HYBRID PROPULSION TECHNOLOGY PROGRAM

Phase I—Final Report
Volume I Executive Summary
Contract NAS8-37775

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NASA
National Aeronautics and Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812
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Volume I Executive Summary

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Prepared for: National Aeronautics and Space Administration
George C. Marshall Space Flight Center
Huntsville, AL 35812

Prepared by: Aerojet Solid Propulsion
Aerojet TechSystems
Martin Marietta Aerospace

c/o Aerojet Solid Propulsion
Box 15699C
Sacramento, CA 95813-1699
FOREWORD

This is the final report for Contract NAS 8-37775, a research technology study entitled "Hybrid Propulsion Technology Program Phase I." The study was performed for NASA-MSFC by Aerojet with vehicle effects analysis provided by Martin Marietta.

This report has been assembled in two volumes for clarity. Volume I is an executive summary with an overview of the study program, methodology of trade studies, study results, and Phase II and III planning.

Volume II is a compilation of detailed study charts with facing page annotation added as required for explanation.

The NASA-MSFC Study Manager was Ben Shackelford. Bob Friedman was the Aerojet Program Manager, supported by Art Kobayashi, Technical Advisory Group Manager; Don Culver, Technical Manager; Bill Barnette and Larry Hoffman, Solid and Liquid Component Project Engineers, and Brian Strickfaden, Life Cycle Costing. Craig Hansen, of MMAG supported Aerojet with vehicle integration studies.

The contract period of performance was 6 March 1989 through 23 October 1989.
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A Logical Scale-Up, Low Risk Approach to HRB Technology Demonstration
1.0 INTRODUCTION

The study program described herein was contracted to evaluate concepts of hybrid propulsion, select the most optimum, and prepare a conceptual design package. Further, this study required preparation of a technology definition package to identify hybrid propulsion enabling technologies and planning to acquire that technology in Phase II and demonstrate that technology in Phase III.

1.1 PROGRAM PHILOSOPHY

Our program was orientated to perform a study aligned with NASA priorities. The selection criteria were therefore prioritized as:

- Flight safety and reliability
- Low life cycle cost
- Performance
- Other important criteria
  - Availability (development risk, etc.)
  - STS compatibility

We evaluated two design philosophies for Hybrid Rocket Booster (HRB) selection, Figure 1. The first is an ASRM modified hybrid wherein as many components/designs as possible were used from the present ASRM design. The second was an entirely new hybrid optimized booster using ASRM criteria as a point of departure, i.e., diameter, thrust time curve, launch facilities, and external tank attach points.

We selected the new design based on the logic of optimizing a hybrid booster to provide NASA with a next generation vehicle in lieu of an interim advancement over the ASRM. The enabling technologies for hybrid propulsion are applicable to either and vehicle design may be selected at a downstream point (Phase III) at NASA’s discretion.

1.2 RESULTS

The completion of these studies resulted in the chart shown in Figure 2, ranking the various concepts of boosters from the RSRM to a turbopump fed (TF) hybrid. The scoring resulting from our Figure of Merit (FOM) scoring system (see Section 2.1) clearly shows a natural growth path where the turbopump fed solid liquid staged combustion hybrid provides maximized payload, minimum GLOW, and the highest safety, reliability, and low life cycle costing.

2.0 STUDY PROGRAM METHODOLOGY

We performed the study program in five logical steps based on the proven methodology.
The Turbo-Pump FED SLSC Hybrid is a Logical Growth Path to Provide:

- Flight Safety and Reliability
- Low LCC
- Performance

With:

Abort and Throttling

We then performed sensitivity and optimization studies and created a conceptual design incorporating the selected concepts. Finally, hybrid enabling technologies were identified and Technology Acquisition Plans (Phase II) and Demonstration Plans (Phase III) were defined.

### 2.1 FIGURE OF MERIT (FOM)

The FOM is the heart of the selection process, and we selected a well defined method in use at Aerojet. Our assignment of a numerical rating system prior to concept/component selection precludes bias and provides selection
3.1 Concept Definition

3.1.1 Requirements for Vehicle and Hybrid Propulsion System, Concepts and Screening

3.1.2 Hybrid Propulsion System Concept Trade Studies

3.1.3 Figure of Merit Methodology, Concept Ranking and Selection

3.1.4 Conceptual Design Definition of Selected Concepts

3.2 Technology Definition

Figure 3. Our Technical Approach Is Powerful and Logical

data automatically. By this use of the FOM system, Aerojet was able to make selections without influence of personal preference.

Using the baseline of the existing SRM program, five categories influencing the program were selected (Figure 4). The percentage each contributes to the whole is based on ALS data and becomes the maximum score points available in each category. Minimum (zero) points are the SRM baseline, and maximum are the ultimate to be expected. As an example, if the SRM has a payload capability of 24,950 kg (55,000 lb) then any booster with the same capability will get zero points. Conversely, if 38,550 kg (85,000 lb) lift is the ultimate then that unit will receive 14 points (the maximum in that category). Therefore, the FOM model contains LCC relative weighting factors that determine the maximum score a candidate may achieve in each cost category. It also contains weighting factor design parameter sensitivities. These two are functionally related to create the model that ties the concept parameter to the cost impact. A scoring format is included to sum the results of each category for each evaluated concept. The bases for the relative weighting factors and their design parameter sensitivities are the baseline system scenarios or requirements selected; that is, mis-
sion models, launch vehicle, and facilities. The result is an automated selection process that numerically rates the concept under study and provides numerical scoring for selection.

![Figure 4](image)

**Figure 4.** We Established the Category Cost Relationships vs Number of Missions From the Baseline Life Cycle Cost

2.2 EVALUATION PROCESS

The evaluation process screened from coarse to fine with immediate elimination of elements that did not pass (e.g., toxic propellants).

We considered propellants, combustion schemes, and propulsion subsystems to be three fundamental aspects of the rocket booster. We studied them in series in the order shown in Figure 5 (most to least fundamental) during the first three concept tasks in order to geometrically reduce the amount of work to be done. During design and technology tasks these distinctions collapsed and all work was done in parallel.

![Figure 5](image)

**Figure 5.** Our Five Step Methodology Uses Series and Parallel Processing as Appropriate

Our approach included an early yes/no type qualitative screening of developed concepts and a subsequent quantitative selection, based on scores computed with life cycle cost and payload to LEO data (see Figure 6). Your

![Figure 6](image)

**Figure 6.** Aerojet Study Criteria Concur With NASA MSFC HRB Priorities
priorities were considered in the screening process and some during selection.

We performed eleven selection studies to identify the best HRB concept for eight scenarios. Nine of the studies results are shown in Figure 7. Two additional ones showed that small HRBs and reusable HRBs score more poorly than large expendable ones, whereas a recoverable engine module scenario scored better. The chart shows that all scenarios need the same design for best scores, except small HRBs will be cylindrical their entire length, whereas our large HRBs have a short tapered section just ahead of the aft skirt. All scenarios use eight turbopumps and thrust chambers to maximize the score of our solid liquid staged combustion concept, which burn LO₂ and a solid hydrocarbon fuel rich solid propellant in the aft-mounted TCAs.

The results of the studies are shown in the nine charts of Figure 8.

<table>
<thead>
<tr>
<th>Scenarios</th>
<th>Best Scores</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>1</td>
</tr>
<tr>
<td>Reusable</td>
<td>No</td>
</tr>
<tr>
<td>No. HRB Flights</td>
<td>2</td>
</tr>
<tr>
<td>Flight Rate</td>
<td>1/wk</td>
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<table>
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<tr>
<th>Concept Selections</th>
<th></th>
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<tbody>
<tr>
<td>Level 1 Propellants</td>
<td>LO₂ + #8</td>
</tr>
<tr>
<td>Level 2 Combustor</td>
<td>SLSC (D)</td>
</tr>
<tr>
<td>Level 3 Feed System</td>
<td>TF/EBBC</td>
</tr>
<tr>
<td>Nozzle Exit Pressure</td>
<td>41.37 kPa (6 psi)</td>
</tr>
<tr>
<td>No. TCAs, 0 &quot;Out&quot;, HRB</td>
<td>4</td>
</tr>
<tr>
<td>No. TPAs/HRB</td>
<td>4</td>
</tr>
<tr>
<td>No. Solid Cases/HRB</td>
<td>1</td>
</tr>
<tr>
<td>Solid Case Shape and Tank Shape</td>
<td>Coni-Cyl, Coni-Cyl, Cyl, Cyl, Coni-Cyl, Coni-Cyl, Cyl, Cyl</td>
</tr>
<tr>
<td>Tank Shape</td>
<td>Cone/Cyl, Rev. Hd.</td>
</tr>
<tr>
<td>TCA Cooling (Throat)</td>
<td>LO₂ Regen</td>
</tr>
<tr>
<td>Tank Pressure</td>
<td>Autog. *</td>
</tr>
</tbody>
</table>

*Turbo Exhaust Bleed - No Heat Exchanger or Regulator Required

Figure 7. Task 3 Concept Selection Summary
We evaluated solid propellants for use with LO₂ and LO₂ densified with solidified H₂O₄ and H₂O₂ particulates. One is pure fuel, and others are fuel with a small amount of solid oxidizers. We selected pure fuel 8B, a PEBC hydrocarbon and a fuel-rich selection No. 8, both with LO₂. It is the same as No. 8B with solid oxidizers and HCl scavengers added to the PEBC hydrocarbon. H₂O₂ had been screened out on the basis of safety.

We selected the solid/liquid staged combustion scheme, because it had nearly twice the score of the best "single stage combustion" hybrid. The SLSC version with the hybrid gas generator did not score as well as the simpler one with the fuel-rich solid gas generator (solid case). All candidates used LO₂ with either No. 8 or No. 8B solid propellants.

Turbopump fed HRBs scored much better than pressure fed designs when the turbine is driven with bleed gases (not by gases to pass through the injector). Pressure fed variants suffered from low payload delivery to LEO, because of heavy tankage and pressurization weights. The topping cycle score was lower than the bleed cycle because its low specific impulse hurt its payload capability. The low Iₚₑₑ is caused by the relatively poor combustion efficiency of a gas/gas injector when used with O₂/hydrocarbon systems.

Figure 8. HRB Concept Scoring, Sheet 1 of 3
We selected four thrust chamber assemblies per large HRB, because the only numbers that package well at the vehicle base are 1, 3, 4, and 7. Four has the highest score of these candidates and is within ~2% of the score of the unworkable five TCA options. Total TCA weight drops as the number increases, minimizing at about 4 or 5/HRB. Learning curve effects also favor use of a greater number of identical TCAs. Use of multiple TCAs also allows operation with failed TCAs if the system concept properly accounts for this factor.

We selected four turbopump assemblies per large HRB, because they have the best score, and we get a one to one correspondence with four thrust chambers. Scores are higher for four TPAs, because their total weight minimizes at 4 or 5 and the learning curve reduces production costs of identical units.

We selected large area ratio nozzles, because they score much better or a little better than small nozzles, depending upon the design and use scenario. For our selected design, turbopump fed, the large nozzle advantage can be as large as 15 to 20%. More frequent flights accentuate the payload carrying advantage of large nozzled boosters. The smaller TF and PF nozzles fit the current mobile launch platform (for single nozzled HRBs), and the large ones expand exhaust gases to the generally accepted "best" value of 41.37 kPa (6 psia.)
For high flight rates, expendable HRBs outscore reusable ones, largely because of payload carrying differences. For lower flight rates the scores are equal. These results apply to both the selected turbopump fed and pressure fed HRBs. HRB reuse does not appear to offer many advantages, because refurbishment costs are high and learning curve production cost savings are not realized.

Reusable engine modules outscore expendable engines, because only relatively small, lightweight, and high valued HRB elements are recovered and refurbished. Thus, HRBs ought to be designed with recovery module integration in mind.

Use of two large HRBs per STS is favored over eight small ones by nearly 2 to 1 on the basis of scores. Small HRBs require more assembly hardware, add drag, and increase the amount of tankage and case hardware to be built along with their weights. Payload carrying losses reduce small HRB desirability considerably. Thrust chamber and turbopump development and production costs are not affected, because they are the same units in either scenario.
2.3 HRB CONCEPT SELECTION

The result of our selection system is a solid/liquid staged combustion cycle, pump fed, expander bleed burn-off cycle as shown in the flow schematic in Figure 9. The main features and benefits of this design are noted as is the operational sequence. Our concept design specification is summarized in Figure 10.

3.0 CONCEPT DESIGN

During the concept design phase of the study, we continued to prioritize the same criteria as we used in the scoring/selection process, i.e.

- Flight safety and reliability
- Low life cycle cost
- Performance
- Other important criteria
- Availability (development risk, etc.)
- STS compatibility

HRB Operational Sequence is as follows:

- Chill Down and Bleed in the LO₂ Pump and Injector With Bleed Valve
- Open Facility GN₂ Valve to Spin Turbo-pump With Turbine Bypass Valve Closed
- IGNITE Solid Propellant Grain
- Combustion in Thrust Chamber Begins when LO₂ and Solid Grain Fuel-rich Warm Gases Meet. LO₂ Bleed Flow in Regenerative Cooling Jacket Receives Heat
- Turbine Receives Heated O₂ and Flashes to GO₂ Drive Fluid in Nozzles. GO₂ Turbine Exhaust Follows N₂ into Burnoff Manifold at € ≤ 3.0 in Rocket Nozzle and Forward to LO₂ Tank Ullage at 1.72 MPa (250 psia). Fuel-rich Boundary Layer Burns Off in Nozzle With GO₂ Turbine Exhaust
- LO₂ System Bootstraps as Solid Grain Fully Pressurizes. Remove Facility GN₂ Line
- Thrust is Controlled With Turbine Bypass Valve That Prevents Regenerative COOLing/Flow Loss
- O₂/F Mixture Ratio is Controlled With Flow Control Plate Forward of Gas / Liquid Injector
- TPA Provides Alternator Power for Valve and TPC Actuators. Ablative Nozzle is Attached With a Flex Seal
- Near End of Operation the LO₂ Ullage Pressure Control Valve is Closed to Let Ullage Pressure Drop to Reduce Tank Weight at Burn-out and Tank Stiffness
- Shut Off LO₂ When Staging and Open Control Plate to Extinguish Solid Propellant

Figure 9. Our HRB Turbine Drive Cycle is Expander Bleed Burnoff Cycle (EBB)
- General Data
  - Propellants: Fuel Grain No. 8, Sat. HC [PEBC] and LO₂
  - Total [4 TCAs] MPL Thrust at Sea Level, 12.24 MN (2.75247 Mlb)
    MPL Thrust at Vacuum, 14.01 MN (3.14874 Mlb)
  - Combustion Scheme: Solid/Liquid Staged Combustion (SLSC)
  - Turbopump Drive Cycle: Expander Bleed Burnoff Cycle (EBB)
  - Gaseous O₂ Autogenous LO₂ Tank Pressurization From Turbine Exhaust
    - LO₂-Cooled Thrust Chamber
  - Dual Ignition System—Oxidizer Rich Liquid or Solid at Forward End of Grain
  - Electromechanical-Actuated TVC System With FlexSeal Mounted Nozzle
  - Turbopump Driven Alternator
  - All Hard Feed and Pressurization Lines and Engine Mounts
  - Solid Case Aft Head Is Engine Recovery Module Structure

- Design Point Data:
  - MPL TCA Pc, 11.72 MPa (1,700 psia)
  - Nozzle Area Ratio, 26.2 - Rao Contour
  - MPL Exit Pressure, 41.37 kPa (6 psia)
  - Throat Diameter, 45.7 cm (18 in.)
  - Exit Diameter, 233.7 cm (92 in.)
  - Combustion Mixture Ratio (CMR), 2.60
  - Liquid/Solid Mixture Ratio (LSMR), 1.90
  - MPL Iₛₚ VAC and Iₛₚ SL, 303 and 265 sec
  - 4 TCAs/TPAs Total Design Weight, 8346 kg (18,400 lbm)
  - Silica Phenolic/Nonmetallic Honeycomb Nozzle, GO₂ Cooled at ε = 3
  - Turbine Inlet Pressure and Temperature, 11.31 MPa (1,640 psia) and 478°K (860°F)
  - LO₂ Pump Outlet Pressure, 14.06 MPa (2,040 psia)
  - Solid Grain MPL Pressure, 12.89 MPa (1,870 psia)

Figure 10. Aerojet HRB TF Engine Concept Design Specification Summary

Figures 11 through 14 show our HRB concept with the design features that fulfill the criteria outlined above.

Figure 15 is an overview of the conceptual booster with the design features that we have incorporated to create an optimized booster. Details of the design, including engine layout, are included in the technical volume of this report.
Figure 11. Our Design Provides Many Safety and Reliability Benefits – Safer Concept and Engine Out Operation
Figure 12. We Have Provided Life Cycle Cost Benefits – Lowest Weight and Simplest Concept
Figure 13. Payload Performance Benefits Are Derived From Our Design - Lowest Weight and Highest Isp
- Retains Basic Launch Facility Configuration
- Maintains ET Attach Points
- Reduces Aerodynamic Drag
- Provides Increased Payload Capability

Figure 14. The Design is Compatible With the Space Transportation System

4.0 PLANNING: PHASE II - TECHNOLOGY ACQUISITION AND PHASE III - TECHNOLOGY DEMONSTRATION

In conjunction with our conceptual design, we have identified enabling technologies to bring an HRB to fruition. These are outlined in Figure 16.

Further, we evaluated schedules, costs, and test requirements for Phases II and III. The details of these studies may be found in the last section of the technical volume. In addition, we surveyed our in-house test capabilities and the test capabilities of government owned facilities.

The test planning for Phases II and III is summarized in Figure 17. We have selected thrust scaleup ratios that decrease as size increases. This reduces scaleup risk and provides a logical pattern of data throughout the range of potential application of the hybrid booster.

The test program for the Phase II 31 to 356 kN (7 to 80 klbf) units will be performed at the Aerojet Sacramento, CA facility where we have in-house capability requiring minimum modification.

The 1.8 MN (400 klbf) large-subscale demonstration will best fit the NASA MSFC Test Stand 116 capability which will be completely modified and will be available during Phase III.

Testing of the full sized 3.6 MN (800 klbf) HRB should be planned at MSFC on the planned/modified FI stand.
Spray On Foam Insulation

Forward Separation Motors and Nose Cap (Rotated View)
LiAl Liquid Oxygen Tank Wall

3.81 m (150 in.) Diameter

Forward Mount Ring (Thrust Takeout) (at SRB Location)

External Tank (Rotated into View)

47.55 m (156 ft) Overall Length
Hemispherical Forward Solid Case Head

Nested Aft Tank Head (Insulated)

Insulated, Single LO₂ Feedline From Tank Low Point

Cylindrical Solid Case — Forward Section

Removable Insulation

Igniters and Grain Ports

Solid Propellant Grain
- High Surface Area
- Low Gas Velocity
- Extinguishable

Aft Mount Ring at SRB Location
FOLDOUT FRAME

Rigid Engine Inlet Ducts (4)

Area Fins Aft

Engine Inlet Grain Scallops (4)

Throttleable Rigidly Mounted Engine With LO₂ Turbopump (4)

Heat Shield

4.95 m (195 in.) Max Dia
Conical Graphite Hoop Wrapped Steel Solid Case — Finocyl Section
+ Reduces Drag
+ Increases Stiffness

Single Case Joint Without Continuous Internal Insulation
+ Face Seals
+ Not in Bending Moment Load Path
+ Pyro-separation for Engine Module Recovery (Optional)

LO₂ Pressurization Manifolding and

Hemispherical Aft Solid Case Head

Aft Separation Motors on Steel Aft Skirt —
6.73 m (265 in.) Max Dia (Rotated into View)
Figure 15. SLSC HRB Features
1. Solid Propellant Gas/Liquid Injector (Gas/Liquid Injectors Successfully Tested)
   - Benefit: > 15% Higher Combustion Efficiency vs Forward Injection; Improves Isp, Weight, Cost, and Payload

2. Fuel-Rich Propellant and Ignition (Similar Propellant Successfully Tested)
   - Benefit: Provides for Reduced LCC and ~20% P/L Advantage of SLSC Concept

3. Fuel Rich Gas Control Plates (Routine With Fixed Plates)
   - Benefit: Improved Both P/L and Cost:
     - Allows Safe Aborts With TCA Out
     - Provides Independent MR Control for Improved Propellant Utilization
     - Increases Isp by Providing Uniform Gas Flow to Injectors
     - Protects Injector
     - Reduces Development Cost (Ignition and Stability)

4. GO2 Bleed Burnoff in Nozzle (Routine Without Combustion)
   - Benefit: Improves Both P/L and Cost
     - Renders Low Cost Cycle Feasible
     - Reduces Turbine Bleed Isp Loss
     - Protects Flex Seal and Cools Nozzle

Figure 16. We Have Selected and Prioritized Our HRB Technology

<table>
<thead>
<tr>
<th>HRB Project Phase</th>
<th>Engine Vacuum Thrust Level</th>
<th>Thrust Scale-Up Ratio</th>
<th>Test Duration</th>
<th>Duration Scale-Up Ratio</th>
<th>Solid Case</th>
<th>Purpose</th>
</tr>
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<tbody>
<tr>
<td>II.a.</td>
<td>31 kN (10 klbf)</td>
<td>12.0</td>
<td>4 sec</td>
<td>3.0</td>
<td>BATES Motor 0.305 m (12 in.) dia</td>
<td>Solid Propellant Injector (Performance)</td>
</tr>
<tr>
<td>II.b.</td>
<td>356 kN (80 klbf)</td>
<td>5.0</td>
<td>12 sec</td>
<td>3.0</td>
<td>Super BATES Motor 0.711 m (28 in.) dia</td>
<td>Solid Propellant Gas Control Plate Bleed Burnoff (Performance)</td>
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<tr>
<td>III.</td>
<td>1.8 MN (400 klbf)</td>
<td>2.0</td>
<td>36 sec</td>
<td>3.5</td>
<td>Stage 2 Peacekeeper Motor 2.34 m (92 in.) dia</td>
<td>Cold GO2 Turbine LO2 Cooled TCA SS Splitline TVC</td>
</tr>
<tr>
<td>Development and Production (Large HRB for STS)</td>
<td>3.6 MN (800 klbf)</td>
<td>2.0</td>
<td>128 sec</td>
<td>3.5</td>
<td>Production Motor 3.71 m (146 in.) dia</td>
<td>Flight</td>
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Figure 17. A Logical Scale-Up, Low Risk Approach to HRB Technology Demonstration