION SYSTEM

UNCOMMON COMPONENTS

ITEM 1 HEAT EXCHANGERS (2)
ITEM 2 GAS GENERATOR
ITEM 3 CONTROL VALVES (2)
ITEM 4 EXHAUST VALVES (2)
ITEM 5 RP-1 TANK

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	HEAT EXCHANGER EXTERNAL/INTERNAL LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN LOSS OF HELIUM PRESSURE APPROX 200 FT OF TUBING	LOSS OF HEAT TRANSFER AND HELIUM PRESSURE LOST OR DEGRADED MISSION	3	1/2	6
2	GG EXTERNAL LEAKAGE OR COMBUSTION INSTABILITY	POTENTIAL OF RICH BURN RESULT IN GG BURN THROUGH/ EXPLOSION	4	2	8
3	CNTL VALVES GIVES WRONG MIX TO GG	AS ABOVE	4	[illegible]	16
4	EXHAUST VALVE GIVES WRONG FLOW RATE	AS ABOVE	4	[illegible]	16
5	RP-1 SEALS EXTANT LEAKAGE OF RP-1	LOSS OF RP-1 DELIVERY LOST OR DEGRADED MISSION	3	2	6
				TOTAL =	52

Figure 2-47. Gas generator heated helium cascade pressurization system criticality.

CONCLUSIONS
FOR THE
PRESSURE FED PRESSURIZATION SYSTEM COMPARISON

THE GAS GENERATOR OPTION INTRODUCES ADDITIONAL HARDWARE THAT HAVE POTENTIALLY CATASTROPHIC FAILURE MODE EFFECTS

THE TRIDYNE OPTION HAS AN OVERALL LOWER RELATIVE CRITICALITY IN TERMS OF POTENTIAL FAILURE MODE EFFECTS AND PROBABILITIES OF OCCURRENCE

THE RELIABILITY RATING FACTORS (RRF) ARE AS FOLLOWS

	CR1 = 12 TRIDYNE CASCADE SYSTEM	CR2 = 18 MOTOR CASE/NOZZLE	CR3 = 52 GAS GENERATOR
RRFs	54	46	0

Figure 2-48. Conclusions for the pressure-fed pressurization system comparison.

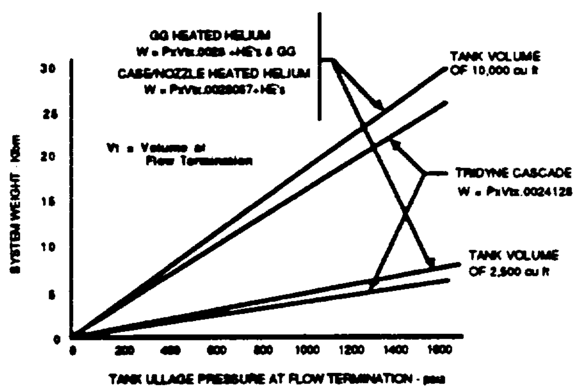


Figure 2-49. Pressurization systems weight.

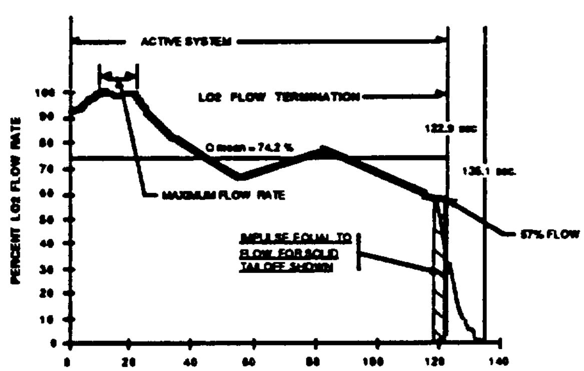


Figure 2-50. The pump-fed system simulates the flow requirements for a thrust profile of a solid rocket motor.

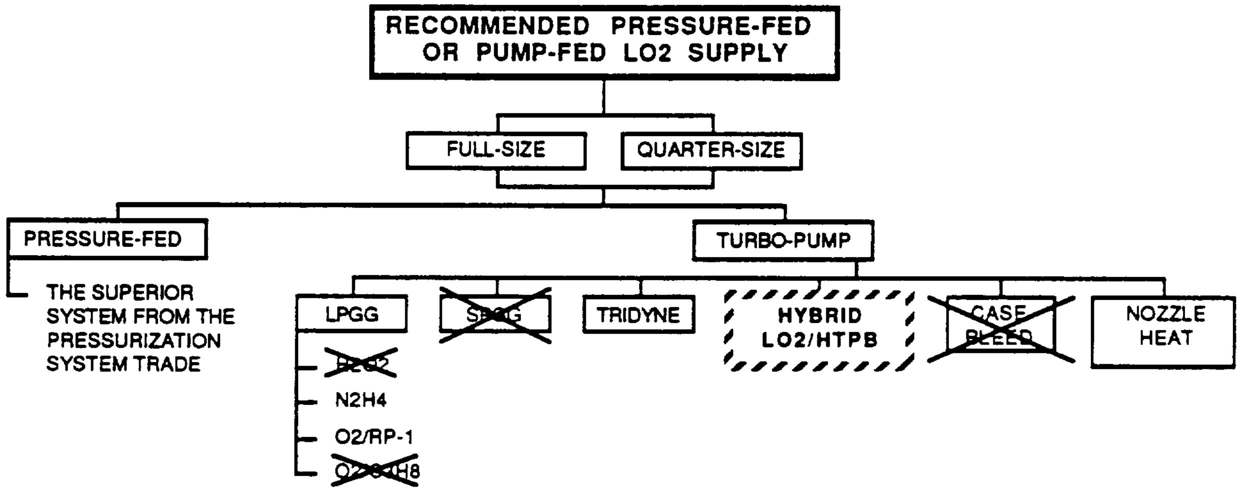


Figure 2-51. Pressure-fed verses pump-fed trade tree.

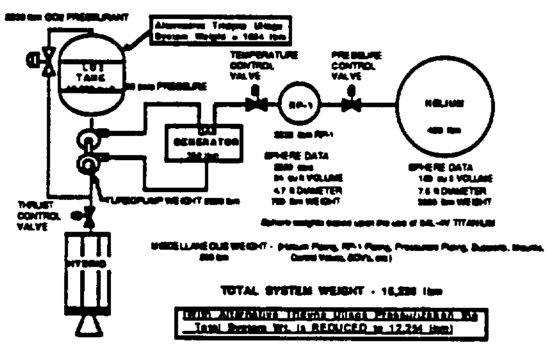


Figure 2-52. LO2/RP-1 gas generator turbo-pump system.

the initial combustion, the remaining LO2 flow is mixed with the hot combustion gases. The cool mixed flow drives the turbo-pump turbine and exhausts into the case. All of the LO2 passes through the gas generator and is vaporized. The LO2 tank ullage is also pressurized by the oxygen-rich cool gas.

Figure 2-53 presents the LO2/hybrid gas generator driven turbo-pump system. LO2 from the oxygen tank is burned with solid fuel to drive the turbine-driven turbo-pump. Most of the gas generator combustion products exhaust overboard. Approximately 5% of the LO2-rich gas generator exhaust is cooled with LO2 to pressurize the ullage.

Figure 2-54 presents the motor heat turbo-pump system. Liquid oxygen absorbs heat from the motor combustion and drives the turbo-pump turbine before discharging overboard. Some of the gaseous oxygen is cooled with liquid oxygen to pressurize the tank ullage.

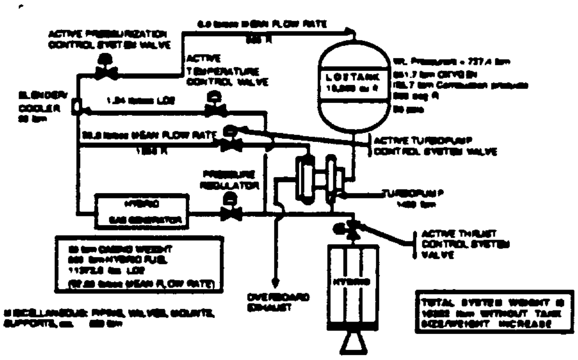


Figure 2-53. LO2/hybrid gas generator turbo-pump system.

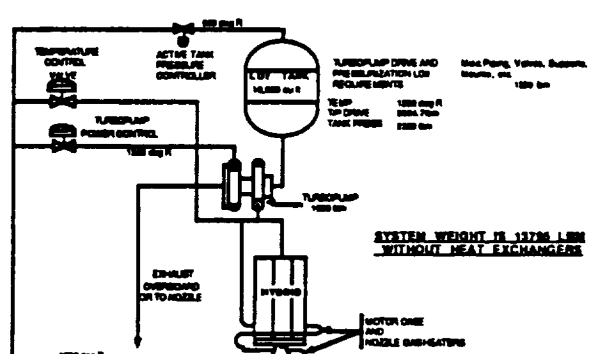


Figure 2-54. Heat turbo-pump system.

Figure 2-55 presents the hydrazine turbo-pump system. Helium pressurized hydrazine is decomposed to drive the turbo-pump turbine and the exhausts overboard. An independent Tridyne system pressurizes the tank ullage.

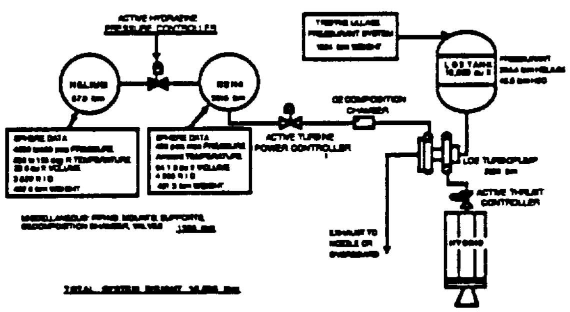


Figure 2-55. N2H4 turbo-pump system.

Figure 2-56 presents the Tridyne turbo-pump system. The cascading Tridyne drives the turbo-pump and pressurizes the tank ullage. The Tridyne gases discharge overboard.

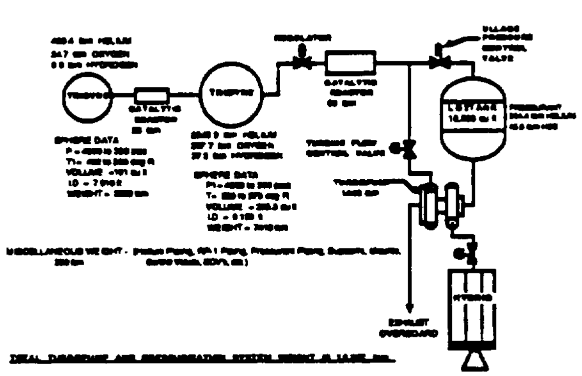


Figure 2-56. Tridyne turbo-pump system.

2.5.4.4 Ranking of Pump-Fed Concepts. Figure 2-57 shows that the Hybrid turbo-pump system is the preferred system. It was considered the safest. It is an independent system that burns a solid, inert fuel with an easily controlled single liquid oxidizer. Thrust control and combustion temperatures are readily obtained.

[illegible]	[illegible]		[illegible]		[illegible]		[illegible]		[illegible]	
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	10	2	66	13	15	3	5	1	8	1
[illegible]	16	3	27	5	34	7	0	0	23	5
[illegible]	20	3	20	3	0	0	25	4	35	5
[illegible]	0	0	10	1	40	8	25	4	25	4
[illegible]	15	3	15	3	15	3	40	8	15	3
[illegible]	0	0	30	3	40	4	0	0	30	3
[illegible]		11		28		23		17		21
[illegible]		5		1		2		4		3

Figure 2-57. Ranking of pump-fed concepts.

The motor heat turbo-pump system has the best reliability as documented by Figures 2-58 through -64.

The non-recurring costs were lowest for a Tridyne-driven turbo-pump. It is an independent system that is developed without inclusion in the motor development. It has a low-temperature turbine and uses a single fluid to both drive the turbine and to pressurize the tank ullage. The recurring costs of the motor heat-driven turbo-pump were the lowest, because once the expensive integrated motor/pressurization system is developed, this pump uses the available heat source and the existing liquid oxygen.

Figure 2-65 provides the developed pump-fed system weights. The hydrazine system has the lowest system weight and thus provides the highest performance. The motor heat turbo-system is also preferred by operational considerations. It uses an already available heat source and pressurant supply.

2.5.4.5 Ranking of Pressure-Fed Versus Pump-Fed Concepts. Figure 2-66 presents the ranking for both concepts.

The preferred concept is to pressure feed the liquid oxygen to the injectors for subsequent combustion with the grain. It is superior considering flight safety and reliability, life cycle costs, and operating considerations. The pressure-fed does sacrifice some performance because it requires a heavy wall robust tank.

The pressure-fed concept is most attractive because it is a compact system without turbo-machinery or another combustion process. It is developed as a separate system from the case/grain motor development. It requires little attention at the launch site. Its attractiveness is enhanced by new-technology lightweight pressure tanks. The tanks need structural stability for stack support prior to lift-off. Their wall thicknesses to provide

RP 1 GAS GENERATOR DRIVEN TURBOPUMP SYSTEM

UNCOMMON COMPONENTS

ITEM 1 RP 1 SPHERE TANK
ITEM 2 GAS GENERATOR (GG)
ITEM 3 HELIUM PRESSURE VESSEL
ITEM 4 TURBOPUMP

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	EXTERNAL/INTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF RP-1 FUEL	LOSS OF TANK PRESSURE AND RP 1 FUEL LOST OR DEGRADED MISSION	3	2	6
2	GG EXTERNAL LEAKAGE OR COMBUSTION INSTABILITY	POTENTIAL RICH BURN, RESULT IN GG BURN THROUGH/ EXPLOSION	4	2	8
3	EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF HELIUM/PRESSURE	LOSS OF TANK PRESSURE LOST OR DEGRADED MISSION	3	2	6
4	TURBOPUMP SEAL LEAKAGE BETWEEN RP-1 AND LOX PUMP STAGES DUE TO MECHANICAL FAILURE	MIXTURE OF RP-1 AND O2 COULD PRODUCE FIRE/ EXPLOSION	4	2	8
				TOTAL =	[illegible]

Figure 2-58. LO2/RP-1 turbo-pump criticality.

HYBRID GAS GENERATOR TURBOPUMP AND PRESSURIZATION SYSTEM

UNCOMMON COMPONENTS

ITEM 1 HYBRID GAS GENERATOR (GG)
ITEM 2 BLENDER/COOLER

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	GG EXTERNAL LEAKAGE OR COMBUSTION INSTABILITY	POTENTIAL O2 RICH BURN RESULTS IN GG BURN THROUGH/ EXPLOSION	4	2	8
2	INTERNAL/EXTERNAL LEAKAGE DUE TO MECHANICAL FAILURE	LOSS OF O2 PRESSURE LOST OR DEGRADED MISSION	3	2	6
				TOTAL =	14

Figure 2-59. Hybrid turbo-pump criticality.

CASE / NOZZLE HEATED PRESSURANT AND TURBOPUMP DRIVE LO2

UNCOMMON COMPONENTS

ITEM 1 MOTOR CASE AND NOZZLE GAS HEAT EXCHANGER

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	EXTERNAL/INTERNAL LINES/FITTINGS LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN LOSS OF O2 OTHER FAILURE MODES WOULD BE RESIDENT IN ENGINE	LOSS OF HEAT TRANSFER FUNCTION DEGRADED MISSION	3	2	6
				TOTAL =	6

Figure 2-60. Motor heat turbo-pump criticality.

MONOFUEL HYDRAZINE PUMP FED SYSTEM

UNCOMMON COMPONENTS

ITEM 1 N2H4 SPHERE TANK
ITEM 2 HELIUM SPHERE TANK
ITEM 3 DECOMPOSITION CHAMBER
ITEM 4 - TURBOPUMP
ITEM 5 TRIDYNE PRESSURE FED SYSTEM (SEE PRESSURE FED SYSTEM WITH A CRITICALITY = 12)
ITEM 6 TRIDYNE PRESSURE FED VALVES (4) NOT TAKEN INTO ACCOUNT IN THE PRESSURE - FED SYSTEM BECAUSE THEY WERE COMMON COMPONENTS FOR THAT TRADE STUDY

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	EXTERNAL/INTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF N2H4	EXTERNAL LEAKAGE CAUSES UNCONTAINED N2H4 HAZARD POTENTIAL FIRE HAZARD	4	2	8
2	EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF HELIUM/PRESSURE	LOSS OF TANK PRESSURE LOST OR DEGRADED MISSION	3	2	6
3	INTERNAL/EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE	LOSS OF N2H4 /PRESSURE EXTERNAL LEAKAGE CAUSES FIRE HAZARD	4	2	8
4	TURBOPUMP SEAL LEAKAGE BETWEEN N2H4 AND O2 PUMP STAGES DUE TO MECHANICAL FAILURE	MIXTURE OF N2H4 AND O2 COULD PRODUCE FIRE/EXPLOSION	4	2	8
					SEE NEXT PAGE

Figure 2-61. N2H4 turbo-pump criticality.

MONOFUEL HYDRAZINE PUMP FED SYSTEM (CNTD)

UNCOMMON COMPONENTS

ITEM 1 N2H4 SPHERE TANK
ITEM 2 HELIUM SPHERE TANK
ITEM 3 DECOMPOSITION CHAMBER
ITEM 4 TURBOPUMP
ITEM 5 TRIDYNE PRESSURE FED SYSTEM (SEE PRESSURE FED SYSTEM WITH A CRITICALITY = 12)
ITEM 6 TRIDYNE PRESSURE FED VALVES (4) NOT TAKEN INTO ACCOUNT IN THE PRESSURE - FED SYSTEM BECAUSE THEY WERE COMMON COMPONENTS FOR THAT TRADE STUDY

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
5	SEE PRESSURE-FED TRIDYNE OPTION				12
6	VALVES GIVE HIGH/LOW OR ERRATIC FLOW DUE TO MECHANICAL FAILURE VALVES LEAK/FAIL OPEN	PRESSURIZATION FAILURE GIVES DEGRADED/LOST MISSION	3	2	6
				TOTAL =	[illegible]

Figure 2-62. N2H4 turbo-pump criticality (Cont.).

TRIDYNE TURBOPUMP AND ULLAGE PRESSURIZATION

UNCOMMON COMPONENTS

ITEM 1 - TRIDYNE SPHERE TANK (2)
ITEM 2 - CATALITIC REACTOR (2)
ITEM 3 - TURBOPUMP

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB	CRITIC
1	EXTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF TRIDYNE/PRESSURE	EXTERNAL LEAKAGE CAUSES LOSS OF TRIDYNE. LOST OR DEGRADED MISSION	3	1X2	6
2	EXTERNAL/INTERNAL LEAKAGE DUE TO SEAL OR MECHANICAL FAILURE RESULTS IN LOSS OF TRIDYNE /PRESSURE	LOSS OF TRIDYNE PRESSURE LOST OR DEGRADED MISSION	3	1X2	6
3	TURBOPUMP SEAL LEAKAGE BETWEEN TRIDYNE AND O2 PUMP STAGES DUE TO MECHANICAL FAILURE	MIXTURE TRIDYNE AND O2 COULD PRODUCE FIRE/EXPLOSION	4	2	8
				TOTAL =	[illegible]

Figure 2-63. Tridyne turbo-pump criticality.

THE MOTOR HEAT TURBOPUMP SYSTEM HAS THE HIGHEST RELIABILITY

- O THE HYDRAZINE AND THE LO2/RP-1 OPTIONS INTRODUCE ADDITIONAL HARDWARE THAT HAVE POTENTIALLY CATASTROPHIC FAILURE MODE EFFECTS
- O THE HYBRID GAS GENERATOR AND THE TRIDYNE OPTIONS HAVE APPROXIMATELY THE SAME RELATIVE CRITICALITY IN TERMS OF POTENTIAL FAILURE MODES EFFECTS AND PROBABILITY OF OCCURRENCE
- O THE CASE/NOZZLE HEATED PRESSURANT OPTION HAS THE HIGHEST POTENTIAL RELIABILITY

THE RELIABILITY RATING FACTORS (RRFs) ARE

	CR1=14 HYBRID	CR2=28 LO2/RP-1	CR3 = 6 MOTOR HEAT	CR4=48 N2H4	CR6 = 20 TRIDYNE
RRFs	27	16	34	0	20

Figure 2-64. The motor heat turbo-pump system has the best reliability.

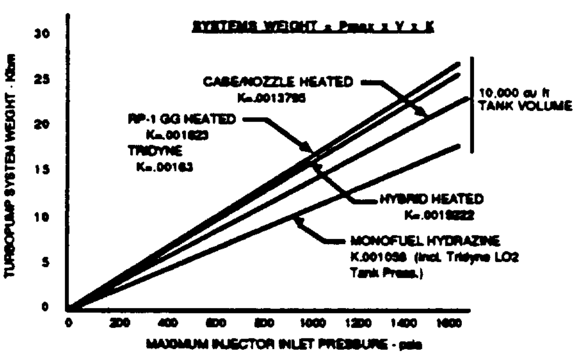


Figure 2-65. Pump-fed system weights.

CONCEPT / CRITERIA (RATING FACTOR)	PRESSURE-FED SCORE*	PRESSURE-FED WEIGHTED SCORE	PUMP-FED SCORE*	PUMP-FED WEIGHTED SCORE
FLIGHT SAFETY & RELIABILITY				
• FLIGHT SAFETY (0.20)	70	14	30	6
• RELIABILITY (0.20)	55	11	45	9
LIFE CYCLE COST				
• NON RECURRING (0.15)	70	10	30	5
• RECURRING (0.15)	70	10	30	5
PERFORMANCE				
• DELTA VELOCITY (0.20)	30	6	70	14
OPERATIONAL CONSIDERATIONS				
• LAUNCH SITE (0.10)	70	7	30	3
TOTALS		58		42
RANK		1		2

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST

Figure 2-66. A pressure-fed concept is preferred.

this support exceed those required for a low-pressure, pump-fed system. As a result, their increase in weight is modest to accommodate the higher pressures of the pressure-fed concepts.

2.5.5 REUSABLE VERSUS EXPENDABLE

2.5.5.1 Objective. The objective of the reusable versus expendable system analysis and trade study was to evaluate the feasibility of a water recovery HPT booster and the degree of recovery (entire booster, oxidizer tank only, case and nozzle only). These concepts were then evaluated against a fully expendable HPT booster.

2.5.5.2 Assumptions. The recovery method considered only the method currently in use with the STS solid rocket booster (SRB) system (i.e., chute landing into the ocean and towed by boat to shore for cleaning and refurbishment). The existing recovery system on the STS SRB appears appropriate for an entire HRB recovery system, and therefore was used as a point of departure for the trade comparisons. For each of the HRB sizes (full and quarter size) four scenarios were evaluated. They were the entire HRB recovered booster, the LO2 tank-only recovery, the case and nozzle-only recovery, and an expendable booster.

We assumed 1) a recovery attrition rate of 10% (90% recovery reliability) that a vehicle or recovered component was lost due to system malfunctions (parachute failure, high seas, etc.), and 2) that the components had a reusability life of 20 flights similar to the existing SRBs.

This trade study considers vehicles with a chamber/case pressure of 900 psia and a tank pressure of 1100 psia. The volume of the case was assumed to be 5,300 cubic feet and the volume of the tank was assumed to be 10,300 cubic feet. The tank, case and interstage are assumed to be graphite epoxy structures, with the tank having an aluminum liner. These assumptions provided engineering direction for estimation of impact damage, booster cost, and refurbishment cost.

2.5.5.3 Trade Tree. The trade tree illustrated in Figure 2-67 presents the options that were considered in the reusable versus expendable system analysis and trade study.

The point of departure for trade evaluation was the full-size, pressure-fed, composite case and tank, STS SRB method of water recovery for the entire HRB concept. This has been shown on the trade tree by highlighting the boxes around each section to clearly define the baseline configuration. The trade evaluated expendable versus recovered systems. Within the recovered systems three options were considered: recovering the entire hybrid rocket booster, recovering the oxidizer tank only, and recovering the case and nozzle only.

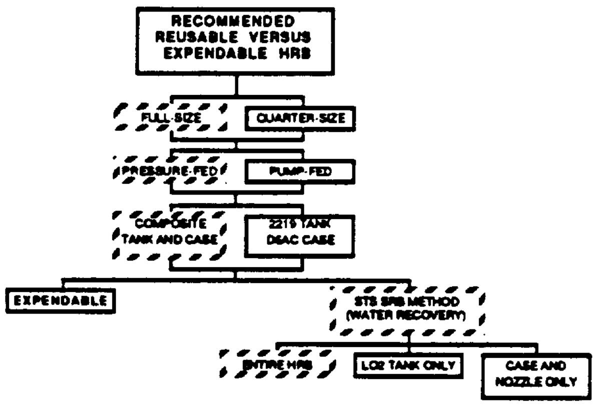


Figure 2-67. Reusable vs. expendable trade tree.

2.5.5.4 Analysis and Studies. The existing solid rocket booster is illustrated in Figure 2-68 to define the components and specifically the existing recovery system. For the entire HPT recovered booster, the existing recovery system would be implemented into the hybrid rocket booster. This would save the design, development, test, and evaluation (DDT&E) costs associated with a new booster recovery system.

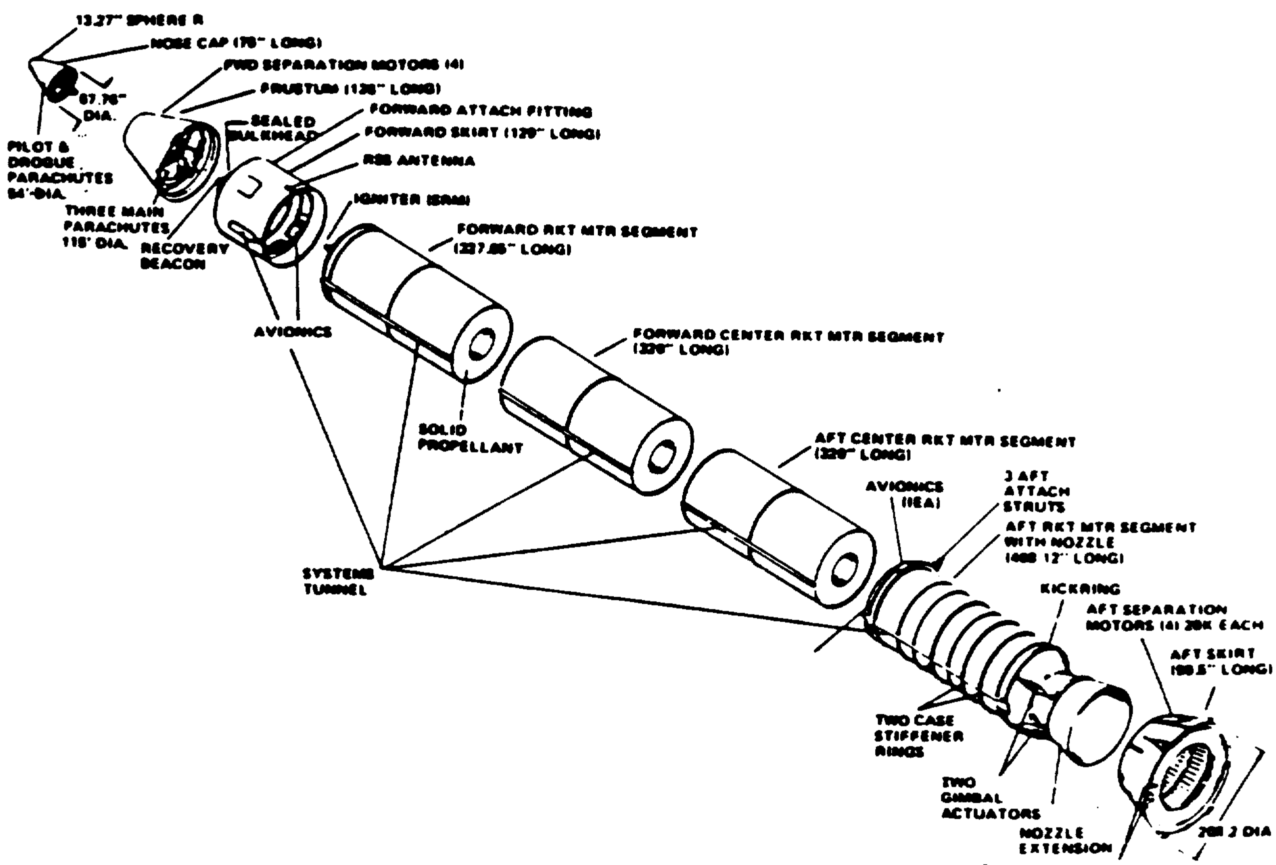


Figure 2-68. Existing solid rocket booster.

Although the SRB is a fully recovered system, there are expended components that have to be replaced after every flight. These include: the nose cap, the nozzle extension, the separation charges, the attach hardware, and the propellant. In an HRB (entire booster recovery) the expended items will include: the nose cap, separation charges, the attach hardware, and the propellant.

The HRB parachute sequence would be deployed in a similar manner to the SRB sequence (Figure 2-69). Therefore, an HPT recovered system would be employed as follows: the nose cap is separated and the pilot parachute deploys, the pilot parachute deploys the drogue bag and drogue parachute. The drogue disreefs to full inflation, drogue and frustum deploy with main parachutes. The booster and the frustum splash down.

At water impact the main chutes detach deploying the tow pendant. The frustum impacts the water at 60 feet per second and the boosters at 85-90 feet per second at a range of 141 miles. The frustum and the booster each have their own location aids and are recovered.

Recovery of the splashed-down HPT booster would proceed as illustrated in Figure 2-70: the boat approaches the boosters and verifies safing, the nozzle is inspected and towline attached, the dewatering unit is installed in the nozzle, the boosters are decompressed and dewatered, the HRBs are floated to a log-mode position and towed to port. The boosters are towed back to shore for refurbishment and re-flight.

During the recovery process an HRB has the added advantage of a separated oxidizer and fuel as opposed to an SRB where the oxidizer and fuel are premixed together in the same case. Residual solid in an SRB is very hazardous as compared to residual grain in an HRB which is inert. This would make it safer to handle and recover the HRB.

The performance of the entire HPT recovered booster was evaluated using the existing SRB recovery system, which has an overall length of 202 inches and a weight of 9,027 lbm. The system weights were calculated for: the case (composite), the tank (composite), the interstage (tank to case), the nozzle, any residual fuel, the insulation, the ignitor system, and the oxidizer feed system. These component weights were summed with the recovery system weight to yield the entire HRB recovered weight of 102,000 lbm. The SRB recovered weight was taken from current literature and reflects the "parachute weight" or the weight of the booster before it takes on water. The recovered boosters are illustrated in Figure 2-71.

The other recovery options considered included the tank-only recovery and the case- and nozzle-only recovery. The recovered weights and thus the performance was calculated as described in the entire HPT recovered booster system. The recovery system weight used is the ratio of the recovery system weight (9,027 lbm) required to recover the SRB (160,000 lbm) related to the weight required to be recovered in these two concepts. These recovery options are described above in Figure 2-72.

The tank-only recovery considered the weight of the tank and the weight of the feed system. This yielded 39,500 lbm. Scaling this weight to the SRB weights yielded a weight for the tank-only recovery system of just under 3,000 lbm. Summing these weights, we found that the recovered weight of the tank-only recovery was 42,000 lbm, as previously illustrated in Figure 2-72.

The case- and nozzle-only recovery included the weighs of the case, nozzle, residual fuel, insulation and ignitor system. Summing these components we determined a weight of 47,000 lbm. Solving for the recovery system weight we found a weight of 2,600 lbm was required. These weights yielded a total recovered weight for case- and nozzle-only recovery of 50,000 lbm.

The expendable concept would require a self-destructing system that would assure that the booster does not float in the ocean and present hazards to vessels. This could be accomplished by incorporating a system that would vent the booster and assure that it sinks to the bottom of the ocean.

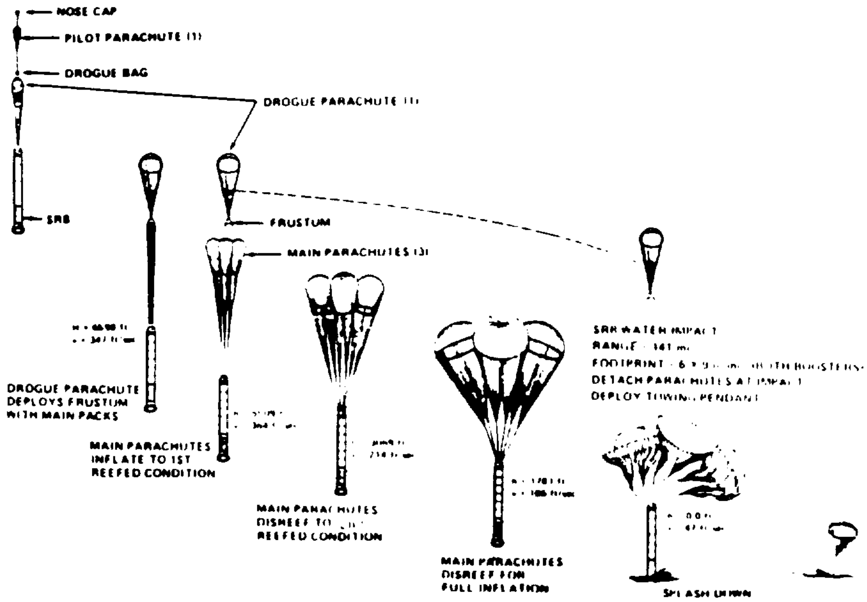


Figure 2-69. SRB decelerator subsystem deployment.

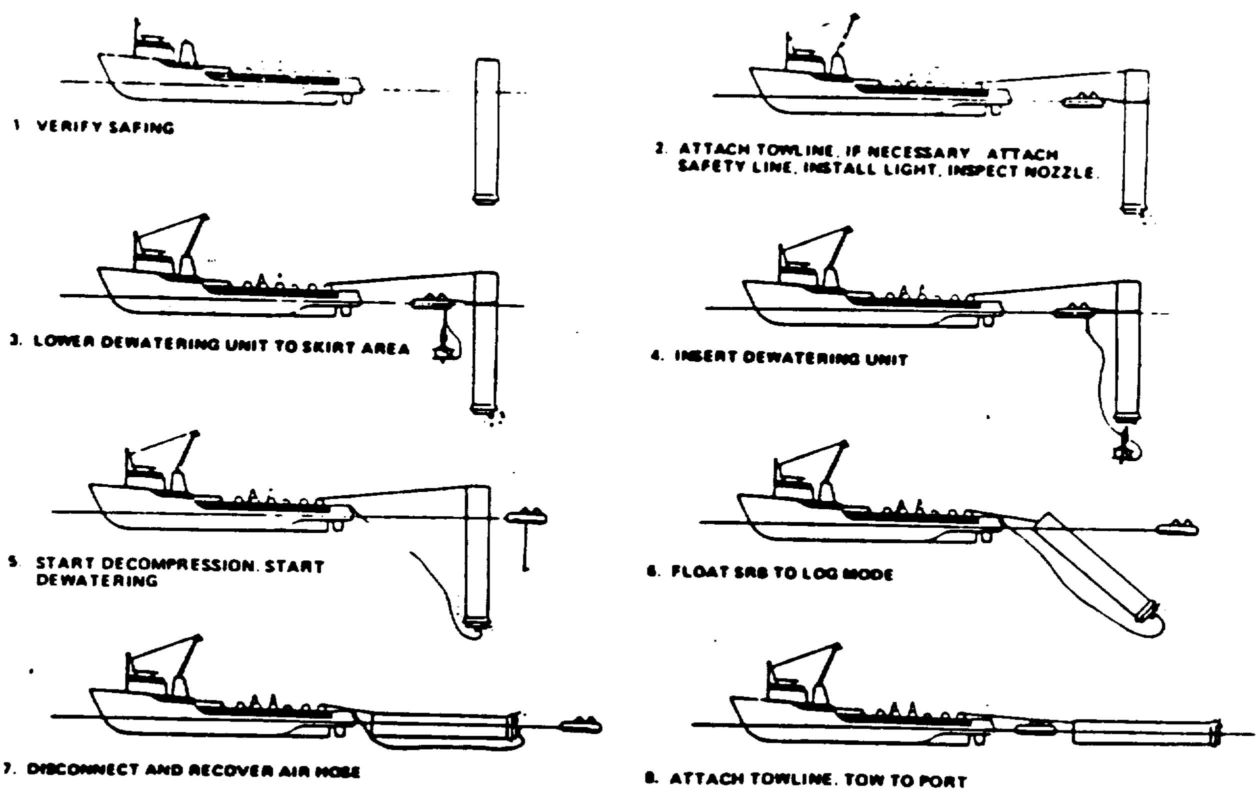


Figure 2-70. Nozzle plugging, dewatering and towback.

The quarter-size motors were then evaluated. The size of the motor did not tend to drive the solution of the trade study. The pump-fed systems were set aside, because the pressure-fed versus pump-fed pressure analysis and trade study selected a pressure-fed configuration. The ranking chart for the full-size, pressure-fed booster reflects the findings for the quarter-size, pressure-fed booster.

2.5.5.5 Ranking. Figure 2-73 presents the ranking of the reusable and expendable concepts. The expendable was considered the safest and most reliable system, because it did not require an additional system on board to recover the booster. By eliminating the recovery system, the failure modes and fault paths were reduced as illustrated in Figures 2-74 through 2-78.

The nonrecurring costs were lowest for the expendable and entire booster recovery. The expendable does not have a recovery system so no DDT&E costs would be incurred. The entire booster recovery uses the existing SRB recovery system, so again no DDT&E costs would be required.

The recurring cost analysis was based on the assumptions, 1) a recovery attrition rate of 10%, and 2) a reusability life of 20 flights. Applying these assumptions it was determined that the entire booster reusable concept was the most economical and was scored at 50 points.

The expendable concept was marked second and scored at 30 points. The LO2 tank-only and case- and nozzle-only concepts were equivalent in recurring cost analysis and they were the least desirable alternatives.

The performance ranking (Figure 2-73) of the reusable versus expendable trade study was performed on the basis of ΔV. For example, the decrease in ΔV caused by the recovery system weight necessary to recover the entire booster was approximately 66 feet per second, which is less than 1% of vehicle velocity at SRB staging.

The expendable was the preferred concept, because it did not incur a weight penalty for a recovery system. The tank-only and the case- and nozzle-only recovery were ranked second as their recovered weights were very similar.

The entire HPT recovered booster was ranked the lowest on the basis of performance, because it suffered the greatest reduction in ΔV due to the fact that its burn-out weight is the heaviest to provide for the recovery system.

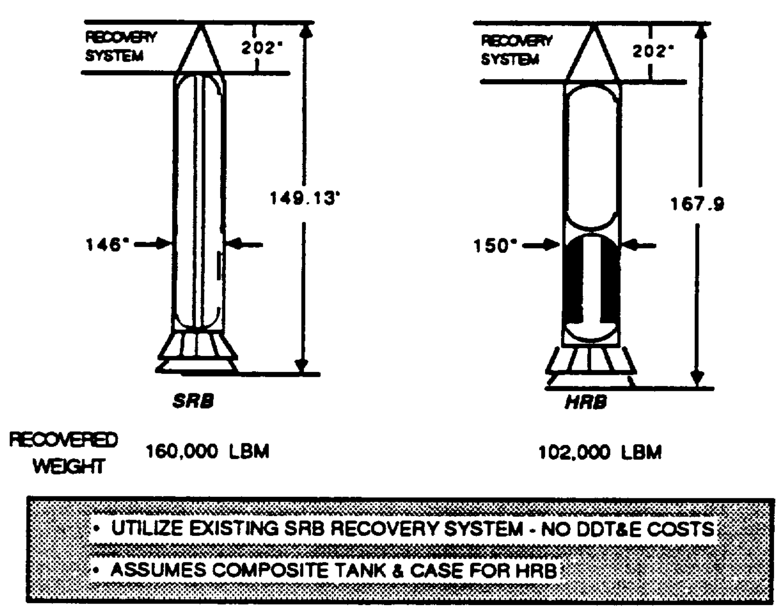


Figure 2-71. Recovery of the entire booster.

The ranking for operational considerations places both the reusable and expendable as equivalent. The recoverable booster has recovery gear to prepare, install, and check-out prior to launch. The expendable booster has a destruct system to also prepare, install, and check out. Both tasks were considered to be of the same order of magnitude. They would require further detailed analyses to identify any significant differences for operations.

2.5.6 CONFIGURATION AND MATERIALS OF TANK AND CASE

2.5.6.1 Objectives. The objectives of the configuration and materials of the HPT tank and case system analyses and trade

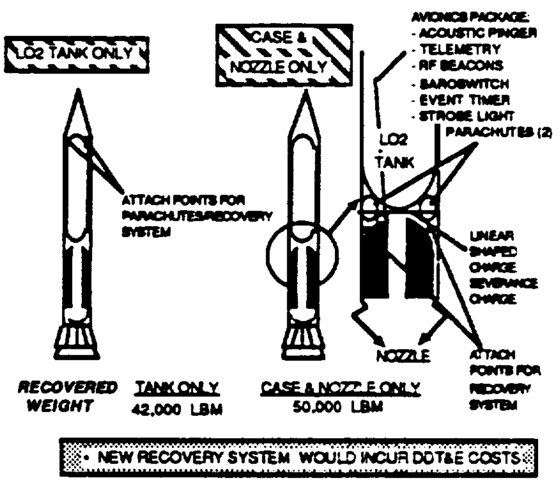


Figure 2-72. Recovering portions of an HRB.

Criteria (Rating Factor) / Concept	Reusable: Entire Booster Score*	Weighted Score	LO2 Tank Only Score*	Weighted Score	Case, Nozzle Only Score*	Weighted Score	Expendable Score*	Weighted Score
Flight Safety & Reliability								
• Flight Safety (0.20)	30	6	10	[illegible]	17	3	30	7
• Reliability (0.20)	30	6	20	4	16	3	30	7
Life Cycle Cost								
• Non Recurring (0.15)	40	6	12	2	6	1	40	6
• Recurring (0.15)	50	7.5	10	1.5	10	1.5	30	4.5
Performance								
• Delta Velocity (0.20)	21	4	24	5	24	5	31	6
Operational Considerations								
• Launch Site (0.10)	25	2	10	[illegible]	10	1	60	6
Totals		31.5		17.5		14.5		38.5
Rank		2		3		4		1

*Scored from 0 to 100 where 100 is the best

Figure 2-73. Ranking of reusable vs. expendable concepts.

studies were: 1) to recommend either a separate tank and case or an integrated tank/case with a common separating bulkhead, 2) to recommend the materials of construction for the tank and case, and 3) to provide tank and case parametric data for other analyses and trades.

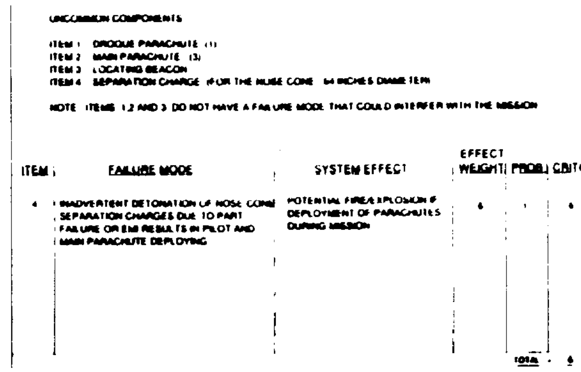


Figure 2-74. Reusable: entire booster criticality.

2.5.6.2 Assumptions. The assumptions presented in Table 2-2 are typical for sizes between the quarter- and full-size boosters using HTPB and liquid oxygen. The pressures assumed represent a pressure-fed liquid oxygen tank and a pressure drop of 200 psig through the oxygen injectors. The ultimate safety factor was 1.4, which is greater than that currently used on the solid rocket motor which is 1.3. A safety factor of 1.4 was also selected to provide ready comparison with studies previously completed for liquid rocket boosters.

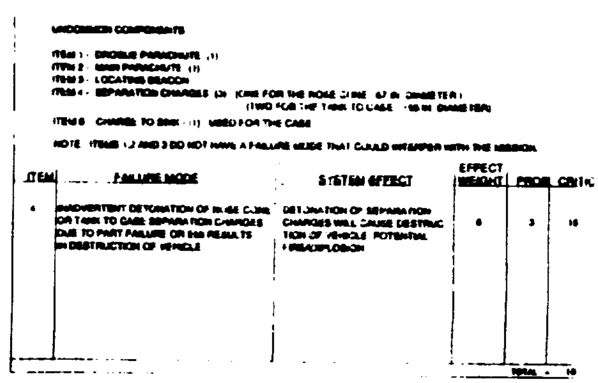


Figure 2-75. Reusable: LO2 tank-only criticality.

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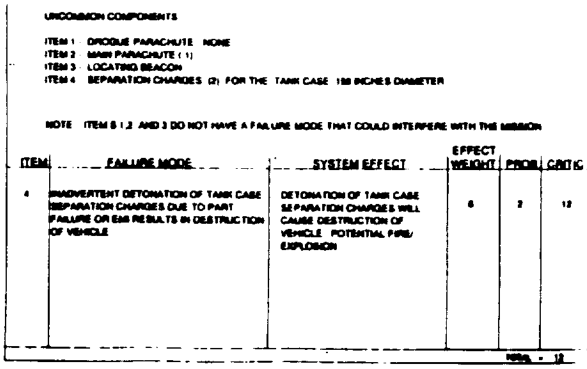


Figure 2-76. Reusable: case, nozzle-only criticality.

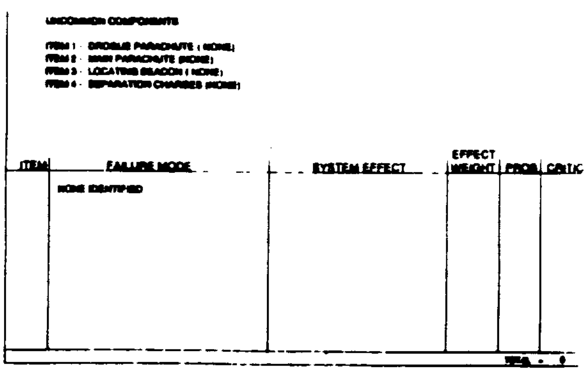


Figure 2-77. Expendable: entire booster criticality.

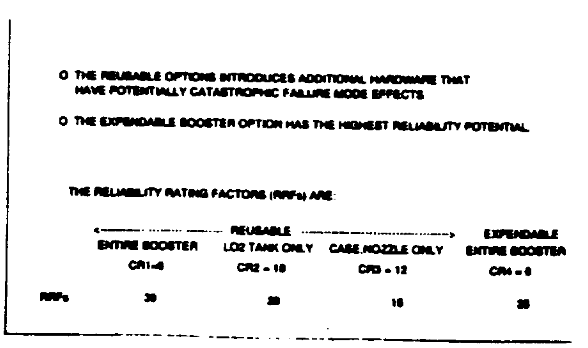


Figure 2-78. The expendable booster concept has the best reliability.

2.5.6.3 **Trade Tree.** The alternatives studied are presented in Figure 2-79. The full-size booster using a pressure-fed liquid oxygen tank was emphasized. The separate tank and case versus a common bulkhead was investigated.

Aluminum, 2090 aluminum-lithium, and graphite/epoxy with a 1,100 aluminum liner were studied as the materials of construction for the tank. Materials studied for the case were D6AC and graphite/epoxy. The materials selected represent both currently used materials and materials that may have cost or performance advantages when used to construct boosters. For example, 2219

Table 2-2. The assumptions are typical for an ASRM-size booster.

1. The tank volume is 10,000 cubic feet for the full-size HPT booster.
2. The case volume is 5,256 cubic feet for the full-size HPT booster.
3. The tank and case pressures are between 650 and 1,650 psi, and 800 and 1,100 psi, respectively.
4. The full size tank and case diameters are 150 inches.
5. The ultimate safety factor shall be 1.4 for both maximum operating pressure and other loads.

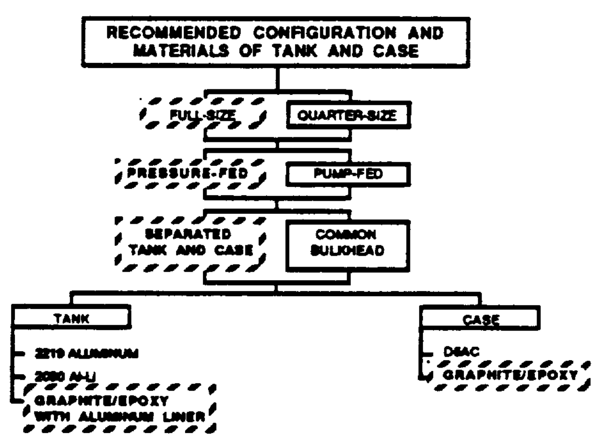


Figure 2-79. The configuration and materials trade tree considered all viable alternatives.

aluminum alloy was used to construct tankage for the Saturn stages and is currently used to manufacture the external tank of the STS. D6AC is used to fabricate the current solid rocket motor case for the SRBs of the STS.

The higher strength 2090 aluminum-lithium alloy is advocated as a replacement for 2219 to reduce stage invert weights. The 2090 alloy has tested as being questionable for use with liquid oxygen, especially at higher pressures, but is included herein to assess its other possible benefits.

Composite cases have been used for smaller solid rocket motors, but are not in production for the sizes of this study. Prototype cases have been built. Graphite/epoxy tanks require a liquid oxygen-compatible liner such as 1100 aluminum, because the base material is not oxygen-compatible. This technology is evolving through interest in reducing the cost and weight of booster tankage.

2.5.6.4 Separated Tank/Case vs. Common Bulkhead

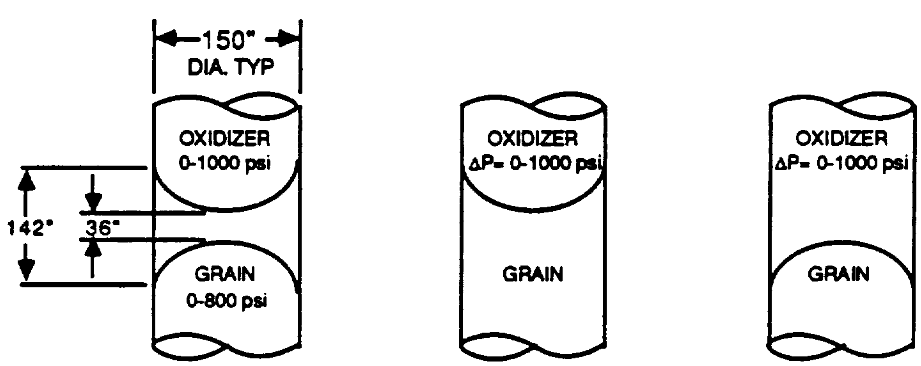
Analyses and Studies

Figure 2-80 presents the configurations studied and summarizes dimensions and masses. The separate tank and case were taken as the base case. A separation of 36 inches was allowed as separation between the tank bottom and the case head to provide space for access and installation of plumbing and other components. The distance between the tangencies of the tank bottom and the case head calculated at 142 inches when both were considered √2 bulkheads.

Both the tank bottom and case head are primarily in tension, although it is recognized that the tank bottom will experience compression stresses during tanking.The common bulkhead provides for an overall booster length reduction of 71 inches. The length decrease occurs by fully using the space provided in the base case for the tank bottom, the case head, and access. The decrease in length eliminates the weight of the tank-to-case adapter and the case forward head. These weight savings were estimated at 12,500 pounds mass.

The common bulkhead in tension is preferred over a compression bulkhead. It would be a challenge to develop a compression bulkhead for a 150-inch diameter booster able to withstand 1,000 psi delta pressure. Regardless, the tension bulkhead is carried as most typical of a common bulkhead.

Launch vehicles have incorporated both separated and common bulkheads for their



BOOSTER LENGTH (IN.)	BASE	-71	-71
BULKHEAD(S)	IN TENSION	IN TENSION	IN COMPRESSION
WEIGHT, lbm	BASE	-12,500	?

Figure 2-80. A common bulkhead reduces booster length and weight.

tankage. Boosters more commonly have separated tanks, while upper stages are more prone to have a common bulkhead between the propellants to increase their mass fraction. For example, the external tank separates the liquid oxygen forward with its own tank from the aft liquid hydrogen tank.

The Titan core has separate tanks for its hypergolic propellants. The first stage of the Saturn V had separated liquid oxygen and RP-1 tanks. An exception for boosters is the Atlas, which has a common bulkhead separating liquid oxygen and RP-1. But the Atlas is a stage and one-half booster; the payload-to-tankage exchange ratio is more significant than for the booster stage of a multi-stage launch vehicle.

Ranking

The ranking presented in Figure 2-81 strongly recommends a separated tank and case. Flight safety and reliability, life cycle cost, and operational considerations all improve at the expense of performance.

For flight safety and reliability, the two independent bulkheads provide assurance that the liquid oxygen will not inadvertently combine with the grain. Each bulkhead is designed to accommodate its own loads and environments. There is no opportunity for a common bulkhead reversal.

CRITERIA (RATING FACTOR) \ CONCEPT	SEPARATED TANK/CASE		COMMON BULKHEAD	
	SCORE*	WEIGHTED SCORE	SCORE*	WEIGHTED SCORE
FLIGHT SAFETY & RELIABILITY				
• FLIGHT SAFETY (0.20)	95	19	5	1
• RELIABILITY (0.20)	100	20	0	0
LIFE CYCLE COST				
• NON RECURRING (0.15)	70	10	30	5
• RECURRING (0.15)	70	10	30	5
PERFORMANCE				
• DELTA VELOCITY (0.20)	0	0	100	20
OPERATIONAL CONSIDERATIONS				
• LAUNCH SITE (0.10)	80	8	20	2
TOTALS		67		33
RANK		1		2

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST

Figure 2-81. A separated tank and case are preferred.

Life cycle costs are less for a separated tank and case, because each can be developed and manufactured separately in manageable lengths for their own requirements. The requirements differ primarily in that the tank must be compatible with liquid oxygen at cryogenic temperature, and the case contains an inert fuel at ambient temperature.

The additional adapter and two bulkheads are less costly than a vacuum-jacketed intermediate bulkhead and its joints within the tank and/or case. In addition, the liquid oxygen flow controls and the grain igniters would be installed external to the tank and case outside diameters. Their life cycle costs would increase to accommodate their unusual locations.

The common bulkhead is preferred by the performance criteria. There is a 12,500-pound savings in burn-out weight. This weight savings will increase the vehicle delta velocity at booster burn-out by 89 feet per second, which is less than 1% of the total delta velocity at staging.

The common bulkhead is also attractive as a method to reduce the overall booster length. The decrease of 71 inches may be significant if the HPT booster were being examined to replace an operational booster, and it was important to fit within the existing envelope. The separated tank/case is preferred during operations. It provides a safe, logical place to divide the booster into sections for transport, handling, and erection. A common bulkhead would probably require joints in the tank or case. These joints are highly undesirable, because they would be exposed to temperature, pressures, and the fluids/gases during booster flights.

2.5.6.5 Materials of Tank and Case

LO2 Tank

The LO2 tank baseline design is a cylindrical self standing monocoque structure with $\sqrt{2}$ bulkheads. Tank walls are constructed of filament wound Im7/8551-1 graphite/epoxy 0.80 inches thick with an integral 1100-0 aluminum alloy liner 0.05 inches thick. The aluminum liner prevents LO2 from coming into contact with the graphite/epoxy. Tank volume is 10,000 cubic feet with a diameter of 150 inches.

The alternate materials for the LO2 tank are 2219 aluminum and 2090 aluminum-lithium. A structural analysis of the LO2 tank was performed using the loads from the LRB program. Flight loads, internal pressure, and "twang" loading conditions were used to size the LO2 tank. Both pump-fed and pressure-fed configurations were sized.

Weight/volume versus pressure of all three tank materials is shown in Figure 2-82. The graphite/epoxy tank with an integral aluminum liner shows the lowest

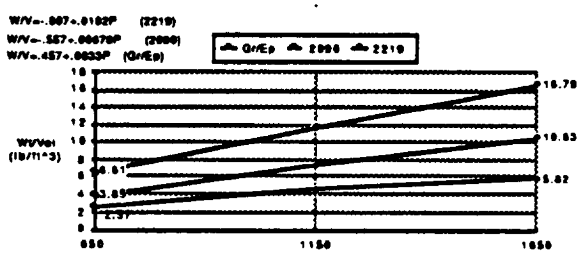


Figure 2-82 Tank weight/volume vs. pressure.

weight/volume value, 2090 aluminum at a higher value and 2219 aluminum at the highest value. The graphite/epoxy material shows a weight/volume for a given pressure advantage over 2219 aluminum and 2090 Al-Li of 2.57 and 1.5, respectively.

Figure 2-83 shows the weight vs pressure for the baseline pressure-fed 10,000ft^3 LO2 tank. The figure reflects the weight savings with the graphite/epoxy material showing the lowest weight design by a factor of 2.57.

The baseline LO2 tank pressure is 1,000 psi for the pressure-fed system. The pump-fed system LO2 tank pressure is 0.0 psi. Figures 2-84, 2-85 and 2-86 show the structural weights of the LO2 tank for a pump- and pressure-fed system, and list the structural weights required for each load condition. The pump system tank is sized for structural stability whereas the pressure-fed system is sized for pressure loads. Tank weights show that graphite/epoxy represents the lightest weight design, 2090 aluminum-lithium is second, and 2219 aluminum is third.

A lighter weight design for the pump-fed monocoque tanks is achievable by allowing partial pressure stabilization. The negative axial loads on the tank create a stability critical structure. The structure is then sized for stability. When the structure is sized for stability the stress within the tank wall does not achieve maximum strength of the material. This results in an inefficient design.

Pressurizing the tank creates a positive axial load that cancels the negative axial loads.

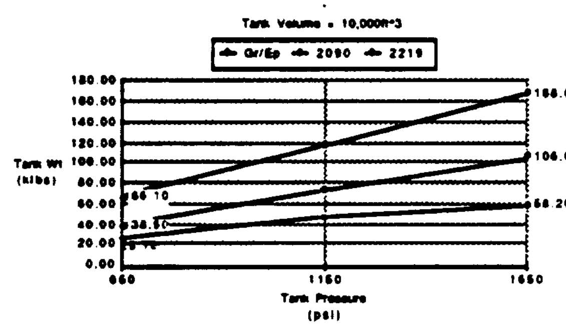


Figure 2-83. Tank weight vs. pressure.

Tank Data: Volume 10,000ft^3; Dia. 150in; Mtl. gr/ep; F.S. 1.4	Pump-Fed zero pressure		Pressure-Fed 1000psi	
Load Condition	Weights (lbs)			
Flight Loads: Strength	14,373	t=.4	14,373	t=.4
Flight Loads: Stability	23,353	t=.7	23,353	t=.7
Pressure Loads:	0		37,097	t=1.16
Tank Weight	23,353		37,097	

Figure 2-84. Graphite/epoxy pump-fed and pressure-fed LO2 tank weights.

The pressure creates hoop loads in the tanks and the tank wall is sized to the hoop loads. This makes full use of the materials strength and maximizes the efficiency of the design. The ground handling loads are not as severe as the flight loads and thus the LO2 tank is stable during preflight operations. The pressure required for stabilization would be less than 500 psi.

Manufacturing of the aluminum LO2 tank is in two parts, the end domes and the cylinder. The cylinder is made up of skin panels that have a constant thickness. They are roll-formed and fusion-welded. The domes are made of a series of constant thickness gore sections fusion welded together. The domes are fusion-welded to the cylinder.

An alternative manufacturing method for the end domes is spin-forming, which would eliminate the gore section details by spin-forming the end dome to shape from a single sheet of aluminum. The spin-forming process has been demonstrated on small

Tank Data: Volume 10,000ft^3; Dia. 150in; Mtl. 2090; F.S. 1.4	Pump-Fed zero pressure		Pressure-Fed 1000psi	
Load Condition	Weights (lbs)			
Flight Loads: Strength	12,341	t=.27	12,341	t=.27
Flight Loads: Stability	34,797	t=.75	34,797	t=.75
Pressure Loads:	0		61,707	t=1.33
Tank Weight	34,797		61,707	

Figure 2-85. 2090 Al-Li pump-fed and pressure-fed LO2 tank weights.

Tank Data: Volume 10,000ft^3; Dia. 150in; Mtl. 2219; F.S. 1.4	Pump-Fed zero pressure		Pressure-Fed 1000psi	
Load Condition	Weights (lbs)			
Flight Loads: Strength	20,155	t=.4	20,155	t=.4
Flight Loads: Stability	37,416	t=.75	37,416	t=.75
Pressure Loads:	0		100,774	t=2.02
Tank Weight	37,416		100,774	

Figure 2-86. 2219 Al pump-fed and pressure-fed LO2 tank weights.

domes and could be a less costly process. The manufacturing processes for both 2219 aluminum alloy and 2090 aluminum-lithium alloy are the same and represent standard practices with minimum manufacturing risk.

The graphite/epoxy tank with an integral aluminum liner is manufactured by filament-winding the graphite/epoxy around the 1100-0 aluminum liner.

The objective of this process is to have compressive contact stress between the liner and graphite/epoxy tank wall. To do this the tank is wound at cryogenic temperature so that when filled with LO2, the liner stays in contact with the composite. This is critical because a disbond coupled with a liner leak at the same location could result in sufficient contact area between LO2 and organic composite to create a hazard. By keeping a compressive contact stress at the liner interface, voids can be minimized or eliminated, and the resulting compressive stresses in the liner will prevent the formation and propagation of cracks and keep any existing cracks closed.

To achieve compression in the liner, the tank is wound at cryogenic temperatures. After winding the tank is brought to elevated temperatures and cured.

Filament winding of the graphite/epoxy is a mature and fully automated process. However, tooling to handle pressurizing and to cooling of the aluminum liner during filament winding needs to be developed. To validate the manufacturing process and analytical analysis, a subscale test article needs to be built and tested.

Manufacturing of the graphite/epoxy tank is an automated process and will result in less manufacturing time than the aluminum tanks. Development time and funding is needed to mature the process of filament winding at cryogenic temperatures. The construction of the aluminum tanks is straightforward with a minimum of development time. The graphite/epoxy tank with an aluminum liner is ranked above the aluminum tanks for produceability, but requires more development of the manufacturing process.

Case

A structural analysis of the case was performed using the flight loads from the LRB program. Flight loads, internal pressure, and "twang" loading conditions were used to size the case.

The baseline rocket motor case is a cylindrical, self-standing monocoque structure with √2 bulkheads. The walls are constructed of filament wound in Im7/8551-1 graphite/epoxy 1.04-inches thick. The rocket motor case volume is 5256ft^3 and has a 150-inch diameter. Internal pressure is 800 psi. A wall thickness of 0.97 inches is required for stability.

The alternate material for the case is D6AC high-strength steel alloy. The D6AC case at 800 psi is sized for strength, but under 800 psi it is sized for stability. The skin thickness reduced from 0.44 inches for the case at 800 psi to 0.42 inches for pressure under 800 psi. Thus the weight for the baseline D6AC case is close to the minimum structural weight. The graphite/epoxy case will reduce to a lower weight if the pressure is reduced.

Weight/volume versus pressure of the case for both candidate materials is shown in Figure 2-87. The graphite/epoxy tank shows a weight/volume value less than half than that for D6AC high-strength steel alloy. The actual weights listed in Figure 2-88 for the baseline show that the graphite/epoxy material is half the weight of the D6AC material.

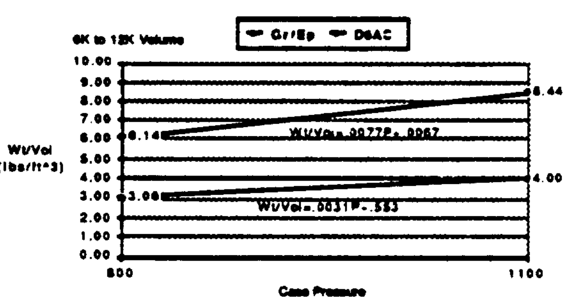


Figure 2-87. Case weight/volume vs. pressure.

The manufacturing process of the rocket motor case is dependent on how the solid grain rocket fuel is configured. An advantage to using composite materials for the case is that the solid grain fuel can be cast to shape and the graphite/epoxy material can be filament wound over the fuel. The graphite/epoxy is then cured using elevated temperatures and an outside wrap is wound around the case to provide curing pressure.

The D6AC steel case configuration can be segmented like the present SRM or it can be made with a removeable end dome allowing the fuel to be cast into place. The second is the most promising for a cost-efficient process. The case cylinder is made up of plate stock roll-formed and fusion-welded together. The end domes would be made of gore sections fusion-welded together. One dome would have a structural joint allowing attachment to the cylinder, the other dome would be fusion-welded to the cylinder.

Case Material	Baseline Rocket Motor Case (lbs)
Graphite/epoxy Im7/8551-7	16,594
D6AC High Strength Steel	33,273

Case Volume = 5256ft^3

Figure 2-88. Case baseline weights.

The manufacturing process for the graphite/epoxy tank would be the most automated process even if it had the same configuration as the D6AC case. The automated process would represent the less expensive manufacturing process.

Ranking

Figure 2-89 provides the ranking of the tank and case materials. The GR/EP tank and case are preferred for their highest ranking in flight safety, nonrecurring cost, performance, and operational considerations.

There are no apparent flight safety discriminators among the tank/case materials concept traded. It is assumed that all concepts could be constructed to meet the load requirements plus an adequate safety factor. Any weight-saving material would add to the performance margin and therefore contribute indirectly to greater safety margins. The lightest combination was selected for that reason.

Similar or constant components were not included in the reliability evaluations, because they cancel out in comparison of options. Criticality for a component was defined as the product of the failure mode (worst-case) weighting times the probability of occurrence as shown in Figure 2-90. Figure 2-91 presents the criticality calculations for the

CONCEPT / CRITERIA (WEIGHTING FACTOR)	2219 TANK D6AC CASE SCORE*	2219 TANK D6AC CASE WEIGHTED SCORE	GR/E TANK D6AC CASE SCORE*	GR/E TANK D6AC CASE WEIGHTED SCORE	2219 TANK GR/E CASE SCORE*	2219 TANK GR/E CASE WEIGHTED SCORE	GR/E TANK GR/E CASE SCORE*	GR/E TANK GR/E CASE WEIGHTED SCORE
FLIGHT SAFETY & RELIABILITY								
• FLIGHT SAFETY (0.20)	20	4	20	4	20	4	40	8
• RELIABILITY (0.20)	50	10	30	6	30	6	0	0
[illegible]								
• NON RECURRING (0.10)	30	6	0	0	60	7	30	3
• RECURRING (0.10)	30	3	0	0	40	6	40	8
PERFORMANCE								
• DELTA VELOCITY (0.30)	0	0	20	7	10	3	60	18
[illegible]								
• LAUNCH SITE (0.10)	0	0	40	4	10	1	80	8
TOTALS		22		20		20		32
RANK		3		4		2		1

* SCORED FROM 0 TO 100 WHERE 100 IS THE BEST

Figure 2-89. GR/EP tank and case are preferred.

FAILURE MODES CONSEQUENCES (FMC)	WEIGHTS
CATASTROPHIC FAILURE	4
LOST OR DEGRADED MISSION	3
LOSS OF SAFETY OPTIONS (OPERATIONAL OR HARDWARE REDUNDANCY)	2
NO IMMEDIATE EFFECT	1
PROBABILITIES OF OCCURRENCE (PO)	
HIGH: "WILL HAPPEN" TO 1E-2	4
MEDIUM: 1E-2 TO 1E-4	3
LOW: 1E-4 TO 1E-6	2
VERY LOW: 1E-6 AND LESS	1

COMPONENT CRITICALITY = FMC WEIGHT X PO WEIGHT

Figure 2-90. Failure modes weighting and probabilities of occurrence.

tanks and cases when constructed from alternative materials. Figure 2-92 presents final ratings from the reliability analyses. It concludes that the failure modes introduced by all options are potentially catastrophic. Also that GR/EP tank and cases introduce a higher probability of occurrence.

The nonrecurring costs for manufacturing of the 2219 aluminum tanks is less than that required to manufacture the aluminum-lined graphite/epoxy tank. In the latter case, the concept is to build an aluminum tank and overwrap the tank with a graphite/epoxy composite material.

The manufacturing development of filament-winding graphite/epoxy or fusion-welding D6AC rocket motor cases does not represent a high technical risk. Both processes are mature and do not require extensive development , because they are both used today on rocket motor cases.Nonrecurring costs can be shared if both the tank and case are manufactured from composite materials. They have a common diameter and share a common dome shape.

Recurring Cost

Filament-winding the graphite/epoxy over the

ITEM	FAILURE MODE	SYSTEM EFFECT	EFFECT WEIGHT	PROB.	CRITIC
GR/E TANK	TANK EXTERNAL/INTERNAL ALUMINUM LINER AND/OR GRAPHITE EPOXY LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN LOSS OF LO2	LOSS OF LO2 COULD RESULT IN FIRE/EXPLOSION	4	3	12
GR/E CASE	CASE EXTERNAL LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN FLAMES OVERHEATING THE LH2/LO2 TANKS	OVERHEATING OF LH2/LO2 TANKS RESULTS IN FIRE/EXPLOSION	4	3	12
2219 TANK	TANK EXTERNAL/INTERNAL LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN LOSS OF LO2	LOSS OF LO2 COULD RESULT IN FIRE/EXPLOSION	4	2	8
D6AC CASE	CASE EXTERNAL LEAKAGE DUE TO SEALS OR MECHANICAL FAILURE RESULTS IN FLAMES OVERHEATING THE LH2/LO2 TANKS	OVERHEATING OF LO2/LH2 TANKS RESULTS IN FIRE/EXPLOSION	4	3	8

Figure 2-91. Tank and case criticality.

THE RELIABILITY FACTOR R_i FOR OPTION (i) IS CALCULATED AS FOLLOWS

$$R_i = 100\left(1 - \frac{(CR_i - CR_{min})}{CR_{max} - CR_{min}}\right)$$

THE RELIABILITY RATING FACTOR FOR OPTION (i) IS

$$RRF_i = \frac{R_i}{\underset{i}{\mathrm{SUM}}\ R_i}$$

	CR1 = 16 2219 TANK (AL) D6AC CASE	CR2 = 20 GR/E TANK D6AC CASE	CR3 = 20 2219 TANK GR/E CASE	CR4 = 24 GR/E TANK GR/E CASE
RRFs	50	25	25	0

Figure 2-92. Tanks and cases made from standard materials have the best reliability.

will show a cost advantage over the aluminum LO2 tank fusion-welding process.

Because filament-winding is more automated than roll-forming of D6AC plates and fusion welding them together, it represents the lower cost manufacturing process.

Performance

The performance criteria is ranked on the basis of the differences in booster burn-out weights and reflected as delta velocity. Graphite/epoxy LO2 tanks with an integral aluminum liner offer the lightest weight design. The aluminum materials 2219 and 2090 aluminum alloy show a marked difference in weight performance. The 2090 aluminum-lithium shows a significant advantage to 2219. Its higher strength and lower density make it the preferable material for high-pressure LO2 tanks over 2219.

For a zero-pressure, pump-feed system, the 2090 aluminum-lithium loses the majority of its advantage because the tank is sized for stability. The tank wall thickness for both tanks (2090 aluminum-lithium and 2219) are the same for a zero-pressure tank. 2090 aluminum-lithium is lighter than the 2219 due to its lower density.

A graphite/epoxy rocket motor case offers the lighter weight design over D6AC high-strength steel. The graphite/epoxy case is half the weight of D6AC.

Figure 2-93 summarizes the weight and corresponding delta velocity for the various material combinations for the tank and case. The materials selected have a significant impact on the delta velocity. For example, the penalty in incorporating current materials of 2219 aluminum and D6AC steel results in a delta velocity decrease at booster burn-out of approximately 535 feet per second. This is a significant value, because it represents almost six percent of the vehicle velocity at that time.

The GR/EP tank and case are preferred at the launch site, because tanks and cases manufactured from GR/EP are lighter and easier to handle. Given a total vehicle performance requirement, the GR/EP tank and case will also be smaller, because the stage mass fraction is greater and less propellant is required to accomplish a given mission.

Parametric Data

Figure 2-94 presents the initial weight-to-volume ratios incorporated in our system analyses and trade studies. The external tank at 0.64 and the LRB at 2.10 and 3.80 are included as reference. The pump-fed LRB and HPT tanks have the minimum thicknesses required to support the STS stack and to also resist the twang loads from SSME ignition.

MATERIALS				
TANK	GR/E	GR/E	2219	2219
CASE	GR/E	D6AC	GR/E	D6AC
WEIGHT (lbm)	BASE	+16,471	+64,500	+80,971
ΔV (ft/sec)	BASE	-114	-428	-535

Figure 2-93. New technology significantly increases booster performance.

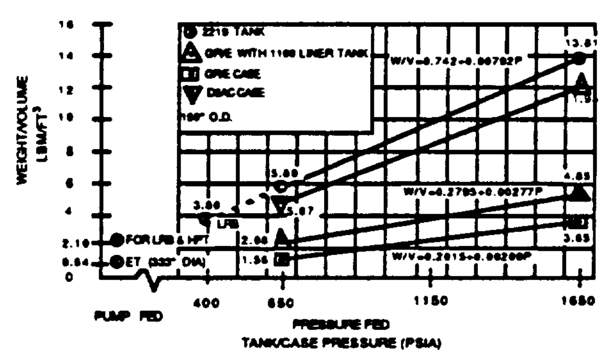


Figure 2-94. Tank and case weight/volume ratios.

Figure 2-95 presents the scaling laws developed during this program. They are more conservative than the initial weight-to-volume ratios. The "pressure required for stability" refers to the equivalent pressure necessary to support the STS system and to accommodate the "twang" load. Tanks and cases designed for the noted pressures have sufficient wall thickness for all pre-flight loads. Lower pressure tanks and cases need additional wall thickness for pre-flight loads.

TANK		PRESSURE REQUIRED FOR STABILITY
2219	W = V [0.007 + 0.0102 X P]	366 psig
GR/EPp	W = V [0.457 + 0.0033 x P]	569 psig
CASE		
D6AC	W = V [0.0067 + 0.0077 x P]	764 psig
GR/EPp	W = V [0.553 + 0.0031 x P]	746 psig

Note: W in lbm, P in psi, V in cubic feet

Figure 2-95. HPT tank and case scaling laws.

2.6 CONCEPTUAL DESIGN PACKAGE

2.6.1 SELECTED HYBRID PROPULSION CONCEPT. The results of the system analyses and trade studies are shown in Figure 2-96. From these results the final hybrid propulsion concepts (ASRM-size and quarter-ASRM-size) were derived. Both concepts use the classical HP methodology, as described in Section 2.3.1.

As illustrated in Figure 2-97, the ASRM-size hybrid propulsion system satisfied the ASRM planned envelope requirements. The quarter-ASRM-size hybrid propulsion system was then sized using the same fuel composition.

Hybrid propulsion systems can use differing solid fuel compositions that will vary the actual specific impulse and mixture ratio of the resultant booster. The hybrid propulsion parameters can thus be adjusted to satisfy specific booster requirements. In this case we have constrained the size of the HPT Booster to the planned diameter and length of the ASRM. In doing so we have sacrificed some performance.

However, the positive responsiveness to variable propellant compositions is an attractive feature of hybrid propulsion, and the technology developed capitalizes upon this. These various fuel blends are accommodated with the applicable codes.

The 500-psia chamber pressure was recommended by our system analysis and trade study as the preferred pressure level for a pressure-fed booster. It remains constant for the range of booster sizes investigated.

Our system analyses and trade studies recommended a new-technology liquid oxygen tank that reduces the HPT booster burn-out weights when compared to current-technology tanks.

Figure 2-98 depicts a LO2 tank constructed with a graphite epoxy shell over an aluminum inner lining. Notice that the tank weight is reduced to 33,217 LBM, a reduction of some 20% as compared to conventional 2219 aluminum tanks. The weight reduction is reflected as an increase in payload for the fixed envelope booster or as a reduction in booster size for a given performance level.

The maximum operating pressure requires a tank wall thickness sufficient to support the STS stack during launch processing. When pressurized, it is more than sufficient to resist the "twang" loads during main engine ignition.

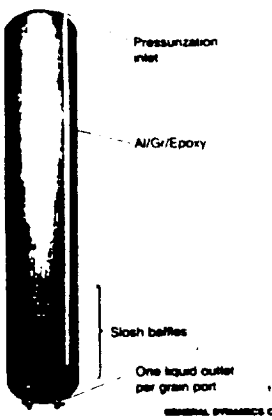
While a composite LO2 tank is very attractive to support a pressure-fed LO2 system, it is not an enabling technology. A 2219 aluminum tank processed using current technology is an acceptable alternative.

ASRM Size	Characteristic	1/4-ASRM Size
2.59 M	Nominal vac thrust (lbf)	648K
150	Diameter (inches)	90
1790	Length (inches)	1248
272.5	Isp vac actual (sec)	274.7
1.27	Mixture ratio	1.29
500	Average P_c (psia)	500

Figure 2-97. Our selected configurations are interchangeable with current boosters.

TRADES	ASRM SIZE	1/4 SIZE
CONCEPT	CLASSICAL	CLASSICAL
BOOSTER O.D. (inches)	150	90
PROPELLANT	LO2/HTPB/GAP/ZN	LO2/HTPB/GAP/ZN
PERFORMANCE		
MIXTURE RATIO	1.27	1.29
Isp (THEO. VAC., sec.)	272.5	274.7
LOADING EFFICIENCY	75	73
FUEL SLIVER (%)	0.7	0.0
NUMBER OF PORTS	4	2
PRESSURES (psia)		
TANK	500	500
COMBUSTION	743	743
CASE MEOP	892	892
OXIDIZER INJECTION	PRE-BURNER	PRE-BURNER
IGNITION SYSTEM	HYPERGOLIC	HYPERGOLIC
THRUST CONTROL	EMA VALVES	EMA VALVES
COMBUSTION STABILITY	GASEOUS INJECTION	GASEOUS INJECTION
PRESSURIZATION SYSTEM	TRIDYNE	TRIDYNE
PRESSURE-FED VS PUMP-FED	PRESSURE-FED	PRESSURE-FED
REUSABLE VS EXPENDABLE	EXPENDABLE	REUSABLE
CONFIGURATION	SEPARATE TANK & CASE	SEPARATE TANK & CASE
MATERIALS		
TANK	GR/E +LINER	GR/E + LINER
CASE	GRAPHITE/EPOXY	GRAPHITE/EPOXY
OXIDIZER SUPPLY	SINGLE FEED TO EACH PORT	SINGLE FEED TO EACH PORT
THRUST VECTOR CONTROL	FLEX SEAL	FLEX SEAL

Figure 2-96. The results of the HPT system analyses and trade studies.



Dimension	ASRM size	1/4 size
Diameter (inches)	150	90
Length (inches)	877	679
Wall thickness (inches)	1.03	0.62
Weight (lbm)	33,217	8,229
MOP (psig)	892	892

Figure 2-98. A new-technology LO2 tank increases payload 20% for less cost.

Figure 2-99 presents the case and its characteristics. All the new enabling HPT occurs within the case, but the materials and manufacturing processes do not require technology development. The composite case will increase flight safety and reliability, and will improve performance for a lower cost, but existing steel cases are satisfactory for HPT.

Hypergolic ignition was selected during our analysis and study as an aid in assuring complete, smooth ignition.

LO2 injectors are recommended to eliminate the complex extra systems necessary to convert the LO2 to gas.

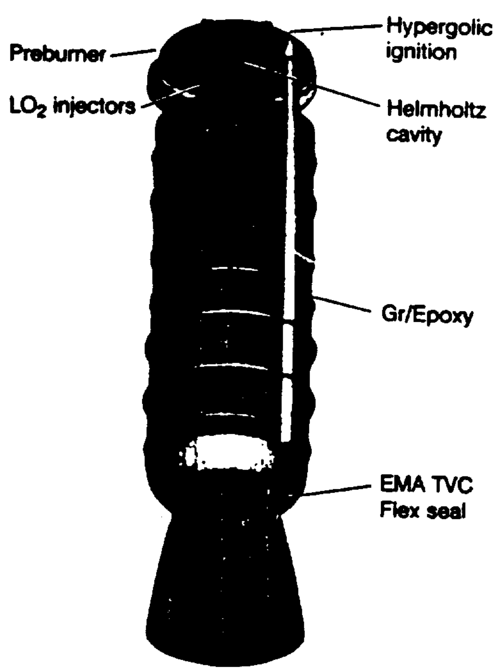
A preburner with Helmholtz chambers is included to improve combustion stability and scaleability.

A current-technology flex seal permits nozzle vectoring by the electromechanical actuators in development for the ALS.

2.6.2 END ITEM SPECIFICATION LEVEL REQUIREMENTS

2.6.2.1 Full-Size Booster. The mass properties and ballistic performance for the full-size HPT booster is delineated in Table 2-2. Figure 2-100 shows the full-size HPT booster thrust profile.

2.6.2.2 Quarter-Size Booster. The mass properties and the ballistic performance for the quarter-size HPT booster are delineated in Table 2-3. Figure 2-101 shows the quarter-size HPT booster thrust profile.



Dimension	ASRM size	1/4-ASRM size
Diameter (inches)	150	90
Length (inches)	682	460
Wall thickness (inches)	0.86	0.52
Case weight (lbm)	13,362	3,405
MEOP (psig)	743	743

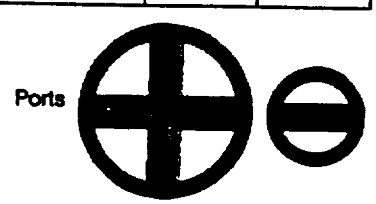


Figure 2-99. The combustion process occurs within the case.

Table 2-2. Mass properties and ballistic performance for full-size HPT booster.

MASS PROPERITIES

Overall Length	149.2 ft. (1790 in.)
Maximum Diameter	12.5 ft. (150 in.)
CASE	
Construction	Graphite/Epoxy
Grain Design	4-Port Wagon Wheel
Fuel	HTPB/ZN/GAP
Lenght	56.8 ft. (682 in.)
Diameter	12.5 ft. (150 in.)
Wall Thickness	0.86 in.
Weight	13,362 lbm
Fuel Weight	497,700 lbm
OXIDIZER TANK	
Construction	Graphite/Epoxy with Aluminum Liner
Oxidizer	LO2
Length	73.1 ft. (877 in.)
Diameter	12.5 ft. (150 in.)
Wall Thickness	1.03 in.
Weight	33,217 lbm
LO2 Weight	704,600 lbm
PRESSURIZATION SYSTEM	
Type	Tridyne
Weight	15,080 lbm
NOZZLE	
Type	Submerged Flex Bearing
Weight	7530 lbm
INSULATION	
Type	Silica filled Buna-N
Weight	6041 lbm

BALLISTIC PERFORMANCE (Figure 2-100)

Case MEOP	743 psig
Tank MOP	892 psig
Chamber Pressure (Avg.)	500 psia
Specific Impulse Actual (vac)	272.5 lbf sec/lbm
MaxThrust (vac)	3.52 Mlbf
Avg Thrust (vac)	2.59 Mlbf
Mixture Ratio (oxidizer/fuel)	1.27

Table 2-3. Mass properties and ballistic performance for quarter-size HPT booster.

MASS PROPERITIES

Overall Length	104 ft. (1248 in.)
Maximum Diameter	7.5 ft. (90 in.)
CASE	
Construction	Graphite/Epoxy
Grain Design	2-Port Wagon Wheel
Fuel	HTPB/ZN/GAP
Lenght	38.3 ft. (460 in.)
Diameter	7.5 ft. (90 in.)
Wall Thickness	0.52 in.
Weight	3405 lbm
Fuel Weight	128,500 lbm
OXIDIZER TANK	
Construction	Graphite/Epoxy with Aluminum Liner
Oxidizer	LO2
Length	56.6 ft. (679 in.)
Diameter	7.5 ft. (90 in.)
Wall Thickness	0.62 in.
Weight	8229 lbm
LO2 Weight	165,500 lbm
PRESSURIZATION SYSTEM	
Type	Tridyne
Weight	3577 lbm
NOZZLE	
Type	Submerged Flex Bearing
Weight	4,280 lbm
INSULATION	
Type	Silica filled Buna-N
Weight	3,364 lbm

BALLISTIC PERFORMANCE (Figure 2-101)

Case MEOP	743 psig
Tank MOP	892 psig
Chamber Pressure (Avg.)	500 psia
Specific Impulse Actual (vac)	274.7 lbf sec/lbm
Max. Thrust (vac)	880 klbf
Avg. Thrust (vac)	648 klbf
Mixture Ratio (oxidizer/fuel)	1.29

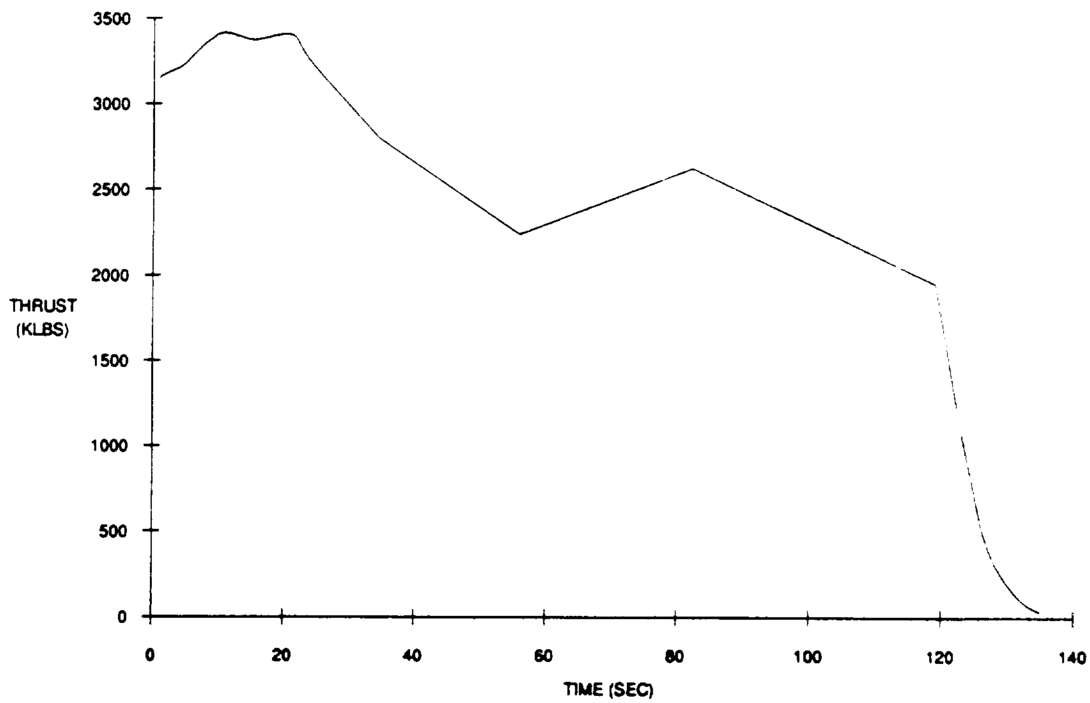


Figure 2-100. Full-size HPT booster thrust profile.

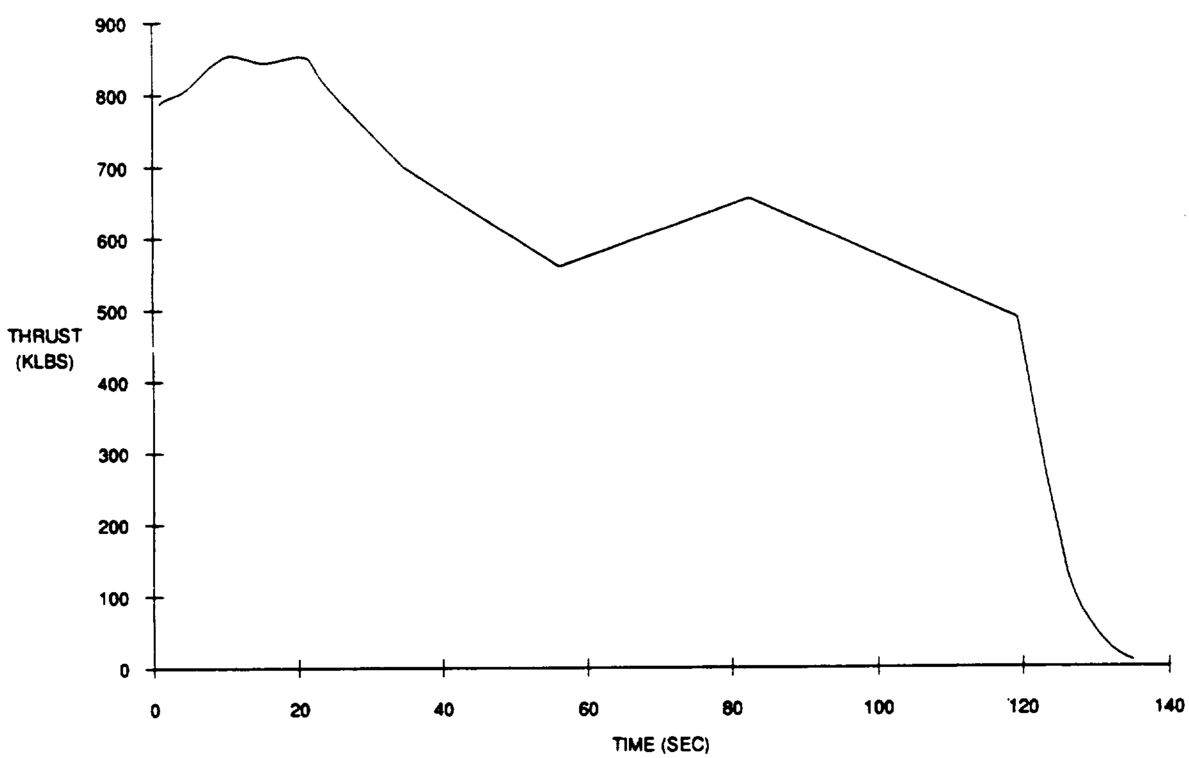


Figure 2-101. Quarter-size HPT booster thrust profile.

3.0 TECHNOLOGY ACQUISITION PLANS

HPT will be acquired in two years, as shown in Figure 3-1. The two-year schedule assumes an orderly progression through the tests, data reduction, and code revisions.

Sections 3.1 through 3.3 describe the technologies that enhance the attractiveness of HP. The enabling technologies are described in Volumes III and IV under the acquisition plans from each of our subcontractors. Acquisition of all enabling technologies would be completed within the scope of the schedule presented in Figure 3-1.

3.1 TRIDYNE PRESSURIZATION SYSTEM

The Tridyne pressurization system, Figure 3-2, was selected as the preferred system during the system analyses and trade studies documented in Sections 2.5.3 and 2.5.4. Figure 3-3 presents a summary of the pressurization systems studied in detail and the reasons for selecting the pressure-fed concept by Tridyne.

The Tridyne pressurization system is not an enabling technology for development of HPT. There are alternative pressurization concepts such as those studied in Figure 3-3 that incorporate current technology; they require only modifications for the specific application to successfully pressurize the liquid oxygen tank of a HRB. But the Tridyne pressurization is most desirable. It is very much an enhancing technology which readily integrates into the hybrid rocket booster to enhance its flight safety and reliability, low life cycle cost, and increased performance.

The recommended Tridyne system stores a He/O2/H2 mixture at high pressure and low temperature to minimize storage bottle size and weight. The discharged mixture flows through a catalytic bed that decomposes and heats the gas. The heated gas is used to pressurize the liquid oxygen ullage or another Tridyne bottle, which cascades into the ullage.

Rocketdyne performed considerable development work with Tridyne in the 1960s and 1970s. They were contracted by the U.S. Army to develop a Gun Breech Scavenger System. The system had a two-cubic-inch catalytic heater containing supported granular catalyst.

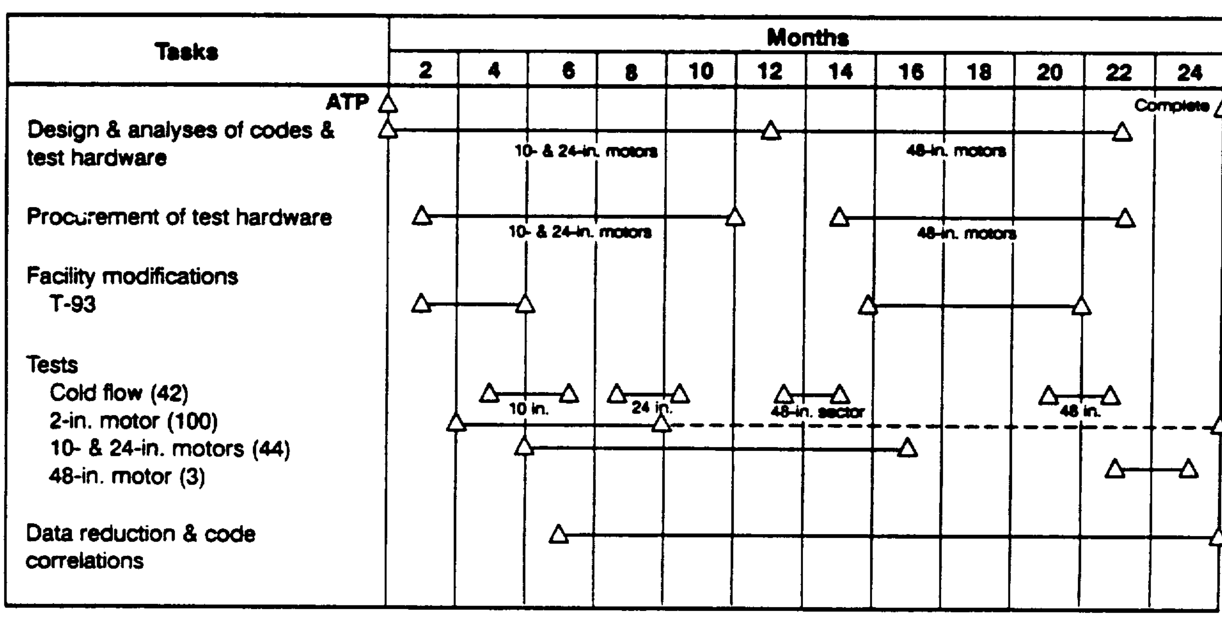


Figure 3-1. Two years are required to acquire the identified HPT.

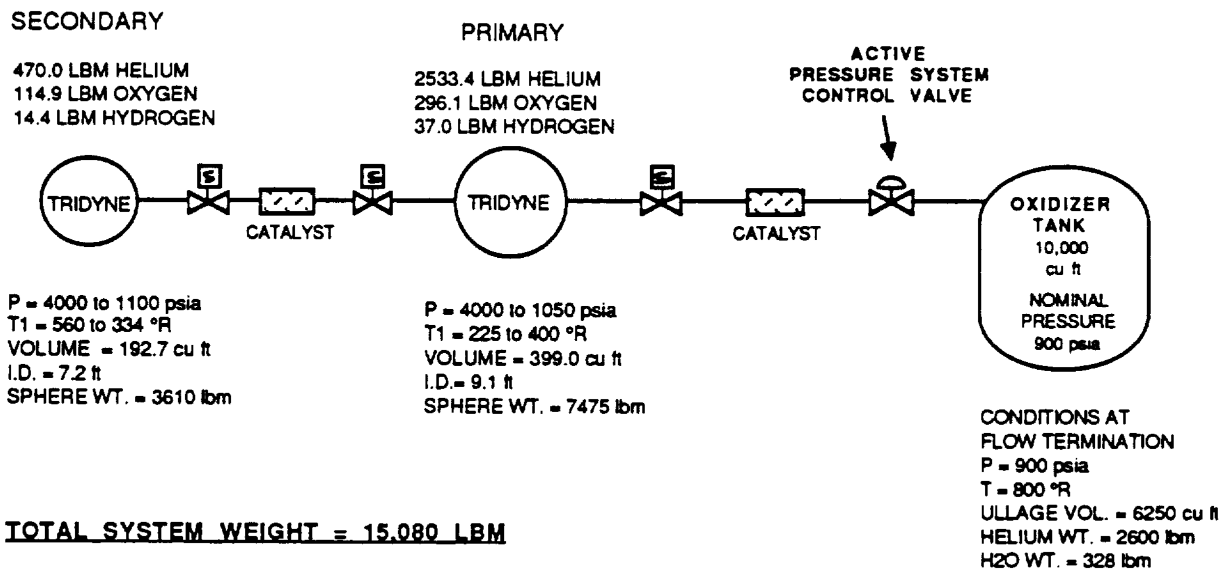


Figure 3-2. The Tridyne pressurization system was selected.

The safety of the usable mixtures was demonstrated by performing tests at pressures to 5000 psi and temperatures to 700 degrees rankine. These tests by NASA, NBS and Rocketdyne demonstrated that the planned operating region is safe. It is well separated from the detonability zone for both helium and nitrogen diluent mixtures.

There are three areas warranting further technology acquisition. These areas are:

- Refinement of codes
- Reliability of the catalytic reaction
- Effect of moisture in the ullage of the liquid oxygen tank

3.1.1 REFINEMENT OF CODES.

Current codes calculate the piping pressure loss using the Darcy-Weisbach equation with friction factors applied as for compressible fluids. Some of the algorithms require an iteration to compute the mean pressure for calculating pressure losses.

Desired additions include the static pressure at the injection inlet, the flow rate to the liquid oxygen tank, the storage bottle pressures and temperatures, and the heat transfer within the Tridyne bottle(s) and liquid oxygen tank. The capability to analyze one or more Tridyne bottles is also highly desirable.

Such an acquired code provides a flexible analysis tool. It also allows for simple change to upgrade computations and incorporate empirical correlations.

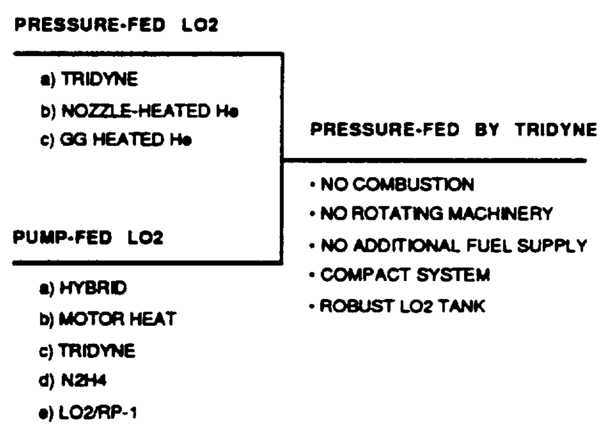


Figure 3-3. The Tridyne system proved to be the best of those studied to pressure feed LO2 to the injectors.

3.1.2 RELIABILITY OF THE CATALYTIC REACTION.

Further technology acquisition is recommended to verify reaction of the low-temperature Tridyne in large catalyst beds. The catalyst has been demonstrated down to 300 degrees rankine. It increases to 400 rankine at the completion of its expulsion.

While use of catalysts of the hybrid propulsion scale is common in the chemical process industry, catalyst acquisition also involves selection of a preferred catalyst support. A honeycomb support is recommended for a granular catalyst bed considering the high gas flow rates of interest.

3.1.3 EFFECT OF MOISTURE IN THE ULLAGE OF THE LIQUID OXYGEN TANK.

Acquisition tests are recommended to demonstrate the effect of moisture in the liquid oxygen tank. One product of the Tridyne reaction is steam at a very low percentage when compared to the allowable concentration of water in liquid oxygen. Condensation of the moisture from the ullage, to the extent of the heat transfer mechanisms between the gas/liquid interface, has been analyzed.

Acquisition testing would confirm the quantity of condensate, and identify its phase. If it is in the form of microscopic ice crystals, as predicted, there will be no adverse affects on the oxygen injection and combustion process.

The schedule for an austere program to acquire the Tridyne is shown in Figure 3-4. It includes seven months to modify the codes and three subsequent months to compare the code with test results. Ten months are shown for acquisition of the technology for a catalyst bed. This time period is incorporated into subscale system tests.

The subscale test rig is shown in Figure 3-5. Its one-bottle arrangement will provide the test results desired for a minimal amount of resources. It will verify the code, demonstrate low-temperature, high-loading catalyst operation, and disclose the effect of moisture in the ullage.

3.2 COMPOSITE LIQUID OXYGEN TANK

One promising area of technology reviewed during Phase I of the HPT program was graphite/epoxy filament-wound liquid oxygen tanks. The composite filament-winding process produces a high-pressure tank with

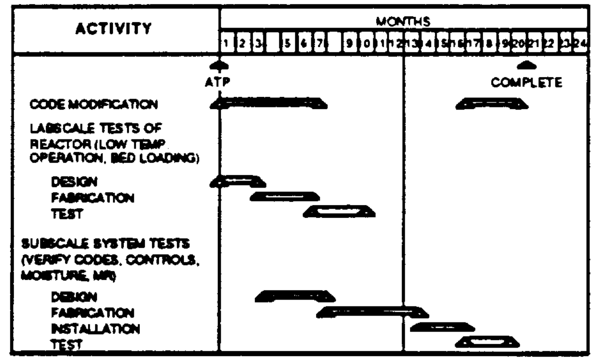


Figure 3-4. Twenty months are required to verify the feasibility of Tridyne pressurization systems.

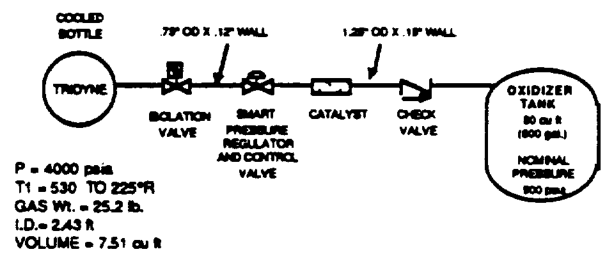


Figure 3-5. The subscale test rig is sufficient to verify feasibility.

many properties that are more attractive than those provided with aluminum.

The carbon fiber filaments identified for use in the composite-winding process have a tensile strength greater than even the popular Kevlar® material. When used in combination with specialized epoxy resin, the resulting structure has properties that exceed most metals.

The most attractive feature of graphite/epoxy LO2 tanks is their light weight. For the 10,000-cubic-foot LO2 tank being baselined for the ASRM-size hybrid booster, a 2219 aluminum tank has been estimated to weigh 88,800 pounds. A functionally equivalent graphite/epoxy tank would be as light as 33,200 pounds. This difference of over 55,000 pounds results in a potential increase of 9400 pounds of payload into orbit on the STS vehicle--almost a 20% boost due solely to using a composite tank.

An additional benefit of the graphite/epoxy tank concerns safety and reliability. Due to the nature of filament-based structures, filament orientation provides highly tailorable directional properties.

Additional material can be easily added during construction where needed. This can provide a greater safety factor where needed without incorporating excessive margin in already adequate regions.

The filament-winding process is also numerically controlled, which aids in precision wrapping, a feature that produces a process that is almost fully automated. Due to this the manufacturing process is less expensive, more repeatable, and faster than aluminum fabrication.

The required inspection of the final product becomes less involved, focusing on anomalies of the winding process rather than continual examination throughout the welding of an aluminum vessel.

Because the process is pre-programmed into the winding machine, there is a reduced labor force needed. These, factors, in addition to the comparable material cost, allow the final product to be completed for significantly less than a metal tank.

The proposed tank is composed of a thin, soft 1100 aluminum liner over-wrapped with IM-7 carbon filament in a 8551-1 epoxy resin. The aluminum liner is necessary to maintain a physical separation of the liquid oxygen from the composite material. This is needed for material compatibility reasons.

A soft aluminum is used because it is highly malleable, which reduces the tendency to pull away from the wrapping during the periods of high thermal strain mismatch. These periods occur when the materials are brought down to LO2 temperatures

3.2.1 EXISTING TECHNOLOGY. Current technology supports the use of Kevlar®-wrapped composite helium bottles on the General Dynamics Centaur. These bottles use a titanium liner, operate at up to 4,000 psig, and contain ambient temperature helium. This does not expose the vessel to cryogenic temperatures during use.

During manufacture, however, the wrapped bottles are pressurized at liquid nitrogen temperature to install a compressive stress into the liner. When warmed back to ambient, the containers and liners have a high stress built in, which both prevents cracks from forming in the liner, and prohibits small existing cracks from growing while under pressurized conditions.

Another current application of advanced composite technology is on the American Rocket Company (Amroc) vehicle. This rocket employs a pressurized, graphite-composite reinforced LO2 tank. The structure of this tank is a heavy-walled aluminum vessel with the surrounding graphite filaments used to provide added strength. Although this application uses composite technology in a cryogenic environment, the low mass of the composite is only partially exploited, because the aluminum liner mass is significant.

3.2.2 ACQUISITION PLAN. Although there are several highly desirable benefits associated with the use of graphite/epoxy for hybrid booster LO2 tanks, we see this technology as an enhancing one, not an enabling one. To take advantage of the desirable features of composite tanks, three different-sized tanks will be constructed: 18, 40, and 90 inches in diameter. We feel that these sizes will be ideal for investigation and allow acquisition of the new technology.

Several 18-inch tanks will be constructed to validate the concept and optimize the construction parameters. A 40-inch tank will be wrapped for high pressure and structural testing. The quarter-ASRM-size (90-inch) tank, the largest tank to be built in this program, will provide insight into tank wrap/liner scaling and processing.

There are several technical areas that need to be examined to acquire this technology, as summarized in Table 3-1.

First is to verify our advanced cryogenic construction process. This method enables the winding of a tank around a structurally unstable liner while providing a high level of built-in compressive stress.

The second area is to establish reliable quality assurance techniques. Non-invasive methods of structural verification will be necessary to allow large-quantity, low lead-time manufacture of the tanks.

The third acquisition area is in new materials. New epoxies, filament materials and processing techniques need to be investigated, because they become relevant. Further study will be focused on LO2-compatible materials to allow linerless tanks.

Table 3-1. Several technical areas will need to be examined.

- Verify thin-liner cryogenic manufacturing process
- Develop quality assurance techniques
- Examine new fiber/resin properties
- Establish optimum wrap and liner configurations
- Maximize process efficiency

The remaining areas involve the process of manufacture. Optimum wrap and liner configurations and improved process efficiency will have to be quantified to enable efficient design practices. A suggested schedule for the acquisition work is shown in Figure 3-6. Since this is an innovative process, testing and manufacture of the test items will be largely based upon current results and recent experience.

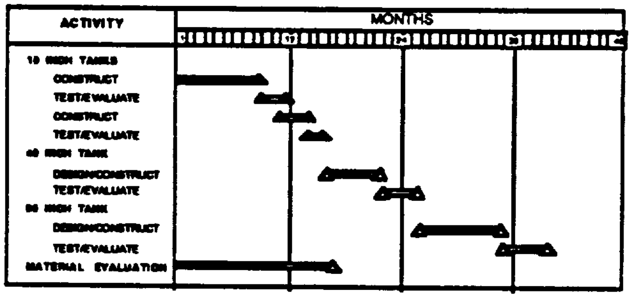


Figure 3-6. Graphite/epoxy LO2 tank technology acquisition schedule.

3.3 ELECTROMECHANICAL ACTUATORS

The use of electromechanical actuators (EMAs) for valve actuation and thrust vector control was determined to be highly desirable from a vehicle system viewpoint. The power, control, and actuation systems of EMAs were found to be of size and weight comparable to conventional pneumatic and gas generator-driven systems.

Benefits of EMAs can be realized in several areas. First, operation and check-out is greatly simplified and easily automated through the use of computer-based testing and evaluation schemes.

Second, EMA use allows the booster to remain in a safe condition on the pad prior to liftoff, not requiring pyrotechnics and pressurized control systems.

Third, EMAs allow the elimination of additional fluid systems that typically require a great deal of condition monitoring.

Fourth, and most critical, EMAs allow mission abort and restart. Whereas common pneumatic and gas-generator systems may be impossible to terminate, restart, or reuse without refurbishing or re-pressurizing, these capabilities are commonplace for electrical components. On a vehicle such as a hybrid booster where one of its more desirable features is potential restarts, this feature is necessary.

3.3.1 TECHNOLOGY ACQUISITION. Though desirable, this technology was determined to be an enhancing one, not an enabling one. Acquisition of EMA principles is currently being explored under an ALS Advanced Development Program (ADP) here at General Dynamics Space Systems Division.

Work is being done on 45- to 75-horsepower TVC actuators that meet the needs of the hybrid booster systems studied here. Additional discretionary funding is also being supplied for work on EMA valve actuation systems.

4.0 LARGE SUBSCALE MOTOR SYSTEM TECHNOLOGY DEMONSTRATION PLAN

This preliminary plan incorporates technologies developed in the acquisition plan into a single large subscale motor system. The purpose is to demonstrate the ability of the technologies and models to predict the performance, behavior, and other characteristics when used to scale to a large thrust representative of a booster application.

The plan includes a description of the motor system to be fabricated, the technologies to be integrated, the test plan including the number of units to be tested, required associated instrumentation, andthe schedule and cost. Potential government test facilities were surveyed and a recommended facility identified along with preparation/modification costs.

4.1 MOTOR SYSTEM

A 90-inch motor size will demonstrate the acquisition of the identified HPT. A 90-inch motor is recommended to demonstrate the acquisition of the identified hybrid propulsion technology. The 90-inch motor, Figure 4-1, is proposed in two configurations.

The initial configuration duplicates hybrid propulsion system having characteristics of the quarter-size HRB defined previously. The second configuration uses a 90-inch case extension to duplicate a port sector of the full-size hybrid propulsion system.

Both configurations have a maximum engine operating pressure of 743 psia and thrust of 880,000 pounds force. The one-size weighs 185,000 pounds mass, while the full-size weighs more at 260,000 pounds.

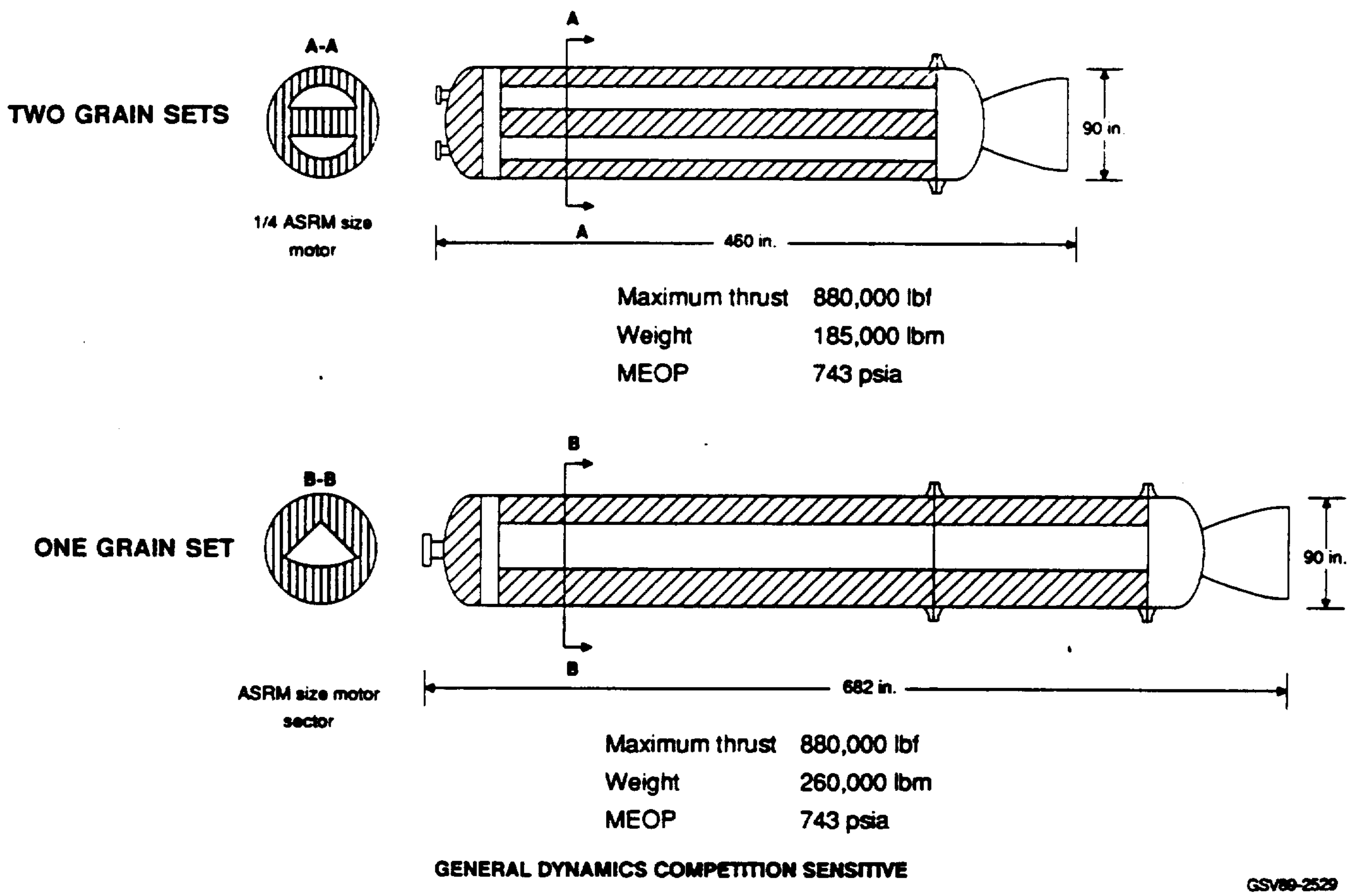


Figure 4-1. A 90-inch subscale motor system is recommended to demonstrate HPT.

The liquid oxygen supply simulator, Figure 4-2, requires no new technology. Due to the basic nature of the hybrid motor configuration, the liquid oxidizer supply system does not need to contain exotic components or be overly complex. Since a hybrid propulsion system requires only a single fluid, precise valve timing and synchronization are not necessary as with a liquid engine test rig. All the recommended hardware is commercially available, with no development or testing time needed.

For the proposed testing, a pressurized LO2 supply tank is required. A compressed gas source is shown as the pressurization system, but any functionally equivalent system would be acceptable.

Either an insulated or uninsulated liquid oxygen tank is necessary. An uninsulated tank is preferred, because this would provide liquid oxygen at the quality more representative of the flight booster.

4.2 TECHNOLOGIES DEMONSTRATED

All the enabling hybrid propulsion technologies would be demonstrated as well as those requiring characterization. These technologies all occur within the case assembly. They include grain ignition; flame holding/flooding; liquid oxygen injection; gas mixing, combustion and flow; grain regression; and combustion stability. Those being characterized include nozzle materials and configuration, case internal insulation, liquid oxygen flow control, the ignition type and sequencing, the oxygen conditioning within the case and the injector durability.

In all cases, the demonstration firings will verify that the codes were acquired during the previous acquisition phase.

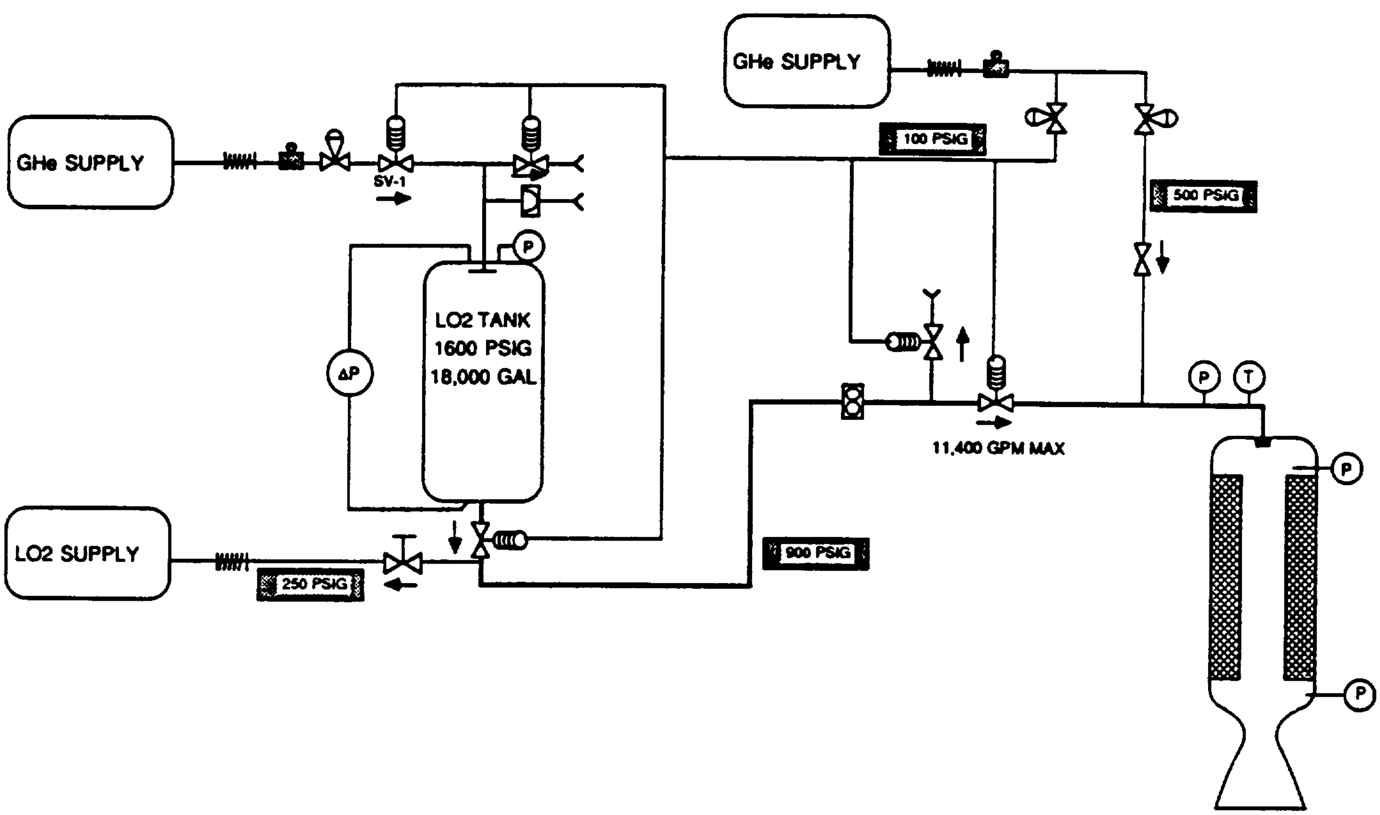


Figure 4-2. The LO2 supply simulator for the demonstration motor firings uses commercially available hardware.

4.3 TEST PLAN

A series of three firings are recommended to demonstrate the technology acquired. The first two tests will consume two grains configured as recommended for the quarter-ASRM-size hybrid propulsion system.

The test sequence will take advantage of a primary HP system characteristic, which is the ability to start and stop the firing at will. It is proposed that there be three firings, as shown in Figure 4-3. Grain/motor inspections will be made after each to verify that the demonstration is proceeding as predicted by the codes. The initial firing will duplicate the start sequence, the second the throttling thrust reduction, and the third the increased thrust followed by decrease and termination decay.

During each firing, the combustion volume will be "bombed" to verify that any pressure oscillations are damped. With proper design, the combustion process is stable.

The second firing sequence would duplicate the thrust time curve of the quarter-ASRM-size hybrid propulsion system, as shown in Figure 4-4. It would fire for the entire flight time without interruption to demonstrate proper operation of the motor as predicted by the codes. Any un-predicted condition that may occur during the initial interrupted series of firings would be demonstrated by this continuous firing.

An extension segment will be added for the third firing to accommodate a different grain geometry. With a length and port profile matching one on the four ports to be used in the ASRM-size motor, additional runs will be used to verify scaling for the next phase.

Because the operating parameters of the two configurations will be similar, use of the same stand is possible without major modification to either the support structure or the LO2 supply system. The firing sequence could follow the profile of either Figure 4-3 or Figure 4-4, with the former preferred including the "bombing." The "bombing" would verify stability for the increased length-over-diameter of the port for the 150-inch motor while maintaining the same diameter.

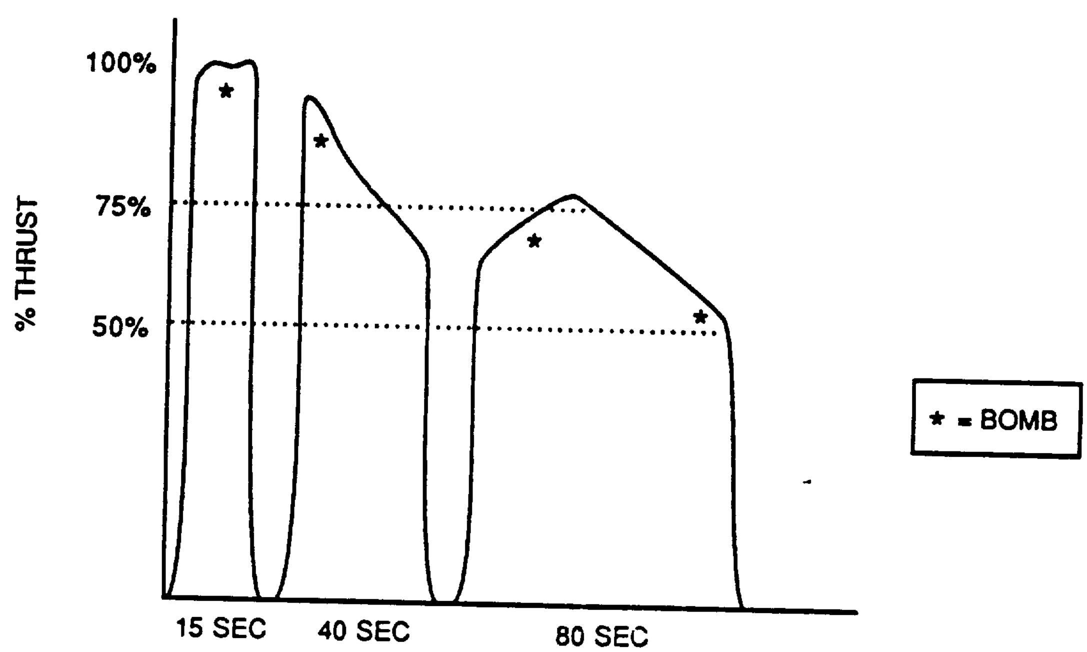


Figure 4-3. Three firings provide a progressive demonstration.

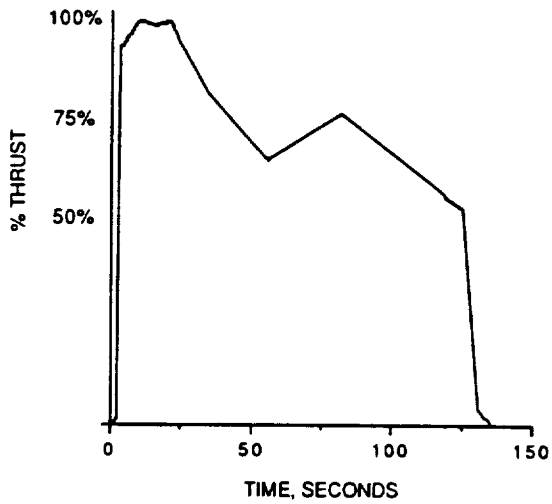


Figure 4-4. The second grain duplicates the flight profile.

It will require an estimated two years at a cost of $34.3 million to demonstrate the identified and acquired HPT, as shown in Figure 4-5. This includes the design, analysis, and procurement of the test hardware, the tests, and the data reduction and codes correlations. It does not include the LO2 system simulator or test facility.

The stands presented in Figure 4-6 were surveyed to accomplish the demonstration firings. All had suitable accommodations to support the motor and resist the thrust forces. All lacked a liquid oxygen supply system that could support the firings.

One stand that is particularly promising is the booster technology simulator located at MSFC (Figure 4-7). It will be modified to support planned booster propulsion tests. The facility will consist of a modified F-1 test stand adapted for use as a vertical firing (nozzle-down) stand for large booster testing. It will be altered to include a 4.055-cubic foot, heavy-walled LO2 tank, a high-capacity pressurization system, and the necessary structure to accommodate a booster of the quarter-ASRM-size. .

Also available on the stand will be a 100-ton crane which allows the erection of the 460-inch-long motor as a single unit. The 682-inch motor will be built up as a two-piece assembly, joined at the case extension interface. An all-new data collection/analysis and control system will also be in place, allowing motor characterization, regulation, and redline monitoring. This ideal test site is a first choice for HPT demonstration.

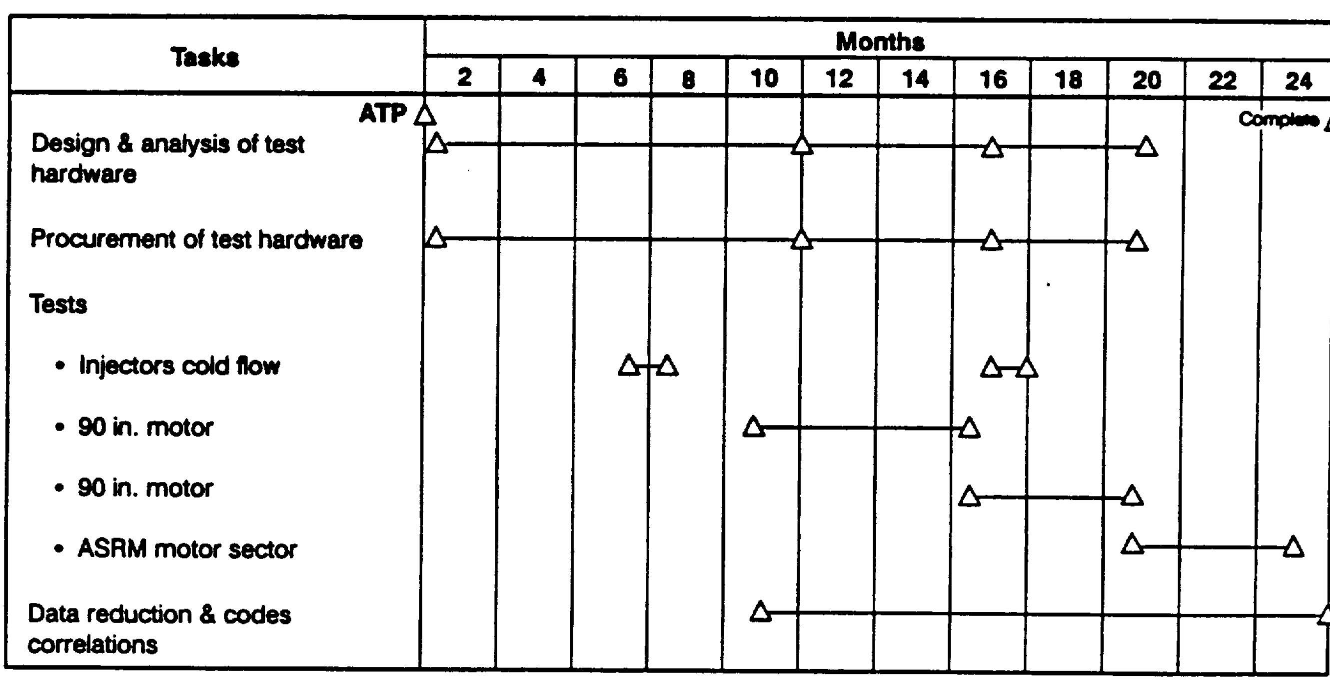


Figure 4-5. Two years are required to demonstrate the identified HPT.