FOREWORD

This final report, Volume II, Book 2 -- OTV Concept Definition, was prepared by Martin Marietta Denver Aerospace for NASA/MSFC in accordance with contract NAS8-36108. The study was conducted under the direction of NASA OTV Study Manager, Mr. Donald R. Saxton, during the period from July 1984 to October 1985. This final report is one of nine documents arranged as follows:

Volume I  Executive Summary
Volume II  OTV Concept Definition and Evaluation
           Book 1  Mission and System Requirements
           Book 2  OTV Concept Definition
           Book 3  Subsystem Trade Studies
           Book 4  Operations
Volume III  System and Program Trades
Volume IV  Space Station Accommodations
Volume V  Work Breakdown Structure and Dictionary
Volume VI  Cost Estimates
Volume VII  Integrated Technology Development Plan
Volume VIII  Environmental Analyses
Volume IX  Study Extension Results

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Table of Contents

1.0 Introduction

2.0 Requirements Overview

3.0 Concept Definition

4.0 Configuration Trades & Analyses

  4.1 Staging/Propellant Trade
  4.2 Retrieval Trade
  4.3 Main Engine Trades
  4.4 General Arrangement Trades & Analyses
  4.5 Redundancy/Man-rating
  4.6 Subsystem Selection Summary

5.0 Concept Selection

  5.1 High Potential Cryo Selection
  5.2 High Potential Storable Selection

6.0 Selected Concepts Definition

  6.1 High Potential Concept Definition -- Cryo
    6.1.1 Ground-based Cryo
      6.1.1.1 General Arrangement
      6.1.1.2 Subsystem Summary Description
        6.1.1.2.1 Aeroassist
        6.1.1.2.2 Propulsion
        6.1.1.2.3 Structures and Mechanisms
        6.1.1.2.4 Avionics
        6.1.1.2.5 Thermal Control
      6.1.1.3 System Weight Summary -- Ground-Based Cryo
      6.1.1.4 Performance on Model Missions
    6.1.2 Space-Based Cryo Family
      6.1.2.1 General Arrangement
        - Initial Space-Based OTV
        - Growth Space-Based OTV
        - Two Stage Space-Based OTV
      6.1.2.2 Subsystem Change Summary
        6.1.2.2.1 Aeroassist
        6.1.2.2.2 Propulsion
        6.1.2.2.3 Structures & Mechanisms
        6.1.2.2.4 Avionics
        6.1.2.2.5 Thermal Control
      6.1.2.3 Dry Wt. Breakdown & Performance Wt. Summary
        - Initial Space-Based OTV
        - Growth Space-Based OTV
        - Two Stage Space-Based OTV
      6.1.2.4 Performance on Model Missions

5-1

6-1

6-2

6-1

6-1

6-1

6-1

6-1

6-5

6-9

6-11

6-16

6-18

6-23

6-25

6-25

6-30

6-30

6-31

6-36

6-39

6-43

6-44

6-54
<table>
<thead>
<tr>
<th>Section</th>
<th>Title</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.2</td>
<td>High Potential Concept Definition--Storable</td>
<td>6-57</td>
</tr>
<tr>
<td>6.2.1</td>
<td>Ground-Based Storable</td>
<td>6-57</td>
</tr>
<tr>
<td>6.2.1.1</td>
<td>General Arrangement</td>
<td>6-57</td>
</tr>
<tr>
<td></td>
<td>ACC Configuration</td>
<td>6-57</td>
</tr>
<tr>
<td></td>
<td>Payload Bay Configuration</td>
<td>6-57</td>
</tr>
<tr>
<td>6.2.1.2</td>
<td>Subsystem Change Summary</td>
<td>6-59</td>
</tr>
<tr>
<td>6.2.1.2.1</td>
<td>Aeroassist</td>
<td>6-59</td>
</tr>
<tr>
<td>6.2.1.2.2</td>
<td>Propulsion</td>
<td>6-61</td>
</tr>
<tr>
<td>6.2.1.2.3</td>
<td>Structures &amp; Mechanisms</td>
<td>6-64</td>
</tr>
<tr>
<td>6.2.1.2.4</td>
<td>Avionics</td>
<td>6-66</td>
</tr>
<tr>
<td>6.2.1.2.5</td>
<td>Thermal Control</td>
<td>6-70</td>
</tr>
<tr>
<td>6.2.1.3</td>
<td>System Weight Summary - Ground-Based Storable</td>
<td>6-70</td>
</tr>
<tr>
<td></td>
<td>ACC Configuration</td>
<td>6-70</td>
</tr>
<tr>
<td></td>
<td>Payload Bay Configuration</td>
<td>6-70</td>
</tr>
<tr>
<td>6.2.1.4</td>
<td>Performance on Model Missions</td>
<td>6-77</td>
</tr>
<tr>
<td>6.2.2</td>
<td>Space-Based Storable Family</td>
<td>6-78</td>
</tr>
<tr>
<td>6.2.2.1</td>
<td>General Arrangement</td>
<td>6-78</td>
</tr>
<tr>
<td></td>
<td>53K-lb PKM</td>
<td>6-80</td>
</tr>
<tr>
<td></td>
<td>25K-lb AKM</td>
<td>6-80</td>
</tr>
<tr>
<td></td>
<td>90K-lb PKM</td>
<td>6-80</td>
</tr>
<tr>
<td></td>
<td>53K-lb AKM</td>
<td>6-80</td>
</tr>
<tr>
<td></td>
<td>Lunar Cryo PKM</td>
<td>6-80</td>
</tr>
<tr>
<td>6.2.2.2</td>
<td>Subsystem Change Summary</td>
<td>6-85</td>
</tr>
<tr>
<td>6.2.2.2.1</td>
<td>Aeroassist</td>
<td>6-85</td>
</tr>
<tr>
<td>6.2.2.2.2</td>
<td>Propulsion</td>
<td>6-87</td>
</tr>
<tr>
<td>6.2.2.2.3</td>
<td>Structures &amp; Mechanisms</td>
<td>6-90</td>
</tr>
<tr>
<td>6.2.2.2.4</td>
<td>Avionics</td>
<td>6-95</td>
</tr>
<tr>
<td>6.2.2.2.5</td>
<td>Thermal Control</td>
<td>6-99</td>
</tr>
<tr>
<td>6.2.2.3</td>
<td>System Weight Summary - Space-Based Storable</td>
<td>6-99</td>
</tr>
<tr>
<td>6.2.2.4</td>
<td>Performance on Model Missions</td>
<td>6-108</td>
</tr>
<tr>
<td></td>
<td>Space-based Storable</td>
<td>6-108</td>
</tr>
<tr>
<td>6.3</td>
<td>Multiple Payload Carrier</td>
<td>6-112</td>
</tr>
<tr>
<td>6.4</td>
<td>Evolutionary Strategy</td>
<td>6-116</td>
</tr>
<tr>
<td>6.4.1</td>
<td>Cryo/Storable Resolution</td>
<td>6-117</td>
</tr>
<tr>
<td>6.4.2</td>
<td>Alternative Cryo Evolutionary Candidates</td>
<td>6-119</td>
</tr>
<tr>
<td>6.4.3</td>
<td>Program Selection</td>
<td>6-123</td>
</tr>
<tr>
<td>6.5</td>
<td>Development Schedule</td>
<td>6-134</td>
</tr>
<tr>
<td>6.6</td>
<td>Summary DDT&amp;E Cost</td>
<td>6-139</td>
</tr>
<tr>
<td>7.0</td>
<td>Reference Documents &amp; Glossary</td>
<td>7-1</td>
</tr>
</tbody>
</table>
# LIST OF FIGURES

<table>
<thead>
<tr>
<th>FIGURE #</th>
<th>TITLE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0-1</td>
<td>Vehicle Concept Definition Approach</td>
<td>1-1</td>
</tr>
<tr>
<td>2.1-1</td>
<td>Design Driver Missions</td>
<td>2-1</td>
</tr>
<tr>
<td>2.1-2</td>
<td>Time Phasing of Mission Requirements</td>
<td>2-3</td>
</tr>
<tr>
<td>2.2-1</td>
<td>OTV's Architectural role is HEO Truck</td>
<td>2-3</td>
</tr>
<tr>
<td>2.2-2</td>
<td>Ground-Basing Imposes Unique Design Constraints</td>
<td>2-5</td>
</tr>
<tr>
<td>2.2-3</td>
<td>Space-Basing Favors More Efficient OTV Design</td>
<td>2-6</td>
</tr>
<tr>
<td>3.0-1</td>
<td>Candidate Propellant Performance</td>
<td>3-1</td>
</tr>
<tr>
<td>3.0-2</td>
<td>Efficient Aeroassist in a Key Design Objective</td>
<td>3-3</td>
</tr>
<tr>
<td>3.0-3</td>
<td>Configuration Options</td>
<td>3-4</td>
</tr>
<tr>
<td>3.0-4</td>
<td>Main Propulsion Candidates</td>
<td>3-5</td>
</tr>
<tr>
<td>4.1-1</td>
<td>Initial Concept Screening Criteria</td>
<td>4-1</td>
</tr>
<tr>
<td>4.1-2</td>
<td>Ground-Based Cryogenic Screening Results</td>
<td>4-2</td>
</tr>
<tr>
<td>4.1-3</td>
<td>Space-Based Cryogenic Screening Results</td>
<td>4-4</td>
</tr>
<tr>
<td>4.1-4</td>
<td>Ground-Based Storable Screening Results</td>
<td>4-4</td>
</tr>
<tr>
<td>4.1-5</td>
<td>Space-Based Storable Screening Results</td>
<td>4-5</td>
</tr>
<tr>
<td>4.2-1</td>
<td>All Propulsive vs Aeroassist</td>
<td>4-6</td>
</tr>
<tr>
<td>4.2-2</td>
<td>Aeroassist vs All Propulsive OTV Benefit</td>
<td>4-7</td>
</tr>
<tr>
<td>4.2-3</td>
<td>Low vs Mid L/D Performance Trade</td>
<td>4-10</td>
</tr>
<tr>
<td>4.2-4</td>
<td>Aeromaneuver Control Modes</td>
<td>4-11</td>
</tr>
<tr>
<td>4.2-5</td>
<td>Aeromaneuver Overview</td>
<td>4-12</td>
</tr>
<tr>
<td>4.2-6</td>
<td>L/D vs Control Corridor</td>
<td>4-15</td>
</tr>
<tr>
<td>4.2-7</td>
<td>Low L/D Aero Configuration Concepts</td>
<td>4-16</td>
</tr>
<tr>
<td>4.3-1</td>
<td>Engine Payback Comparison</td>
<td>4-18</td>
</tr>
<tr>
<td>4.3-2</td>
<td>Cryo OTV Thrust Trade</td>
<td>4-21</td>
</tr>
<tr>
<td>4.3-3</td>
<td>Multiple Burn Cost Trade</td>
<td>4-21</td>
</tr>
<tr>
<td>4.3-4</td>
<td>Optimum Engine Life</td>
<td>4-22</td>
</tr>
<tr>
<td>4.3-5</td>
<td>Ground-Based Storable Engine Selection</td>
<td>4-24</td>
</tr>
<tr>
<td>4.4-1</td>
<td>Cryo Drop Tank Trade</td>
<td>4-25</td>
</tr>
<tr>
<td>4.4-2</td>
<td>2-Engine Ground-Based Cryo Packaging</td>
<td>4-26</td>
</tr>
<tr>
<td>4.4-3</td>
<td>Ground-Based to Space-Based Cryo Structure Evolution</td>
<td>4-27</td>
</tr>
<tr>
<td>4.4-4</td>
<td>Ground-Based Cryo ASE</td>
<td>4-28</td>
</tr>
<tr>
<td>4.4-5</td>
<td>Cryo Space-Based OTV Delivery</td>
<td>4-29</td>
</tr>
<tr>
<td>4.4-6</td>
<td>Cryo Space-Based OTV Subsystem Servicing Locations</td>
<td>4-30</td>
</tr>
<tr>
<td>4.4-7</td>
<td>Space-Based OTV Avionics Servicing Locations</td>
<td>4-31</td>
</tr>
<tr>
<td>4.5-1</td>
<td>Redundancy Trade</td>
<td>4-32</td>
</tr>
<tr>
<td>4.5-2</td>
<td>Avionics and Power System Redundancy</td>
<td>4-35</td>
</tr>
<tr>
<td>5.1-1</td>
<td>Cryo Configuration Summary</td>
<td>5-1</td>
</tr>
<tr>
<td>5.2-1</td>
<td>Storable Configuration Summary</td>
<td>5-3</td>
</tr>
<tr>
<td>FIGURE #</td>
<td>TITLE</td>
<td>PAGE</td>
</tr>
<tr>
<td>---------------</td>
<td>-------------------------------------------------</td>
<td>------</td>
</tr>
<tr>
<td>6.1.1.1-1</td>
<td>Ground-Based Cryogenic OTV Concept</td>
<td>6-2</td>
</tr>
<tr>
<td>6.1.1.1-2</td>
<td>Ground-Based Cryogenic OTV Layout</td>
<td>6-3</td>
</tr>
<tr>
<td>6.1.1.2.1-1</td>
<td>Ground-Based Cryo Aeroshield</td>
<td>6-4</td>
</tr>
<tr>
<td>6.1.1.2.2-1</td>
<td>Ground-Based Cryogenic Propulsion Schematic</td>
<td>6-10</td>
</tr>
<tr>
<td>6.1.1.2.2-2</td>
<td>Ground-Based N₂H₄ RCS Schematic</td>
<td>6-11</td>
</tr>
<tr>
<td>6.1.1.2.4-1</td>
<td>Block Diagram of the Ground-Based, ACC Delivered, Cryogenic Configuration</td>
<td>6-12</td>
</tr>
<tr>
<td>6.1.1.2.4-2</td>
<td>EPS Configuration for the Ground-Based, ACC Delivered, Cryogenic Configuration</td>
<td>6-16</td>
</tr>
<tr>
<td>6.1.1.4-1</td>
<td>Ground-Based 45 Klb Cryo OTV Performance Capability</td>
<td>6-24</td>
</tr>
<tr>
<td>6.1.2.1-1</td>
<td>Initial Space-Based Cryo OTV</td>
<td>6-25</td>
</tr>
<tr>
<td>6.1.2.1-2</td>
<td>Growth Space-Based Cryo OTV</td>
<td>6-26</td>
</tr>
<tr>
<td>6.1.2.1-3</td>
<td>Cryogenic Lunar Logistics Vehicle</td>
<td>6-27</td>
</tr>
<tr>
<td>6.1.2.1-4</td>
<td>Space-Based Cryogenic OTV</td>
<td>6-28</td>
</tr>
<tr>
<td></td>
<td>55K Propellant - 44 Ft. Diameter Aerobrake</td>
<td></td>
</tr>
<tr>
<td>6.1.2.1-5</td>
<td>Space-Based Cryogenic 81K Propellant - 44 Ft. Diameter Aerobrake</td>
<td>6-19</td>
</tr>
<tr>
<td>6.1.2.2.1-1</td>
<td>Space-Based Cryo Aeroshield</td>
<td>6-30</td>
</tr>
<tr>
<td>6.1.2.2.2-1</td>
<td>Space-Based LH₂/LO₂ Schematic</td>
<td>6-34</td>
</tr>
<tr>
<td>6.1.2.2.2-2</td>
<td>Space-Based GH₂/GO₂ RCS Schematic</td>
<td>6-35</td>
</tr>
<tr>
<td>6.1.2.2.3-1</td>
<td>Space-Based Cryo Transportation</td>
<td>6-36</td>
</tr>
<tr>
<td>6.1.2.2.3-2</td>
<td>Space-Based Cryo Transportation</td>
<td>6-37</td>
</tr>
<tr>
<td>6.1.2.2.3-3</td>
<td>Space-Based Aerobrake</td>
<td>6-38</td>
</tr>
<tr>
<td>6.1.2.2.3-4</td>
<td>Aerobrake Release Mechanism</td>
<td>6-38</td>
</tr>
<tr>
<td>6.1.2.2.3-5</td>
<td>Aerobrake Rib Deflection</td>
<td>6-39</td>
</tr>
<tr>
<td>6.1.2.2.4-1</td>
<td>Block Diagram of the Space-Based Cryogenic Configuration</td>
<td>6-40</td>
</tr>
<tr>
<td>6.1.2.2.4-2</td>
<td>EPS Configuration for the Space-Based Cryogenic Configuration</td>
<td>6-43</td>
</tr>
<tr>
<td>6.1.2.4-1</td>
<td>Space-Based 55Klb OTV Performance Capability</td>
<td>6-55</td>
</tr>
<tr>
<td>6.2.1.1-1</td>
<td>Ground-Based Storable - ACC</td>
<td>6-57</td>
</tr>
<tr>
<td>6.2.1.1-2</td>
<td>Ground-Based Storable - Cargo Bay</td>
<td>6-58</td>
</tr>
<tr>
<td>6.2.1.1-3</td>
<td>37K Storable Ground-Based OTV</td>
<td>6-60</td>
</tr>
<tr>
<td>6.2.1.2.2-1</td>
<td>Ground-Based N₂O₄/MMH Schematic</td>
<td>6-63</td>
</tr>
<tr>
<td>6.2.1.2.3-1</td>
<td>Ground-Based Storable OTV ASE</td>
<td>6-65</td>
</tr>
<tr>
<td>6.2.1.2.4-1</td>
<td>Block Diagram of the Ground-Based Storable Configuration</td>
<td>6-66</td>
</tr>
<tr>
<td>6.2.1.2.4-2</td>
<td>EPS Configuration for Ground-Based Storable Configuration</td>
<td>6-69</td>
</tr>
<tr>
<td>6.2.1.4-1</td>
<td>Ground-Based 37.3Klb Storable Performance Capability</td>
<td>6-78</td>
</tr>
<tr>
<td>6.2.2.1-1</td>
<td>Space-Based GEO Delivery Vehicle</td>
<td>6-79</td>
</tr>
<tr>
<td>6.2.2.1-2</td>
<td>Space-Based Unmanned Servicing Vehicle</td>
<td>6-80</td>
</tr>
<tr>
<td>6.2.2.1-3</td>
<td>Space-Based Manned Servicing Vehicle</td>
<td>6-81</td>
</tr>
<tr>
<td>6.2.2.1-4</td>
<td>53K Space-Based Storable OTV</td>
<td>6-82</td>
</tr>
<tr>
<td>6.2.2.1-5</td>
<td>25K Space-Based Storable OTV</td>
<td>6-83</td>
</tr>
<tr>
<td>6.2.2.1-6</td>
<td>90,000 LB Space-Based Storable OTV</td>
<td>6-84</td>
</tr>
</tbody>
</table>
### LIST OF FIGURES (Continued)

<table>
<thead>
<tr>
<th>FIGURE #</th>
<th>TITLE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.2.2.2.2-1</td>
<td>Space-Based N₂O₄/MMH Schematic</td>
<td>6-88</td>
</tr>
<tr>
<td>6.2.2.2.2-2</td>
<td>Space-Based N₂O₄/MMH RCS Schematic</td>
<td>6-89</td>
</tr>
<tr>
<td>6.2.2.2.4-1</td>
<td>Block diagram of the Space-Based Storable Configuration</td>
<td>6-91</td>
</tr>
<tr>
<td>6.2.2.2.4-2</td>
<td>EPS Configuration for the Space-Based, Storable Configuration</td>
<td>6-94</td>
</tr>
<tr>
<td>6.2.2.4-1</td>
<td>Space-Based 53Klb Storable OTV Performance</td>
<td>6-108</td>
</tr>
<tr>
<td>6.2.2.4-2</td>
<td>SB 90 Klb Storable OTV Performance</td>
<td>6-111</td>
</tr>
<tr>
<td>6.3-1</td>
<td>Multiple Payload Carrier and STS ASE</td>
<td>6-113</td>
</tr>
<tr>
<td>6.3-2</td>
<td>Multiple Payload Carrier and OTV (Side View)</td>
<td>6-113</td>
</tr>
<tr>
<td>6.3-3</td>
<td>Multiple Payload Carrier (End View)</td>
<td>6-114</td>
</tr>
<tr>
<td>6.3-4</td>
<td>Multiple Payload Carrier with 3 Spacecraft</td>
<td>6-114</td>
</tr>
<tr>
<td>6.3-5</td>
<td>Multiple Payload Carrier with 4 Spacecraft</td>
<td>6-115</td>
</tr>
<tr>
<td>6.4.1-1</td>
<td>Cumulative Storable/Cryo Comparison</td>
<td>6-118</td>
</tr>
<tr>
<td>6.4.2-1</td>
<td>Alternative OTV Growth Paths</td>
<td>6-119</td>
</tr>
<tr>
<td>6.4.2-2</td>
<td>Option 1 - Ground-Based to Man-Rated Space-Based Man-Rating</td>
<td>6-120</td>
</tr>
<tr>
<td>6.4.2-3</td>
<td>Option 2 - Ground-Based to Space-Based Followed by Man-Rating</td>
<td>6-121</td>
</tr>
<tr>
<td>6.4.2-4</td>
<td>Option 4 - Expendable to Space-Based Man-Rated</td>
<td>6-121</td>
</tr>
<tr>
<td>6.4.2-5</td>
<td>Option 5 - Expendable to Space-Based Followed by Man-Rating</td>
<td>6-122</td>
</tr>
<tr>
<td>6.4.2-6</td>
<td>Ground-Based Cargo Bay OTV</td>
<td>6-122</td>
</tr>
<tr>
<td>6.4.2-7</td>
<td>Option 7 - All Ground-Based</td>
<td>6-123</td>
</tr>
<tr>
<td>6.4.3.1</td>
<td>OTV Evolutionary Strategy Comparison</td>
<td>6-131</td>
</tr>
<tr>
<td>6.5.1-1</td>
<td>Space Transportation Schedule Summary</td>
<td>6-134</td>
</tr>
<tr>
<td>6.5.1-2</td>
<td>Ground-Based OTV Development</td>
<td>6-135</td>
</tr>
<tr>
<td>6.5.1-3</td>
<td>Space-Based OTV Development</td>
<td>6-136</td>
</tr>
<tr>
<td>6.5.1-4</td>
<td>Dedicated ACC Development</td>
<td>6-137</td>
</tr>
<tr>
<td>6.5.1-5</td>
<td>Propellant Scavenging System Development</td>
<td>6-137</td>
</tr>
<tr>
<td>6.5.2-1</td>
<td>Space-Based: Accommodations Time Phasing</td>
<td>6-138</td>
</tr>
<tr>
<td>6.6.1-1</td>
<td>Selected Development Option</td>
<td>6-139</td>
</tr>
<tr>
<td>TABLE #</td>
<td>TITLE</td>
<td>PAGE</td>
</tr>
<tr>
<td>--------------</td>
<td>------------------------------------------------------------</td>
<td>------</td>
</tr>
<tr>
<td>4.2-1</td>
<td>All Propulsive vs Aeroassist Results</td>
<td>4–8</td>
</tr>
<tr>
<td>4.2-2</td>
<td>Aeroassist Characteristics - Configuration vs Weight</td>
<td>4–9</td>
</tr>
<tr>
<td>4.2-3</td>
<td>Low vs Mid L/D Trade Results</td>
<td>4–10</td>
</tr>
<tr>
<td>4.2-4</td>
<td>Aeroentry Error Analysis</td>
<td>4–14</td>
</tr>
<tr>
<td>4.2-5</td>
<td>Aerobrake Concept Comparison</td>
<td>4–17</td>
</tr>
<tr>
<td>4.3-1</td>
<td>Cryo Main Engine Trade Results</td>
<td>4–19</td>
</tr>
<tr>
<td>4.3-2</td>
<td>Recommended Engine Requirements</td>
<td>4–23</td>
</tr>
<tr>
<td>4.5-1</td>
<td>Propulsion System Man-Rating Trade</td>
<td>4–33</td>
</tr>
<tr>
<td>4.6-1</td>
<td>OTV Aerobrake Definition</td>
<td>4–36</td>
</tr>
<tr>
<td>4.6-2</td>
<td>Propulsion System Conclusions/Recommendations</td>
<td>4–37</td>
</tr>
<tr>
<td>4.6-3</td>
<td>Structures Trade Summary</td>
<td>4–39</td>
</tr>
<tr>
<td>4.6-4</td>
<td>Avionics Trade Summary</td>
<td>4–40</td>
</tr>
<tr>
<td>6.1.1.2.2-1</td>
<td>Ground-Based Cryogenic MPS Summary</td>
<td>6–6</td>
</tr>
<tr>
<td>6.1.1.2.2-2</td>
<td>Ground-Based RCS Summary</td>
<td>6–7</td>
</tr>
<tr>
<td>6.1.1.2.4-1</td>
<td>OTV Avionics Equipment List - GB, ACC, Cryogenic Configuration</td>
<td>6–13</td>
</tr>
<tr>
<td>6.1.1.3-1</td>
<td>Stage Weight Summary - GB Cryo 45K Propellant Load</td>
<td>6–18</td>
</tr>
<tr>
<td>6.1.1.3-2</td>
<td>Detailed Dry Weight Breakdown - GB Cryo 45K Propellant Load</td>
<td>6–19</td>
</tr>
<tr>
<td>6.1.1.4-1</td>
<td>Ground-Based ACC OTV GEO Delivery Delta Vs</td>
<td>6–23</td>
</tr>
<tr>
<td>6.1.1.4-2</td>
<td>Performance Analysis for Required Missions</td>
<td>6–24</td>
</tr>
<tr>
<td>6.1.2.2.2-1</td>
<td>Space-Based Cryogenic MPS Summary</td>
<td>6–32</td>
</tr>
<tr>
<td>6.1.2.2.2-2</td>
<td>Space-Based Cryogenic RCS Summary</td>
<td>6–33</td>
</tr>
<tr>
<td>6.1.2.2.4-1</td>
<td>OTV Avionics Equipment List - SB Configuration</td>
<td>6–41</td>
</tr>
<tr>
<td>6.1.2.3-1</td>
<td>Stage Weight Summary - SB Cryo 55K Propellant Load</td>
<td>6–44</td>
</tr>
<tr>
<td>6.1.2.3-2</td>
<td>Detailed Dry Weight Breakdown - Initial SB Cryo 55K Propellant Capacity</td>
<td>6–45</td>
</tr>
<tr>
<td>6.1.2.3-3</td>
<td>Stage Weight Summary - Space-Based Cryo - 81K Propellant Load</td>
<td>6–49</td>
</tr>
<tr>
<td>6.1.2.3-4</td>
<td>Detailed Dry Weight Breakdown - Growth Space-Based Cryo - 81K Propellant Load</td>
<td>6–50</td>
</tr>
<tr>
<td>6.1.2.4-1</td>
<td>Space-Based OTV GEO Delivery Delta Vs</td>
<td>6–54</td>
</tr>
<tr>
<td>6.1.2.4-2</td>
<td>Space-Based OTV Lunar Delivery Delta Vs</td>
<td>6–54</td>
</tr>
<tr>
<td>6.1.2.4-3</td>
<td>Performance Analysis for Required Missions</td>
<td>6–56</td>
</tr>
<tr>
<td>6.1.2.4-4</td>
<td>Performance Analysis for Growth Lunar Missions</td>
<td>6–57</td>
</tr>
<tr>
<td>6.2.1.2.1-1</td>
<td>Ground-Based Storable Aeroshield</td>
<td>6–59</td>
</tr>
<tr>
<td>6.2.1.2.2-1</td>
<td>Ground-Based Storable MPS Summary</td>
<td>6–62</td>
</tr>
<tr>
<td>6.2.1.2.2-2</td>
<td>Ground-Based Storable RCS Summary</td>
<td>6–63</td>
</tr>
<tr>
<td>6.2.1.2.4-1</td>
<td>OTV Avionics Equipment List - Storable Configuration</td>
<td>6–67</td>
</tr>
<tr>
<td>6.2.1.3-1</td>
<td>System Weight Summary - Ground-Based Storable - 37.3K Propellant Load - Perigee Stage, ACC</td>
<td>6–71</td>
</tr>
<tr>
<td>6.2.1.3-2</td>
<td>Weight Statement Ground-Based Storable - 37.3K Propellant Load Perigee Stage - P/L Bay</td>
<td>6–74</td>
</tr>
</tbody>
</table>
### LIST OF TABLES (Continued)

<table>
<thead>
<tr>
<th>TABLE #</th>
<th>TITLE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.2.1.4-1</td>
<td>Performance Analysis for Required Missions</td>
<td>6-77</td>
</tr>
<tr>
<td></td>
<td>Storable Ground-Based ACC, 37.3K Perigee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.2.1-1</td>
<td>Space-Based Storable Aeroshield</td>
<td>6-86</td>
</tr>
<tr>
<td>6.2.2.2.2-1</td>
<td>Space-Based Storable MPS Summary</td>
<td>6-87</td>
</tr>
<tr>
<td>6.2.2.2.2-2</td>
<td>Space-Based Storable RCS Summary</td>
<td>6-88</td>
</tr>
<tr>
<td>6.2.2.2.4-1</td>
<td>OTV Avionics Equipment List - Storable Configuration</td>
<td>6-92</td>
</tr>
<tr>
<td>6.2.2.3-1</td>
<td>Weight Statement - Space-Based Storable, 53K</td>
<td>6-96</td>
</tr>
<tr>
<td></td>
<td>Propellant Load, Perigee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.3-2</td>
<td>Weight Statement - Space-Based Storable, 90K</td>
<td>6-99</td>
</tr>
<tr>
<td></td>
<td>Propellant Load, Perigee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.3-3</td>
<td>Weight Statement - Space-Based Storable, 25.4K</td>
<td>6-102</td>
</tr>
<tr>
<td></td>
<td>Propellant Load, Apogee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.3-4</td>
<td>Weight Statement - Space-Based Storable, 53K</td>
<td>6-105</td>
</tr>
<tr>
<td></td>
<td>Propellant Load, Apogee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.4-1</td>
<td>Performance Analysis for Required Missions</td>
<td>6-109</td>
</tr>
<tr>
<td></td>
<td>Storable, Space-Based, 53K Perigee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.4-2</td>
<td>Performance Analysis for Required Missions</td>
<td>6-110</td>
</tr>
<tr>
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<td>Storable, Space-Based, 53K Apogee Stage</td>
<td></td>
</tr>
<tr>
<td>6.2.2.4-3</td>
<td>Performance Analysis for Required Missions</td>
<td>6-110</td>
</tr>
<tr>
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<td>Storable, Space-Based, 90K Perigee Stage</td>
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</tr>
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<td>6.2.2.4-4</td>
<td>Performance Analysis for Required Missions</td>
<td>6-111</td>
</tr>
<tr>
<td></td>
<td>Storable, Space-Based 53K Apogee Stage</td>
<td></td>
</tr>
<tr>
<td>6.3-1</td>
<td>OTV - Multiple Payload Carrier (Aluminum)</td>
<td>6-115</td>
</tr>
<tr>
<td>6.4.1-1</td>
<td>Storable vs Cryo OTV Ground Rules and Assumptions</td>
<td>6-117</td>
</tr>
<tr>
<td>6.4.1-2</td>
<td>Cryo/Storable Trade Results (PV)</td>
<td>6-118</td>
</tr>
<tr>
<td>6.4.3-1</td>
<td>ACC vs Cargo Bay Trade Study Ground Rules and Assumptions</td>
<td>6-125</td>
</tr>
<tr>
<td></td>
<td>Ground Rules and Assumptions</td>
<td></td>
</tr>
<tr>
<td>6.4.3-2</td>
<td>OTV Delivery/Scavenging Trade Results</td>
<td>6-126</td>
</tr>
<tr>
<td>6.4.3-3</td>
<td>OTV Evolutionary Program Trade:</td>
<td>6-127</td>
</tr>
<tr>
<td></td>
<td>Ground Rules and Assumptions</td>
<td></td>
</tr>
<tr>
<td>6.4.3-4</td>
<td>Option Cost Summary - Constant Dollars</td>
<td>6-129</td>
</tr>
<tr>
<td>6.4.3-5</td>
<td>Option Cost Summary - Discounted Dollars</td>
<td>6-130</td>
</tr>
<tr>
<td>6.4.3-6</td>
<td>OTV Option Results</td>
<td>6-133</td>
</tr>
<tr>
<td>6.6.2-1</td>
<td>DDT&amp;E In Constant FY85 $</td>
<td>6-140</td>
</tr>
<tr>
<td>6.6.3-1</td>
<td>Initial Production Cost (FY85 $)</td>
<td>6-141</td>
</tr>
</tbody>
</table>
1.0 INTRODUCTION

This portion of the OTV Concept Definition and System Analysis Study, Volume II, Book 2, summarizes the flight vehicle concept selection process and results. It presents an overview of OTV mission and system design requirements and describes the family of OTV recommended, the reasons for this recommendation, and the associated Phase C/D Program.

Figure 1.0-1 depicts the overall process followed in developing the OTV concept definitions during this study. Design driver missions were selected and overall system design requirements were identified in Task 1. The results of this activity are summarized in this overview. The first step in the definition process was to do a parametric assessment of the reasonable propellant and staging options. This activity was supported by analyses conducted under Task 6 system trades. These results were coarse screened in accordance with criteria negotiated between MSFC and contractor personnel. Those concepts judged to have no possibility of being developed into a winner were not studied further.

Figure 1.0-1 Vehicle Concept Definition Approach
The bulk of the vehicle concept definition activity was concentrated in the next step. An iterative process of defining the highest potential OTV concepts was conducted. Ground-based and space-based concepts were developed with a view to providing a reasonable evolution from one to the other. We maintained separate system definition activities for both storable and cryogenic options. The reason for this dual approach was that it is impossible to decide between the concepts at the vehicle definition level. At this level, cryos appear to be a clear winner. Storable advantages appear at the operations and space-basing levels, and a selection between them required awaiting the programmatic assessment of Task 4 and 5 evaluations. The system definition activity for storable and cryogenic concepts was supported by system level trades conducted in Task 6 as well as subsystem level trades in the basic vehicle design areas indicated. The subsystem level trades employed system sensitivities to support their specific selections.

Operations and space-based accommodations assessments of the high potential configurations were conducted in Tasks 4 and 5 and the results fed into the programmatic task to support major program decisions. These program decisions coupled with specific design recommendations from the operations and accommodations assessments were then incorporated into the final OTV concept definitions, and final documentation prepared.

The selection process involved a significant change in mission model at the midterm point of the contract. The first portion of the study concentrated on the selection and optimization of high potential vehicle concepts capable of meeting the requirements imposed by the "nominal" Revision 7 OTV Mission Model, and was completed at midterm. After midterm these concepts were evaluated from the launch and flight operations and space-based accommodations viewpoints, and a preferred program capable of meeting the requirements of the 'low' Revision 8 OTV Mission Model was selected. The change in mission models did not have a significant impact on the high potential configuration concepts. It did impact the selection of which of these configurations to include in the preferred program concept. The high potential concepts did not change in spite of a significant reduction in the driving manned mission payload weight because the concept driven by the lesser 20K delivery mission met the new reduced manned performance capability requirement. Changes in traffic levels and initial operational dates did have a significant impact on program selection. A basic MSFC direction was to make decisions that could be justified on the basis of low model traffic levels. While this direction did not change any fundamental decisions, it did make narrow the margin on some of the choices (for example, cryogenics over storables).

Results are presented in the following sequence. Section 2.0 presents a requirements overview. Sections 3.0 through 5.0 present the complete process of selecting the high potential Orbital Transfer Vehicles. The candidate concepts considered are identified, the major system trade results that discriminate between concepts are summarized, and the resulting high potential concepts in both cryogenic and storable categories are selected. These high potential concepts are described in detail in Sections 6.1 and 6.2. The storable concepts were not recommended for development, but they represent a significant data base and could prove desirable in certain mission scenarios. The final sections in this report (Sections 6.3 to 6.5) present the reasons for recommending the selected evolutionary cryogenic OTV program, and a description of the schedule and cost of this program.
2.0 REQUIREMENTS OVERVIEW

This section presents a summary of the mission and system requirements that were most influential in establishing the preferred OTV design concepts. Driving missions are reviewed, and the most significant system requirements are discussed. A more complete treatment of the mission and system requirements is documented in Volume II, Book 1: Mission and System Requirements.

2.1 DRIVING MISSIONS

The missions from the Revision 8 OTV mission model that drive the design of the flight vehicle are summarized in Figure 2.1-1. Drivers are categorized by operational era (pre and post Space Station) and model (low and nominal). The nature of the low model is particularly important, as it is to be used to justify major configuration decisions. As far as the driver missions are concerned, the only differences between the low and high models are deletion of the driving lunar mission and the less difficult planetary missions. The most important aspect of the low model is its lower traffic level, which tends to make it difficult to justify expenditure of development money. It is important to note that the Rev. 8 model is downgraded from the Rev. 7 model by the incorporation of the Mobile Geosynchronous Service Station (MGSS) concept. This concept reduces the geostationary roundtrip requirement from 14000 pounds to 7500 pounds. The following OTV concept development is keyed to find the best way to perform these driving missions.

<table>
<thead>
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</tr>
</thead>
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<td></td>
<td>UP LB</td>
<td>DN LB</td>
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<tr>
<td>GEO DELIVERY</td>
<td>12000</td>
<td>2000</td>
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<td>5000</td>
<td>@C3 = 50</td>
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<tr>
<td>GEO DELIVERY</td>
<td>20000</td>
<td>0</td>
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<tr>
<td>PLANETARY DELIVERY</td>
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<td>@C3 = 98</td>
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<tr>
<td>UNMANNED GEO SERVICING</td>
<td>7000</td>
<td>4500</td>
</tr>
<tr>
<td>MANNED GEO SERVICING</td>
<td>7500</td>
<td>7500</td>
</tr>
<tr>
<td>MANNED LUNAR SORTIE</td>
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</tbody>
</table>

* REQUIRES TWO Launches

Figure 2.1-1 Design Driver Missions
Prior to the time when Space Station is available, the mission model requires only payload delivery missions. The only retrieval requirement is for the OTV itself and, in the case of multiple GEO delivery missions, the multiple payload airborne support equipment. The merit of retrieving this equipment is, of course, subject to economic evaluation. The driving planetary mission in the pre-space-based portion of the low model is not a payload performance driver, but it does impose unique mission design problems. For example, retrieving OTV from planetary inject mission involves a retro maneuver and a return orbit perhaps as long as four days. During this period of time, the Orbiter's orbit precesses out of the OTV orbit plane and complex plane change maneuvers are required. The nominal model, pre Space Station, planetary driver mission does drive payload capability and will require multiple STS missions and onorbit assembly to implement it. This need is, by MSFC direction, not to drive the selection of OTV systems.

The low model in the post Space Station era introduces a number of new driving missions. The 20,000 pound delivery mission is the pacing payload requirement. The unmanned servicing mission is the first to require retrieval of a sizable payload. The manned GEO servicing mission increases this retrieval payload requirement, and introduces the problem of man-rating the vehicle. The nominal model introduces the very difficult, from a payload performance point of view, manned lunar sortie mission. The 80K lb up and 15K lb back requirement drives both propellant required and the retrieval weight designing the retrieval system (probably an aeroassist device). As in the case of the nominal model planetary mission, this lunar mission requirement is not to drive OTV system selection.

Figure 2.1-2 shows the time phasing associated with the major driving missions for the low and nominal mission models. The dry points are associated with the delay in introducing new capabilities associated with the low model. Availability of the space-base is delayed two years, the requirement for manned operation six years. The manned lunar sortie is extended out of the window under consideration in this study. Since major decisions are to be justified by the low mission model, this situation is critical to this study.

2.2 SYSTEM REQUIREMENTS

System requirements have been derived from these driving missions, and are compiled in their entirety in Volume II, Book 1 of this report. The following paragraphs present highlights of these system requirements and the considerations that led to their selection.

ARCHITECTURAL ROLE—The basic design philosophy followed for the OTV concepts developed in this study is illustrated in Figure 2.2-1. OTV provides the muscle to reach high earth orbit. Finess for advanced operations at high orbit is to be provided by the payloads. In the case of unmanned servicing, a GEO OMV is carried aloft, and it has the six degree of freedom translation capability and servicing systems required to perform the servicing functions. Similarly, manned servicing functions are accomplished using an MGSS. For the interface between the Space Station and the OTV, departure is implemented by provision of a small delta-V to achieve safe separation distance by the hangar system. OTV retrieval from a safe separation distance is to be implemented with OMV maneuvers. In this way, the design of the OTV has been concentrated on providing efficient orbit transfer.
Figure 2.1-2 Time Phasing of Mission Requirements

Figure 2.2-1 OTV's Architectural Role is HEO Truck
REUSE--The OTV defined herein eventually become, by study requirement, reusable. Retrieving the OTV for reuse decreases payload capability on a single ground-based launch, or decreases mission payload delivered per pound of propellant required at a space station. Thus the reusable OTV must be justified on the basis of the value of the retrieved stage, the value of retrieving mission hardware, or the value of manned operation in high earth orbit. Trades to date indicate reusability is justifiable, but that the winning margin can be increased by the use of aeroassisted OTV retrieval. This concept is incorporated in all of the high potential configurations selected.

AEROASSIST--The basic requirements imposed on the aeroassist device are that it survive the aeroheating environment imposed, and that it perform its maneuver accurately in the face of anticipated variations in upper atmosphere density, navigation sensor accuracy, et.al. The designs presented in this report reflect selection of ballistic coefficients that are compatible with projected heatshield materials and a lift to drag ratio that provides adequate maneuver control. Our studies have shown that the ballistic coefficient during reentry from GEO must be less than 10 pounds per square foot and that the L/D must be 0.116 or greater. In addition, the heat shield must be large enough to protect the stage and attached retrieved payload from the aerodynamic wake. All of the configurations presented in this report meet these requirements.

MAN RATING--Based on agreements reached at the First Quarterly Review, all manned OTV must at a minimum provide a fail-safe-return capability. This is interpreted to mean that the OTV must be able to return its crew safely to the vicinity of Space Station (or Orbiter, if ground based) after suffering one credible failure. Unmanned vehicles should use a degree of redundancy that is economically justifiable.

GROUND/SPACE TRANSITION--The ground-based OTV defined herein must, by study requirement, evolve to space-based operation. It is not a requirement that the ground-based vehicles be operable in the space-based mode. The changes between configurations should be a cost effective compromise between: Proof of system and subsystem concept in an initial ground-based mode; maximizing ground-based single launch capability; minimizing space-based propellant requirements; and providing efficient OTV packaging for delivery of the ground-based OTV to LEO as an assembled, loaded unit and the space-based OTV in sub-units readily assembled at space station.

UNIQUE GROUND-BASED REQUIREMENTS--This study considered two techniques for carrying a loaded ground-based OTV to low earth orbit (LEO). Effort was concentrated on ascent in an aft cargo carrier (ACC) attached to the external tank. These ACC OTV were packaged to fit the dedicated ACC as defined in Reference 1. If necessary, an additional 36 inches of length is possible within the ACC compartment. Cargo bay ascent of cryogenic OTV was explored in previous Phase A studies. Comparisons were made with these cryogenic cargo bay configurations after previous results were normalized with current subsystem design data, as described in Volume III. Storable cargo bay OTV were not previously studied, and special emphasis was required in this study. In this case, the requirements of Reference 2 were met in defining OTV general arrangement. In the payload bay application, it is important to minimize OTV length so its payload can be as long as possible. Return to earth in the cargo bay is the only retrieval option available.
ACC or cargo bay utilization has a marked effect on the design requirements the OTV must meet, as summarized in Figure 2.2-2. The net weight available for the flight ready OTV and its payload is shown assuming reasonable estimates for required ASE and orbiter fuel required to support necessary rendezvous maneuvers. Note that the ACC operations scenario selected requires an additional rendezvous between the orbiter and OTV during the ascent phase of the mission. In both cases JSC's projected orbiter capability (Reference 3) and a 140 nautical mile, 28.5 degree inclined ascent injection orbit have been assumed. Net weight available for ACC and cargo bay OTV and their payloads is nearly identical. Loaded OTV adaptation to the ACC structure requires less structural ASE than adaptation to the cargo bay longerons and keel and provision of propellant dump capability. This essentially compensates for the ACC structural weight penalty. The total volume available is greater for the ACC configuration. The ascent envelope available in the ACC favors short, large diameter OTV configurations and blunt mechanically deployed aeroassist configurations. The cargo bay envelope demands long configurations that are under 15 feet in diameter, and favors the use of the inflatable ballute aeroassist approach. The ACC configuration poses two unique operational problems for the ground-based OTV -- onorbit rendezvous and mating of a payload carried aloft in the cargo bay with the OTV, and partial disassembly of the OTV for retrieval in the cargo bay. Special ASE is required for ACC/OTV retrieval, and it must be launched in the cargo bay. 900 pounds of OTV support equipment is included in the 2100 pounds shown in Figure 2.2-2. This figure reflects an OTV designed to ease cargo bay mounting, but it is recognized that this figure could grow to the point where net LEO STS capability could favor the cargo bay vehicle. In both cases, stowage of a used aeroassist device for return to the ground and reuse appears difficult to accomplish. Another difference between the ascent locations is that abort dump provisions are required for the cargo bay, and higher structural margins are required for the safety of the orbiter crew. The ACC location does result in loss of the OTV in the event of an STS abort during ascent that requires return the launch site, while the cargo bay OTV is recovered in the same situation. Growth of the ground-based OTV capability to accomplish the advanced missions in the mission model requires multiple launches to achieve the required lift capability for both ascent locations, although volume constraints, when encountered, are eased with the use of the ACC.

<table>
<thead>
<tr>
<th>DESIGN CONSIDERATIONS</th>
<th>ACC ASSESSMENT</th>
<th>CARGO BAY ASSESSMENT</th>
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<tbody>
<tr>
<td>PERMISSABLE OTV GLOW</td>
<td>67279 lb to 140 NMI</td>
<td>67000 lb to 140 NMI</td>
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<td></td>
<td>REFLECTS 2100 lb ASE</td>
<td>REFLECTS 5000 lb ASE</td>
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<tr>
<td>VOLUME CONSTRAINTS</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>FAVORS SHORT, MULTITANK CONFIG.</td>
<td>CYRO REQUIRES TANDEM TANKS</td>
</tr>
<tr>
<td></td>
<td>MORE P/L VOLUME</td>
<td>P/L VOLUME CONstrained</td>
</tr>
<tr>
<td>RETRIEVAL</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>CYRO REQUIRES LH2 TANK REMOVAL</td>
<td>ASCENT PROVISIONS</td>
</tr>
<tr>
<td></td>
<td>UNIQUE RETRIEVAL ASE</td>
<td>SUPPORT RETRIEVAL</td>
</tr>
<tr>
<td>SAFETY</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>EVACUATE LH2 FOR DISASSEMBLY</td>
<td>PROPPELLANT DUMP REQUIRED</td>
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<tr>
<td></td>
<td>STRUCTURE FACTOR 1.25</td>
<td>STRUCTURE FACTOR 1.4</td>
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<td>AEROSHIELD</td>
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<td></td>
<td>EXPENDABLE DESIGN</td>
<td>FLEX SHIELD FOR STORABLE</td>
</tr>
<tr>
<td></td>
<td>FLEX SHIELD</td>
<td>BALLUTE FOR CYRO</td>
</tr>
<tr>
<td>OPERATIONS</td>
<td></td>
<td></td>
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<tr>
<td></td>
<td>COMPLEX OTV/STS ASCENTOPSNS</td>
<td>OPNS LESS COMPLEX</td>
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<td>ON ORBIT P/L ATTACHMENT</td>
<td>SHORTER MISSION TIMELINE</td>
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<td>DISSASSY FOR RETRIEVAL</td>
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<td>GROWTH MISSIONS</td>
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<tr>
<td></td>
<td>MULTIPLE MISSIONS FOR LIFT CAPACITY</td>
<td>MULTIPLES FOR LIFT</td>
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<td></td>
<td>VOLUME LIMITED</td>
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<td>ASCENT ABORT</td>
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<tr>
<td></td>
<td>OTV LOST</td>
<td>OTV RECOVERED</td>
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</tbody>
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Figure 2.2-2  Ground-Basing Imposes Unique Design Constraints
UNIQUE SPACE-BASED REQUIREMENTS-The general layout of the space based OTV and concepts for its operation at a spacebase were evolved together. The limited availability of crewmen for EVA activity at the Space Station made it a derived requirement that a design that would be space maintainable/serviceable using automation be evolved. While the space-based OTV need not be delivered as an operable unit, it is necessary that deliverable sub-units be efficiently packaged for delivery to Space Station by STS, readily assembled in space, and returnable in the Orbiter cargo bay. It is also a derived requirement that a capability be provided to load and offload OTV propellant in the micro-g environment of the space station. It is a requirement that the space-based OTV propellant tanks be protected against the meteoroid environment defined in Reference 4. In the contract extension activity reported in Volume IX, the added impact of LEO debris was assessed. Cost effectiveness considering system weight and life cycle maintenance as well as achieving single manned mission probability of no penetration in the order of 0.999 provide acceptable shield selection criteria. Figure 2.2-3 summarizes some of the impacts on the design of the OTV that result from the use of the space-based mode. Our data indicates that more efficient OTV design results from space-basing, particularly for the advanced missions that cannot be accomplished in a single shuttle launch. Counter to this, much of the required operations technology is new and is weighed from a programmatic point of view in our trades. Further impacts of a large OTV exist in required space station accommodations—larger hangars and propellant farms. Our cost analyses reflect accommodations capable of supporting OTV's large enough to support the low Revision 8 mission model.

<table>
<thead>
<tr>
<th>DESIGN CONSIDERATIONS</th>
<th>FAVORABLE IMPACTS</th>
<th>UNFAVORABLE IMPACTS</th>
</tr>
</thead>
<tbody>
<tr>
<td>GROSS LAUNCH WEIGHT</td>
<td>NOT CONSTRAINED BY SINGLE LAUNCH CAPABILITY</td>
<td>UNIT PACKAGING MUST REFLECT ASCENT VEHICLE CONSTRAINTS</td>
</tr>
<tr>
<td>SIZE</td>
<td>DESIGN OPTIONS ARE RELATIVELY UNCONSTRAINED</td>
<td>ON-ORBIT ASSY &amp; C/O IS NEW PROBLEM</td>
</tr>
<tr>
<td>EARTH/LEO TRANSPORT</td>
<td>ONLY DELIVERABLE PARTS TO CB/ACC VOLUME LIMITS</td>
<td>METEROID SHIELDING REQ'D NEW REFLT INSPECTION ENV'M'T</td>
</tr>
<tr>
<td>SAFETY</td>
<td>ESCAPES STRESSFUL LAUNCH-LOADED ENVIRONMENT</td>
<td></td>
</tr>
<tr>
<td>AEROSHIELD</td>
<td>ELIMINATION OF REFURL REQUIREMENT ENABLES REUSE</td>
<td></td>
</tr>
<tr>
<td>OPERATIONS</td>
<td>AUTOMATED LAUNCH &amp; MAINT CENTRALIZED FACILITY FACILITATES ON-ORBIT ASSY</td>
<td>REMOTE OPS IS NEW TECH, REQ. BETTER ACCESS, COMPLEX HANDLING FIXTURES, ET. AL.</td>
</tr>
<tr>
<td>GROWTH MISSIONS</td>
<td>NOT LIMITED BY BASIC STS CAPABILITY</td>
<td></td>
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</tbody>
</table>

Figure 2.2-3 Space-Basing Favors More Efficient OTV Design
COMMUNICATION INTERFACES--The OTV will receive command updates and
navigation information and downlink telemetry during its free flight mission.
As a consequence, interfaces between OTV and TDRSS and GPS must be implemented.

PERFORMANCE MARGINS--The OTV's developed in this study reflect the
following reserves and margins in estimating payload performance capability:

a. Flight Performance Reserve equivalent to 2% mission delta-V requirements
b. 15% on estimated dry weights
c. 10% on estimated ACS propellant requirements
d. 20% on estimated fuel cell reactant requirements.
3.0 CONCEPT IDENTIFICATION

The major system candidates that were investigated are summarized in the following paragraphs. An overview of the configuration trades that discriminated between these candidates is presented in Section 4.0, immediately following.

PROPELLANT CANDIDATES—The primary propellant selection issue addressed in this study is the selection between storable propellants and cryogenic propellants. The most appropriate propellants within these categories were also addressed at a subordinate level. Figure 3.0-1 shows the range of propellants that are potentially applicable to the OTV. Of the several high energy propellant combinations, only oxygen/hydrogen is a viable candidate. Only candidates using fluorine are competitive or superior in performance, and the operational problems associated with fluorine are not considered acceptable. Similarly, the only viable room temperature storable propellant combination is nitrogen tetroxide/monomethyl hydrazine. The other alternatives are not sufficiently superior in performance to overcome the N2O4/MMH advantage of being already operationally established in the STS program. Another group of propellants was considered— the space storable options that involve the use of mild cryogenics. Of these options, liquid oxygen in combination with methane, propane or monomethyl hydrazine provides the leading contenders, and the hydrazine option is considered best. Its performance is representative of the class and it has the advantage that both propellants are operationally established in the STS program. These three propellant combinations (L02/LH2, LO2/MMH, and N2O4/MMH) were considered further in this study activity.

Figure 3.0-1 Candidate Propellant Performance
STAGING OPTIONS—The first step of our concept definition activity, coarse screening, discriminated between the more promising staging concepts, and eliminated those that did not merit the investment of significant study resources. Since different staging options are best for different propellant combinations, separate evaluations were conducted for each propellant. The following four staging options comprise the important options investigated:

- SINGLE STAGE
- TWO STAGE
- PERIGEE STAGE (EXPENDABLE APOGEE STAGE)
- 1 1/2 STAGE (EXPENDABLE DROP TANKS)

Single stage represents the simplest and most operationally desirable reusable approach, providing that its performance is competitive. Two stage is the next most complex reusable solution, one that must be considered for lower performance propellants and very demanding missions. The next alternative requiring consideration is incorporation of a degree of expendability. Two approaches that were considered are expending drop tanks and expending an upper "apogee" stage. We are confident that the best staging solution is encompassed by these candidates. For ground-based vehicles the impact of configuring for both the payload bay and the ACC was considered. While reusable concepts were the primary goal of the study, a comparison with expendable designs was conducted. Resolution of the staging issue is described in section 4.0.

REUSABLE CONFIGURATION OPTIONS -- Three basic reuse issues were considered: Whether aeroassist is superior to expendable and all propulsive reusable approaches; whether zero, low L/D (0.25), or medium L/D (0.25 ≤ L/D ≤ 0.75) aeroassist devices are superior; and whether the aeroshield structure should be inflatable, foldable or rigid. Since vehicle configuration, control methodology and aeroassist option are closely interrelated, this evaluation become a complex systems issue, Figure 3.0-2 presents a carpet plot of the payload capability of single STS launched GEO delivery OTV missions flown in expendable, aeroassist retrieved and all-propulsive retrieved modes. The greatest payload capability is achieved in the expendable mode. While manned missions demand a retrieval capability, retrieving the OTV for reuse on delivery missions can only be justified by the value of the reflown OTV being greater than the penalty associated with less efficient use of the shuttle flight. Figure 3.0-2 shows the payload capability of the aeroassist approach is superior to the all propulsive approach if the weight of the aeroassist device can be kept sufficiently low. We have, therefore, emphasized the development of a lightweight aeroassist device in our subsystem design efforts.
Figure 3.0-2 Efficient Aeroassist in a Key Design Objective

Aeroassist options are highly interrelated with the vehicle general arrangements under consideration, which are in turn interrelated with the ascent to orbit technique (e.g. ACC or cargo bay launch, launch assembled for ground-based operations or modular delivery for space-based operations). The general arrangements deal primarily with how propellant tanks will be arranged, where the engines and aeroassist device will be mounted, and where payloads will be attached. The basic stage arrangement concepts investigated in this study were: (1) Axial with tandem tanks; (2) Axial with cluster tanks; and (3) Transverse with cluster tanks. In case (1) the aerodynamic forces and thrust forces are along the same axis, propellant tanks are mounted in tandem along this axis, and the payload is also mounted along this axis. The only deviation in case (2) is that the propellant tanks are multiple and mounted side by side around the thrust/aerodynamic axis. In case (3), the aerodynamic and thrust axes are transverse to each other, and the tanks are clustered about the aerodynamic axis. Variations within these categories are possible, and some were investigated.

Figure 3.0-3 summarizes the most credible aeroassisted configuration options that were considered in this study. The deployable fabric aerobrake shown in the figure functions well with the four tank configuration using axial thrust. It provides excellent aerodynamic stability and good wake protection for a retrieved payload. The raked ellipse suggested by JSC is a more sophisticated aerodynamic shape that can only be implemented by a somewhat
heavier rigid tile system. This configuration avoids the complexity of penetrating the heat shield for the main engine by arranging the thrust axis transversely. Payload is mounted along the aerodynamic axis, inducing significant CG travel complexity. The aeromaneuvering hypersonic sled offers higher L/D with attendant increase in aerosheild weight. Our studies show this approach is not justified by the benefit of aerodynamic turning of GEO retrieval missions. More ambitious mission requirements could justify this approach. The inflatable ballute appears to have considerable merit for configurations launched assembled in the cargo bay. These tandem tank concepts integrate well with the ballute, but tend to require large ballutes to achieve aerodynamic stability, particularly when retrieving long, heavy payloads. We investigated mechanical drag modulation for aeroassist control, as opposed to roll control of low L/D configurations. We found that the weight and complexity of this approach were unacceptable. The aerospike drag modulation concept is illustrated with our ACC configured folding brake. We found the performance advantage of this approach inconclusive and the technical uncertainty beyond our current capability to assess. Further evaluation of these approaches is documented in Volume II, Book 3, Subsystem Trade Studies.

Figure 3.0-3 Configuration Options
MAIN ENGINE CANDIDATES—The main engine issue is a major driver in the development cost of the OTV, and its impact was considered at the system level. The engine options are shown in Figure 3.0-4. The current technology in OTV class engines is the RL10A-3A/B and near term technology is RL10 derivatives. They represent low risk and proven reliability. Advanced engine technology is currently being funded by NASA Lewis with contracts at Aerojet, Pratt & Whitney, and Rocketdyne. Additional work is funded by the Air Force Rocket Propulsion Laboratory (AFPRL) for 500 lb thrust class engines. We visited the three engine contractors to understand the cost and performance issues of derivative and advanced engines. Advanced engine performance and cost could be reduced to obtain a third option: a small 7500 lbf engine that can meet the mission model and evolve into a more advanced engine if future funding and mission constraints dictate.

<table>
<thead>
<tr>
<th>CRYOGENIC</th>
<th>P&amp;W RL-10 DERIVATIVES</th>
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<tbody>
<tr>
<td>o CURRENTLY COMMITTED TO STS &amp; CELV CENTAUR</td>
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<tr>
<td>o PIP UPDATES ON GOING</td>
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<tr>
<td>o CHOICE OF LOW RISK OPTIONS - TO 470+ SEC</td>
<td></td>
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<tr>
<td>o HIGH PROVEN RELIABILITY</td>
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<tr>
<td>- ADVANCED CRYOGENIC ENGINE</td>
<td></td>
</tr>
<tr>
<td>o NASA LERC TECHNOLOGY PROGRAM IN PROGRESS AT ALRC, ROCKETDYNE, P&amp;W</td>
<td></td>
</tr>
<tr>
<td>o USAF - RPL - XLR-134 - 500 LBS</td>
<td></td>
</tr>
<tr>
<td>- IOC CRYOGENIC ENGINE</td>
<td></td>
</tr>
<tr>
<td>o INITIAL GROUND-BASED - 5 HR LIFE ENGINE TO INITIATE OTV PROGRAM</td>
<td></td>
</tr>
<tr>
<td>o CLEAR EVOLUTION TO ADVANCED CRYOGENIC ENGINE &amp; MAN-RATING</td>
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<table>
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<th>STORABLE</th>
<th>XLR-132 PUMP FED</th>
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<tbody>
<tr>
<td>o AFRPL TECHNOLOGY PROGRAMS IN PROGRESS AT ROCKETDYNE AND ALRC</td>
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</tr>
<tr>
<td>o HIGH PERFORMANCE 342+ SEC</td>
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<td>- OMS PUMP FED</td>
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<tr>
<td>o MAN-RATED</td>
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</tr>
<tr>
<td>o REUSABLE</td>
<td></td>
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<tr>
<td>o DERIVATIVE OF FLIGHT PROVEN SYSTEM</td>
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</table>

Figure 3.0-4 Main Propulsion Candidates
The storable engine technology currently active is the XLR-132 program at AFPRC and pump fed OMS, a derivative to the STS OMS. The XR-132 program is studying 3750 lbf engines, however, we have used parametric data supplied by Rocketdyne in selecting an optimum thrust for storable OTV applications, including the DDT&E for a new engine. These engines represent high performance and the advantages of hypergolic, storable propellants.

The Aerojet pump-fed OMS would use low risk pump technology derived from their XLR-132 work and the flight proven, man-rated and reusable STS OMS engine. No technology work is under way for a new L02/MMH engine and this is a negative factor for this propellant combination.

RELIABILITY AND MAN-RATING—The basic issue is how best to achieve an adequate level of mission success and manned safety. Mission success tends to be an economic issue, while safety merits a higher cost solution. The tools available are parts quality, subsystem internal redundancy, and multiple systems, as far as OTV design is concerned. Programmatically, we have considered and discarded the option of a standby rescue capability. Various combinations of the OTV design approaches have been assessed in developing a basic policy with regard to reliability and failure tolerance.

PROGRAM OPTIONS—The key program issues are: whether or not to start the OTV program with a ground-based phase; whether to configure the ground-based vehicle for the cargo bay or the ACC; and when to introduce a fully man-rated OTV system. Candidate programs illustrating these issues have been devised and evaluated. The results are summarized in Section 6.3 and detailed in the programmatic volume.
4.0 CONFIGURATION TRADES & ANALYSES

This section summarizes the major system trades that led to the selection of the high potential Orbital Transfer Vehicle configurations. They led to selection of the preferred staging concepts for cryogenic and storable propellants, the preferred retrieval options, the preferred main engine selection, the preferred general arrangement and the preferred approach to man-rating and redundancy. In addition, the most significant results of the subsystem trades are summarized. All of these subjects are dealt with more exhaustively in other books and volumes of this final report. See Volume II, Book 4 for operations trades, volume III for system and program trades, and Volume IV for Space Station accommodations.

4.1 STAGING/PROPELLANT TRADE

The first step in our OTV definition Process was to do a parametric assessment of the reasonable propellant and staging options. This activity was supported by analyses conducted under the Task 6 system trades. These results were coarse screened in accordance with criteria negotiated between MSFC and contractor personnel. Those concepts judged to have no possibility of being developed into a winner were not studied further.

Figure 4.1-1 lists the criteria that were used in coarse screening staging concepts. In evaluating ground-based concepts, either cryogenic or storable, it was determined first of all if the concept could capture driver missions in the early years of the mission model. Those missions should also be accomplished within the STS cargo limitation on a single STS launch. Finally, the early ground-based OTV must be designed within the technology level expected to be in place in 1987. Criteria for selecting promising space-based OTV staging configurations includes capture of the driver missions in the complete Mission Model through 2010. Those driver missions were discussed in Section 2.0. For space-based OTV, the required propellant becomes the important factor in measuring relative stage performance because of the cost of delivering the propellant from the ground to the Space Station. Staging simplicity was evaluated recognizing the increased maintenance complexity at the Space Station.

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<tr>
<td>✓ WITHIN 1987 TECHNOLOGY</td>
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<th>BASIS</th>
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<tr>
<td>✓ PROPELLANT REQUIRED</td>
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</tr>
<tr>
<td>✓ STAGING SIMPLICITY</td>
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<td></td>
</tr>
</tbody>
</table>

Figure 4.1-1 Initial Concept Screening Criteria
Separate screening activities were conducted for cryogenic, storable and combination propellants. Perigee stage, 1-stage, 1 1/2 stage and 2 stage options were evaluated.

Figure 4.1-2 presents the results of the ground-based LO2/LH2 coarse screening activity. The most important ground-based screening criteria is gross vehicle weight. The gross weight of the stage and payload are shown for several payload weights and staging arrangements. In the case of perigee kick staging, the weight of the required apogee kick stage is included. These values reflect the 460 second Isp from the RL10CAT-IIB engine, appropriate performance margin, and the following stage burnout weight algorithm:

\[ W_{BO} = 1.3033[2701 + 0.0054688(\text{Prop}) + 0.7838497(\text{Prop})^{2/3}] + 0.01(\text{Prop}) \]

<table>
<thead>
<tr>
<th>PROPELLANT</th>
<th>MISSION</th>
<th>PGE KICK</th>
<th>1 STAGE</th>
<th>1-1/2 STAGE</th>
<th>2 STAGE</th>
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<td>LO2/LH2</td>
<td>GEO 8K</td>
<td>37345</td>
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**CONCLUSIONS**
- 2-STAGE SHOULD BE ELIMINATED: POORER PERFORMER THAN ONE STAGE
- 1-1/2 STAGE SHOULD BE ELIMINATED: INFERIOR TO PERIGEE KICK
- PERIGEE KICK IS A VALID OPERATIONAL MODE
- 1-STAGE IS PREFERRED APPROACH (MAY EXPEND CARRIER)

Figure 4.1-2 Ground-Based Cryogenic Screening Results
This algorithm reflects the ground-based design we developed during our 1983 IR&D studies -- a single engine, 4-tank design using a Nextel covered aerobrake that folds forward for stowage in the aft cargo carrier. The crosshatched candidates are clearly unacceptable because they exceed the capability of a single shuttle launch. The two stage designs should be eliminated from further consideration because their gross weight is greater than the simpler staging approaches. All other options are capable of meeting the initial driving mission requirement of delivering a 12.9K lb payload to GEO. Some may not be able to retrieve multimission support equipment. This possibility required further evaluation. 1 1/2 stage, using drop tanks, provides a definite gross weight advantage over single stage, but not as much as the perigee stage approach. Both of these approaches expend mission hardware -- and the perigee stage approach is preferred because of its gross weight advantage. It is clear the ground-based cryogenic OTV should be a single stage. We believe it should be sized to enable maximum performance in the single stage mode, and be used offloaded in the perigee stage mode whenever there is an operational advantage (e.g., S/C is designed with an apogee kick, the mission can be manifested with another spacecraft, etc).

The space-based OTV coarse screening differs from the ground-based case in that the missions anticipated are more ambitious and that the selection criteria changes. Note that the screening shown in Figure 4.1-3 was done for the Rev. 7 mission model, which was more demanding than Rev. 8, but the changes were not in a direction tending to invalidate the conclusions reached here. In the space-based case, the technology availability date does not drive and gross weight no longer needs to be within the STS cargo limitation. Transport to LEO is still the most important part of the LCC equation, but the quantification of single launch capability loses its importance in the Space Station era when propellants will be stored onorbit for when they are needed. The array of selection data was generated using the same weight and specific impulse assumptions as for the ground-based case. The later IOC admits the possibility of an advanced engine development, but this possibility does not affect the relative merit of the various staging arrangements. Our conclusions are as summarized in Figure 4.1-3. Two-stage configurations have no discernable advantage for GEO missions -- but are the preferred approach for the driving lunar mission. While the single stage approach is the simplest for GEO missions, 1-1/2 stage must be considered more carefully because it has a definite propellant advantage. We believe the perigee kick approach should be considered as a valid operational mode. In cases where an expendable apogee kick stage is a viable mission approach, propellant logistics cost can be significantly reduced.

Based on these screening results, we optimized a single cryogenic stage approach to GEO and planetary missions that evolved into a two stage design for the driving manned lunar sortie mission. We investigated the 1 1/2 stage alternative to the single stage, and it proved not to be a winner. We used the perigee kick mode for selected planetary missions. We investigated the use of mixed stages (cryo perigee stage with storable apogee stage) and found no significant benefit.

Figure 4.1-4 is an equivalent evaluation of the ground-based storable staging options.
PROPELLANT WEIGHTS (LB)

<table>
<thead>
<tr>
<th>PROPELLANT</th>
<th>MISSION</th>
<th>PROPELLANT QUANTITY</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>PGE KICK*</td>
</tr>
<tr>
<td>LO₂ / LH₂</td>
<td>GEO DELIVERY 20K</td>
<td>53499</td>
</tr>
<tr>
<td></td>
<td>GEO SERV 7K/4.51K</td>
<td>45665</td>
</tr>
<tr>
<td></td>
<td>GEO MANNED 14K/14K</td>
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<tr>
<td>Iₚₛ = 460</td>
<td>LUN MAN. 80K/15K</td>
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<tr>
<td></td>
<td>PLANETARY (17065)</td>
<td>49123</td>
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</table>

* INCLUDES AKM

CONCLUSIONS
- 2-STAGE CONFIGURATIONS SHOULD BE ELIMINATED FOR GEO MISSIONS: POORER PERFORMANCE THAN 1-STAGE
- 2-STAGE CONFIGURATIONS ARE PREFERABLE FOR THE MANNED LUNAR SORTIE: LEAST PROPELLANT REQUIREMENT
- 1-1/2 STAGE MUST BE CONSIDERED FOR GEO MISSIONS SIGNIFICANT PROPELLANT SAVING OVER 1-STAGE
- PERIGEE KICK IS A VALID OPERATIONAL MODE FOR GEO DELIVERY
- 1-STAGE IS PREFERRED APPROACH PRIOR TO LUNAR PROGRAM

Figure 4.1-3 Space-Based Cryogenic Screening Results

GROSS WEIGHTS (LB)

<table>
<thead>
<tr>
<th>PROPELLANT</th>
<th>MISSION</th>
<th>PGE KICK*</th>
<th>1 STAGE</th>
<th>1-1/2 STAGE</th>
<th>2 STAGE</th>
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</thead>
<tbody>
<tr>
<td>N₂O₄/MMH</td>
<td>GEO 8K</td>
<td>43129</td>
<td>74354</td>
<td>70068</td>
<td>74033</td>
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<tr>
<td></td>
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<td>51644</td>
<td>83364</td>
<td>78537</td>
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<tr>
<td></td>
<td>12K</td>
<td>60184</td>
<td>92412</td>
<td>87038</td>
<td>90662</td>
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<tr>
<td></td>
<td>14K</td>
<td>68754</td>
<td>101503</td>
<td>95557</td>
<td>99029</td>
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<tr>
<td></td>
<td>16K</td>
<td>77358</td>
<td>110642</td>
<td>104115</td>
<td>107435</td>
</tr>
</tbody>
</table>

WILL NOT PERFORM REQUIRED MISSIONS
EXCEEDS LIFT CAPACITY OF SINGLE STS FLIGHT

MAXIMUM MISSION IN 1993/1994 -- 12.9K-LB DELIVERY TO GEO

Figure 4.1-4 Ground-Based Storable Screening Results
The parametric data presented in this chart shows the gross weight of the OTV, propellant, and payload for GEO delivery missions from 8 klb to 16 klb. Data is included for the OTV operating as a reusable perigee kick stage, as a reusable single stage, as a reusable stage with expendable drop tanks (1-1/2 stage), and as a completely reusable two stage vehicle. The gross weights in the perigee (PGE) kick stage column include the weight of the expendable apogee kick motor which is required when the OTV operates in this mode. The crosshatching indicates those gross weights that exceed the lift capacity of a single STS flight and a single 1-1/2 stage point that, while marginal for a single STS flight can be discarded because the 8 klb GEO payload will not meet the mission requirements in the ground-based time frame. The conclusion from this study is that for ground-based delivery of payloads to GEO, the perigee mode of operation is the only viable storable OTV staging arrangement.

The conclusion from this study is that for ground-based delivery of payloads to GEO, the perigee mode of operation is the only viable storable OTV staging arrangement.

The parametric data in Figure 4.1-5 indicate the propellant weights required for space-based storable OTVs to perform the driver missions when operating as a perigee stage, single stage, 1-1/2 stage, and two stage vehicle. The gross weight for the perigee kick stage mode includes the weight of the expendable apogee kick motor required to complete the mission. The crosshatching indicates the conclusions that can be drawn from these data. First, it is obvious that the perigee kick stage mode is not suitable for roundtrip missions. The remaining crosshatching indicates those operating modes that require excessive propellant and therefore are not efficient ways to operate. It can be concluded from these data that the perigee kick stage is a very attractive mode of operation for delivery missions from the Space Station. For roundtrip servicing missions to GEO either 1-1/2 stage or two stage operation appears to be the most efficient staging arrangement. The propellant quantities required to accomplish the manned lunar missions with an all storable vehicle are so large as to make the feasibility of the missions questionable.

<table>
<thead>
<tr>
<th>PROPELLANT</th>
<th>MISSION</th>
<th>PGE KICK*</th>
<th>1 STAGE</th>
<th>1-1/2 STAGE</th>
<th>2 STAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>N₂O₄/11H</td>
<td>GEO DELIVERY 20K</td>
<td>68819</td>
<td>97506</td>
<td>91898</td>
<td>90663</td>
</tr>
<tr>
<td></td>
<td>GEO SERV 7K/451K</td>
<td>81779</td>
<td></td>
<td>74529</td>
<td>77542</td>
</tr>
<tr>
<td>ISp = 342.3</td>
<td>GEO MANNED 14K/14K</td>
<td>154988</td>
<td></td>
<td>144175</td>
<td>144291</td>
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<td>LUN MAN. 80K/15K</td>
<td>404101</td>
<td></td>
<td>388075</td>
<td>299566</td>
</tr>
</tbody>
</table>

* INCLUDES GROSS WEIGHT OF AKM NEEDED TO COMPLETE MISSION (18462 LB)

** PROPELLANT WEIGHTS (LB)**

- EXCESSIVE PROPELLANT REQUIRED
- NOT SUITABLE FOR ROUND TRIP MISSIONS

Figure 4.1-5 Space-Based Storable Screening Results
The net conclusion of the storable screening is as follows. All storable delivery missions should use only the perigee kick mode of operation. Both 1 1/2 and 2 stage configurations should be considered for manned and unmanned GEO servicing missions. An all storable concept for implementing the lunar sortie mission appears to require excessive amounts of propellant. As a consequence, a cryo perigee stage in conjunction with a storable upper stage should be considered for this mission in preference to an all storable approach.

The combination propellant LO₂/MMH was investigated in a similar manner. The evaluation resulted in recommendations identical to those reached for the storable options, although the propellant quantities required were somewhat less.

4.2 RETRIEVAL TRADE

Our retrieval trades combined technical and programmatic data to develop a validated position on the economic viability of OTV retrieval and reuse in conjunction with the low Revision 8 Mission Model. We also established the most attractive means of implementing OTV retrieval. Very early in the process, we established the performance advantage of the aeroassisted approach to retrieval over the all propulsive approach. Figure 4.2-1 shows the conditions under which aeroassist provides a net mission propellant savings for two key missions selected from the Rev. 7 mission model. Our technical data shows that an aerobrake weight/recovered weight fraction of 0.19 is achievable for the delivery mission, 0.07 for the 14K lb round trip mission.

![Figure 4.2-1 All-Propulsive vs Aeroassist](image-url)
This indicates a sizeable propellant savings and, since propellant is a major portion of program cost, should indicate an economic advantage. This says nothing about the merit of retrieval and reuse over expendable stages. During our programmatic analyses we addressed the complete economic issue. Figure 4.2-2 presents the results of this evaluation. This chart shows the time development of benefits after payback for the 145 missions in the Rev. 8 low mission model. The benefit data compares the aeroassisted and all propulsive approaches with the expendable approach. The data reflects discounted dollars. The zero dollar line represents the expendable approach. The expendable reference was constructed assuming a mixed fleet of PAM's, IUS's and Centaurs. A stretched Centaur was conceived to implement the more difficult missions in the model. Retrievals were performed by the Centaur or stretched Centaur, but the Centaur was not reused. The figure shows a net benefit for reuse beginning in 1996, with the aeroassist approach yielding the larger benefit. The low model justifies the use of aeroassist — the nominal model would increase its benefit margin. More details relative to this evaluation are presented in Volume III. Table 4.2-1 shows several evaluation parameters comparing aeroassist with all propulsive retrieval that were developed and reported in Volume III. Aeroassist is the superior approach in all respects except development cost. The return on investment column in the figure shows that this investment is economically justifiable.

![Figure 4.2-2 Aeroassist vs All Propulsive OTV Benefit](image-url)
<table>
<thead>
<tr>
<th>REENTRY OPTIONS</th>
<th>ROI (PV)</th>
<th>BENEFITS *PROD (PV)</th>
<th>DDT&amp;E (CV)</th>
<th>LCC (PV)</th>
<th>COST/FLIGHT* (PV)</th>
<th>TOTAL OPS COST (PV)</th>
<th>PAYBACK NO/MISS. (10/YR)</th>
<th>DDT&amp;E/PROD./OPS TOTAL (PV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>ALL PROPELLUSIVE</td>
<td>2.5</td>
<td>2684</td>
<td>776</td>
<td>21390</td>
<td>86M*</td>
<td>4035</td>
<td>69</td>
<td>4810</td>
</tr>
<tr>
<td>AEROASSIST</td>
<td>3.1</td>
<td>3384</td>
<td>820</td>
<td>17953</td>
<td>77M*</td>
<td>3330</td>
<td>61</td>
<td>4150</td>
</tr>
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<table>
<thead>
<tr>
<th>RELATIVE SCORE (10 IS BEST)</th>
<th></th>
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<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>ALL PROPELLUSIVE</td>
<td>8.1</td>
<td>7.9</td>
<td>10.0</td>
<td>8.4</td>
<td>9.0</td>
<td>8.25</td>
<td>8.8</td>
<td>8.6</td>
</tr>
<tr>
<td>AEROASSIST</td>
<td>10.0</td>
<td>10.0</td>
<td>9.5</td>
<td>10.0</td>
<td>10.0</td>
<td>10.0</td>
<td>10.0</td>
<td>10.0</td>
</tr>
</tbody>
</table>

* INCLUDES P/L DELIVERY TO LEO
The basic aeroassist concepts investigated were shown in Figure 3.0-3. These included: the deployable, conical, fabric lifting brake; the blunt raked ellipse lifting brake; the aeromaneuvering hypersonic biconic sled; the inflatable ballute with inflation level modulating drag; the pyramidal brake with mechanical drag modulation; and the conical aerobrake with fluid aerospike. The key descriptive parameters of these concepts are summarized in Table 4.2-2. All of them possess a ratio of aeroassist device to vehicle return weight that will yield an advantage over all propulsive retrieval according to the criteria shown in Figure 4.2-1. The first trade we undertook was to decide between the low and mid L/D concepts. Figure 4.2-3 shows specific configurations developed to compare the impact of L/D on both storable and cryogenic propelled vehicles and Table 4.2-3 shows the resulting trade Parameters. In the storable case, the rigid/flexible aerobrake is a clear winner over the hypersonic biconic sled. The 1.0 L/D of the sled configuration is used to aerodynamically implement a portion of the 28.5 degree turn required to return from GEO to the east launched LEO. A 1410 pound fuel savings results. The dry weight of the aerobraked configuration, including both aeroassist and propulsive differences, was 6 KLB less than the sled configuration. The net initial weight advantage goes to the aerobrake by 4662 pounds, in spite of the lower velocity budget associated with the sled concept. A similar trade was performed for cryogenic configurations incorporating a slant nosed cylinder and low L/D lifting brake concepts. The low L/D concepts were winners for GEO missions. We concentrated further efforts on low L/D concepts.

Table 4.2-2 Aeroassist Characteristics - Config. vs Weight

<table>
<thead>
<tr>
<th>CONFIGURATION</th>
<th>L/D</th>
<th>W/CDA</th>
<th>WA</th>
<th>λ</th>
</tr>
</thead>
<tbody>
<tr>
<td>DEPLOYABLE CONICAL FABRIC LIFTING BRAKE</td>
<td>0.12</td>
<td>10</td>
<td>1500</td>
<td>.07</td>
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<tr>
<td>BLUNT RAKED ELLIPSE LIFTING BRAKE</td>
<td>0.27</td>
<td>15</td>
<td>1800</td>
<td>.08</td>
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<tr>
<td>AEROMANEUVERING HYPERSONIC BICONIC SLED</td>
<td>1.0</td>
<td>70</td>
<td>6800</td>
<td>.27</td>
</tr>
<tr>
<td>INFLATABLE BALLUTE</td>
<td>0.0</td>
<td>6</td>
<td>3700</td>
<td>.15</td>
</tr>
<tr>
<td>MECHANICAL DRAG MODULATION</td>
<td>0.0</td>
<td>8</td>
<td>5640</td>
<td>.22</td>
</tr>
<tr>
<td>70% AEROBRAKE WITH FLUID AERO-SPIKE</td>
<td>0.0</td>
<td>4**</td>
<td>1520</td>
<td>.22**</td>
</tr>
</tbody>
</table>

NOTE: WA = WEIGHT OF AEROASSIST DEVICE
λ = RATIO OF AEROASSIST DEVICE TO VEHICLE RETURN WEIGHT (14K) PAYLOAD
** = DATA APPLIES TO DELIVERY MISSION
Figure 4.2-3 Low vs Mid L/D Performance Trade

Table 4.2-3 Low vs Mid L/D Trade Results

<table>
<thead>
<tr>
<th></th>
<th>STORABLE TRADE</th>
<th>CRYOGENIC TRADE</th>
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<tbody>
<tr>
<td></td>
<td>HYPERSONIC</td>
<td>RIGID/</td>
</tr>
<tr>
<td></td>
<td>BICONIC SLED</td>
<td>FLEXIBLE</td>
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<tr>
<td></td>
<td>(LaRC) L/D = 1.0</td>
<td>AEROBRAKE</td>
</tr>
<tr>
<td></td>
<td>L/D</td>
<td>SLANT-NOSED</td>
</tr>
<tr>
<td></td>
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<td>CRYOGENIC</td>
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<td></td>
<td>CYLINDER</td>
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<td></td>
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<td>ELLIPTICAL</td>
</tr>
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<td></td>
<td></td>
<td>FLEXIBLE</td>
</tr>
<tr>
<td></td>
<td></td>
<td>AEROBRAKE</td>
</tr>
<tr>
<td>L/ΔD</td>
<td>1.00</td>
<td>0.12</td>
</tr>
<tr>
<td>W/CDA</td>
<td>70.0</td>
<td>10.8</td>
</tr>
<tr>
<td>V_{TPS}</td>
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<td>1393</td>
</tr>
<tr>
<td>W_{DRY}</td>
<td>12,585</td>
<td>6553</td>
</tr>
<tr>
<td>FUEL SAVINGS</td>
<td>1410</td>
<td>+4662</td>
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<tr>
<td>BENEFIT</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

- Propellant savings from increasing L/D does not offset vehicle weight increase in TPS.
- The net performance benefit is with low L/D and no inclination steering.
The next configuration trade performed was to decide between mechanical drag control, aerospike drag modulation and lift control for low L/D concepts. Our trajectory simulation was used to compare these three basic approaches to aerobraking: Control corridor parametrics were generated for varying levels of aerospike thrust, drag modulation ratio, and L/D. All trajectories are for a ground-based OTV configuration returning from a geosynchronous mission orbit. All the parametrics were normalized to show impact of the various approaches on the aerodynamic control corridor. For the case of aerospike control, it may be seen from Figure 4.2-4 that the control authority is limited to an approximately 6 mile wide corridor (with correspondingly high propellant usage). The geometric constraints of mechanical drag modulation appear to limit its area variation to less than 3:1. From the chart one can see that this corresponds to a control corridor of 3 nm or less. This represents a somewhat marginal control situation, based on our aeroentry error analysis work. The offset C.G. approach (lift control) appears to offer the largest amount of control for the smallest vehicle impact. For example, L/D values of .25 are easily achievable with the 70 degree Viking aeroshell and result in control corridor widths on the order of 12 nm. This is more than adequate to cover trajectory dispersions. Our conclusion is that lift control is the most promising method of controlling the OTV through the aeropass.

Figure 4.2-4 Aeromaneuver Control Modes
Our next trade study compared alternative means of implementing the low L/D concept with lift control. The first step in this process is establishing precisely how much L/D is required for adequate control. The most efficient configuration will be designed to operate at the lowest L/D capable of meeting control requirements.

Figure 4.2-5 presents an overview of the aeroentry process using low L/D and lift control. The control corridor forms a tunnel within the atmosphere which defines where the vehicle can successfully fly. Note that the bottom of the control corridor is defined by an operational boundary rather than a dynamic one. This is because flying at the bottom of the dynamic corridor causes very depressed perigees in the postaero orbit which requires a large amount of fuel to correct. Just prior to entry the OTV performs a final midcourse correction (entry minus 1 hour), stellar and GPS updates, and a preentry guidance update. After accomplishing these tasks, the OTV establishes an entry attitude which it holds until entry begins at a sensed acceleration of .03 g's.

Figure 4.2-5  Aeroentry Overview
As the entry proceeds, guidance updates (every 10 seconds) refine the desired pointing of the vehicle lift vector. Upon achieving velocity targets, the vehicle initiates a continuous roll to null the fixed lift vector. In a typical trajectory, subsequent roll holds are required to tweak the trajectory. This process continues until the vehicle exits the atmosphere, at which time the apogee and inclination targets for the post-aero orbit have been achieved.

A series of error sources were considered and their impacts normalized to an equivalent variation in vacuum perigee. The RSS total of these effects was then used to size the aerocontrol corridor and the L/D of the vehicle. The sources were grouped into two categories: 1) targeting errors which cause OTV to miss its desired atmospheric aiming point and 2) aerodynamic variations which cause the vehicle to fly a different atmospheric trajectory than expected.

1) Targeting errors - The last opportunity to correct the OTV's downleg trajectory occurs one hour before entry with a midcourse correction burn. All errors prior to this point are nulled out and only those factors that disturb the burn and subsequent trajectory are considered.

   a) Guidance Errors - Experience indicates an error of about 200 ft for this parameter.
   b) Pointing Errors - Midcourse burn attitude errors due to IMU misalignment (after stellar update) and cg trim errors amount to about 0.1 deg. which equates to 130 ft variation in vacuum perigee.
   c) Cutoff Errors - Accelerometer errors and a 10 millisecond shutdown uncertainty.
   d) GPS Error - Estimates of state vector errors for GPS at this stage and 2 fps in velocity. This leads to perigee errors of 845 ft and 9476 ft respectively.
   e) Onboard Clock Error - Very accurate time comes with the use of GPS - not a significant effect.
   f) Nongravitational Effects - Nonbalanced configuration of the RCS jets does not produce pure torques. This is estimated to result in a 320 ft perigee miss. Luni-solar effects will be biased by ground targeting.

2) Aerodynamic Variations - No two aeroentries will be quite the same. The impact of variations in the atmosphere and the vehicle are accounted for here.

   a) Atmospheric Uncertainty - The variation in density has been ground ruled by MSFC at 30%
   b) L/D Uncertainty - An angle-of-attack variation of 1° due to variations in the entry cg
   c) Ballistic Uncertainty - Weight uncertainty - 150 lbs (propellant residual uncertainty), coefficient of drag (C_d) variation = 10% (Shuttle and Viking experience), and brake area variation = 5% (to cover uncertainties in the flex of the support ribs and flexible TPS blanket). The RSS effect of these factors on ballistic coefficient is 12%.
RSS'ing of all the above factors (See Table 4.2-4) yields a net variation in perigee of +1.27 nm. A control corridor of 3.5 will cover this uncertainty with adequate margin. Further closed loop flight simulations were run with the actual local variations in the upper atmosphere encountered on Shuttle flights. The net impact of these variations is to require an increase in control corridor over that indicated by the RSS analysis just described. The detailed trajectory results are presented in Volume II, Book 3. As a result of these analyses, we increased the control corridor requirement to ± 2.5nmi, or a total width of 5 nmi.

Table 4.2-4  Aeroentry Error Analysis

<table>
<thead>
<tr>
<th>EQUIVALENT PERIGEE ERROR</th>
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<tr>
<td>TARGETING ERRORS (MIDCOURSE)</td>
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<tr>
<td>GUIDANCE ERRORS</td>
</tr>
<tr>
<td>POINTING ERROR</td>
</tr>
<tr>
<td>CUTOFF ERROR</td>
</tr>
<tr>
<td>GPS ERROR</td>
</tr>
<tr>
<td></td>
</tr>
<tr>
<td>NONGRAVITATIONAL</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>AERODYNAMIC VARIATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>ATMOSPHERIC UNCERTAINTY</td>
</tr>
<tr>
<td>L/D UNCERTAINTY</td>
</tr>
<tr>
<td>BALLISTIC UNCERTAINTY</td>
</tr>
<tr>
<td></td>
</tr>
<tr>
<td></td>
</tr>
</tbody>
</table>

| RSS |
|     |
| ± 977 FT | ± 0.16 N.M. FROM TARGETING |
| ± 7680 FT | ± 1.26 N.M. FROM AERODYNAMICS |

- ± 7742 FT | ± 1.27 N.M. NET VARIATION
Using the 5 nm control corridor width that results from the aeroentry error analysis it is possible to specify the L/D requirements for the OTV. A series of continuous lift-up and lift-down geosynchronous return trajectories were generated for various L/Ds to define corridor boundaries. The resulting control corridor widths are plotted in Figure 4.2-6. This data shows that an L/D of 0.116 gives the desired 5 nm corridor. This L/D is achieved via an angle-of-attack of 7.2 degrees based on Viking data for this type of aerobrake shape. An analysis of free molecular flow effects shows no significant impact on this angle of attack.

Figure 4.2-6 L/D vs Control Corridor
The primary low L/D concepts are the inflatable ballute, the rigid raked elliptical cone, and the Viking shaped rigid/flexible fabric aerobrake -- as shown in Figure 4.2-7. The ballute and fabric brakes both use flexible thermal protection systems surrounding a rigid spherical segment nose cap with protective doors covering the main engines. The raked ellipse uses rigid thermal protection over the entire exposed area. The propulsive axis is located transversely, eliminating the need for engine doors in the heat shield. Table 4.2-5 provides comparisons of six areas for the three low L/D aerobrake candidates. Design factors for both drag and lift devices; aerobrake/stage characteristics; operational impacts on launch to orbit and Space Station reuse and replacement; payload sizes -- brake dimensions, weights and efficiency ratios; OTV design impacts; and concerns and risks for TPS, control, feasibility, and weight growth are shown. As a final comparison of the ballute concept versus the fixed passive structure, wind tunnel data of these two approaches were compared. References 4 and 5 which were prepared during the Viking development activity provided an additional comparison of an "attached inflatable decelerator" (essentially a ballute) with the rigid shape eventually selected for Viking. The Viking shape was a superior decelerator with a higher drag coefficient, and had a better potential for producing L/D for control purposes. An additional significant difference is static aerodynamic stability. The Viking center of pressure lies 1.01 brake diameters aft of the brake nose, while the ballute is only 0.3 diameters aft. This makes it possible to stabilize a longer stage/payload configuration using a small brake diameter with the Viking shape. Considering the comparative data presented, it is our position that the Viking shaped rigid/flexible aerobrake is the superior low L/D aeroassist concept.

![Figure 4.2-7 Low L/D Aero Configuration Concepts](image-url)
Table 4.2-5 Aerobrake Concept Comparison

<table>
<thead>
<tr>
<th>FACTOR</th>
<th>INFLATABLE BALLUTE</th>
<th>RAKED ELLIPTICAL CONE</th>
<th>RIGID FLEXIBLE AEROBRAKE</th>
</tr>
</thead>
<tbody>
<tr>
<td>DESIGN SUMMARY</td>
<td>BAC STUDIES ZERD 4.6 / 12.3 AREA VARIATION</td>
<td>JSC STUDIES 0.3 OR LOWER 8.1 / 15.1 ROLL CONTROL</td>
<td>MMC STUDIES 0.12 4.0 / 11.6 ROLL CONTROL OFFSET CG</td>
</tr>
<tr>
<td>CHARACTERISTICS</td>
<td>BLUNT CONICAL SPHERICAL NOSE 50 FT</td>
<td>RAKED CONICAL ELLIPSOIDAL NOSE 40 FT 38D X 14L RIGID FLEX CP VARES 1-RADIUS AFT OF AIR BASE (34 FT)</td>
<td>BLUNT CONICAL SPHERICAL NOSE 44 FT 38D X 25L RIGID FLEX WIDE CG LATITUDE (43 FT)</td>
</tr>
<tr>
<td>OPERATIONS</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>SHIP FOLDED FABRIC AS UNIT</td>
<td>SECTIONS ASSY REQUIRED</td>
<td>SHIP ASSY W/FABRIC FOLDED</td>
</tr>
<tr>
<td></td>
<td>NOT PRACTICAL RECHARGE PRESSURANT SIMPLE INSTALL UNIT</td>
<td>YES VISUAL CHECK</td>
<td>YES VISUAL CHECK</td>
</tr>
<tr>
<td></td>
<td></td>
<td>COMPLEX REPLACE TILES OR ASSY</td>
<td>SIMPLE INSTALL AS A SINGLE ASSY</td>
</tr>
<tr>
<td>SIZE</td>
<td>LONG STABILITY</td>
<td>WAKE HEATING</td>
<td>WAKE HEATING</td>
</tr>
<tr>
<td>20K P. DELIVERY</td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>AEROBRAKE DIA FT</td>
<td>40</td>
<td>37</td>
<td>38</td>
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<tr>
<td>AEROBRAKE MASS</td>
<td>1569</td>
<td>1587</td>
<td>1270</td>
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<td>STAGE DRY WT, LB</td>
<td>8970</td>
<td>9489</td>
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<tr>
<td>W</td>
<td>194</td>
<td>0.167</td>
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<td>BRAKE RETURN</td>
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<tr>
<td>75K MAN GEO SORTIE</td>
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<tr>
<td>AEROBRAKE DIA FT</td>
<td>50</td>
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<td>44</td>
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<tr>
<td>AEROBRAKE MASS</td>
<td>2452</td>
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<td>W</td>
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<td>BRAKE RETURN</td>
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<tr>
<td>15K MANNED LUNAR</td>
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<td></td>
<td></td>
</tr>
<tr>
<td>AEROBRAKE DIA FT</td>
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<td>40</td>
<td>44</td>
</tr>
<tr>
<td>AEROBRAKE MASS</td>
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<td>1923</td>
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<td>STAGE DRY WT, LB</td>
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<td>9925</td>
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<tr>
<td>W</td>
<td>0.146</td>
<td>0.077</td>
<td>0.066</td>
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<tr>
<td>BRAKE RETURN</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>V. CIVIL DESIGN IMPACT</td>
<td>GOOD WITH STORABLE PROP, TAMAND AND FLEX TIPS</td>
<td>OVERSIZED FOR MANY MISSIONS, INTEGRATED CONCEPT OPTIMIZED WITH PARALLEL TANKS</td>
<td>GOOD FOR ACC USE END ASCENT LOADS, NOT TANK CONSTRAINT (4 TANKS BEST)</td>
</tr>
<tr>
<td>CONFIGURATION</td>
<td>(PARALLEL TANKS INCREASE LENGTH, PLUMBING &amp; RESIDUAS)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>GOOD</td>
<td>OVERSIZED</td>
<td>GOOD FOR ACC</td>
</tr>
<tr>
<td></td>
<td>USE END ASCENT LOADS, NOT TANK CONSTRAINT</td>
<td>OPTIMIZED WITH PARALLEL TANKS</td>
<td>(4 TANKS BEST)</td>
</tr>
<tr>
<td>CONCERNING RISKS</td>
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<td></td>
<td></td>
</tr>
<tr>
<td>ITPS</td>
<td>SINGLE REUSE ASSY JOINTS LOCAL &amp; GLOBAL LOW A P FLEX TIPS LOW RADIATION TRAP PACKAGING</td>
<td>ASSY JOINTS ON-OPERT ASSY PA WAKE HEAT PROVEN TIPS LOCAL &amp; FLEX TIPS REUSE ASSY JOINTS</td>
<td></td>
</tr>
<tr>
<td>CONTROL</td>
<td>- PREENTRY SPIN TURNDOWN RATIO LIMIT DEFLATION</td>
<td>SIDE FIRING ENGINES ASCENT CG OFFSET FOR LONG RETRIEVED PT</td>
<td>- CG TRIM ERROR ACS LOCATION</td>
</tr>
<tr>
<td>BASIC FEASIBILITY</td>
<td>MODERATE SHAPE STABILITY FLEX TIPS</td>
<td>LOW ON-OPEPT ASSY &amp; MAINTENANCE</td>
<td>MODERATE FLEX TIPS MAINTENANCE</td>
</tr>
<tr>
<td>WEIGHT GROWTH</td>
<td>MODERATE 8200 LB MAX RETURN WEIGHT FOR 50 FT DIA</td>
<td>LOW BLOCK CHG TO INCREASE TANK OR BRAKE SIZE RETURN PT SHAPE &amp; SIZE VARIABLE</td>
<td>LOW MODERATE FOR DIA INCREASE COMPACT STAGE HAS CG MARGIN FOR RETURN GROWTH</td>
</tr>
</tbody>
</table>
4.3 MAIN ENGINE TRADE

Cryo Engine Selection -- The main engine candidates for a cryogenic OTV fit into three classes, as shown in Figure 3.0-4. They can be derivatives of the RL-10 technology, composed of advanced technologies, or established at an initial entry point into the advanced technology. The best selection depends on the use anticipated over the coming decades. We have established our recommendations, as directed by MSFC, based on the Rev. 8 "low" OTV mission model. We performed a comprehensive comparison of the various options that is reported in its entirety in Volume III and supported in Volume II, Book 3. Figure 4.3-1 shows the development of benefits in discounted dollars as mission usage increases. This comparison was made against the RL-10-A-3 as a reference -- this case forms the zero-benefit line on the chart. Five specific engine development possibilities were conceived for comparison with this reference case. The RL10-IIIB at 460 seconds specific impulse and 15000 pounds thrust is representative of the RL10 derivatives. The RL10-IIIB at 470 seconds and 7500 pounds in either single or dual installation is programmatically little different. The advanced engine is characterized at 483 seconds specific impulse, 7500 pounds thrust with a 10 hour service life. The IOC engine represents the lower end of the specific impulse range achievable by new higher chamber pressure engine technology, lower mission life qualification, and less sophisticated capabilities such as continuous throttling and condition monitoring. This is an engine that has a clear evolutionary path to the advanced engine. It is characterized by a 475 specific impulse, a 7500 pound thrust, and a 5 hour life. The other two candidates on the chart show the impact of transitioning from the RL-10 derivative or the IOC engines to the advanced engine. Figure 4.3-1 indicates that the highest front end funding produces the most benefits at the end of the low mission model.

![Figure 4.3-1 Engine Payback Comparison](image-url)
This result must be considered in the light of other considerations such as those shown in Table 4.3-1. These data indicate the RL-10 is 28% better than the IOC engine for return on investment, and 44% lower than the IOC for DDT&E and production and over 40% less expensive in peak funding. The advanced cryogenic engine is 21% better than the RL-10/ADV in benefits, lower in LCC by 11% over the IOC, and lower in engine cost/flight. Payback based on a 54 flight break even for the RL-10-IIB is 11 additional flights for the IOC and 14 additional for the advanced engine. These data indicate the low number of missions biases towards a derivative engine, and the significant advantages of an advanced engine are most beneficial over the long term with increased missions.

Table 4.3-1 Cryo Main Engine Trade Results

<table>
<thead>
<tr>
<th>MAIN ENGINE OPTIONS</th>
<th>ROI (PV)</th>
<th>BENEFITS (PV)</th>
<th>DDT&amp;E + PROD (PV)</th>
<th>LCC (PV)</th>
<th>PEAK FUNDING</th>
<th>ENGINE COST/FLT (PV)</th>
<th>PAYBACK NO. MISSIONS</th>
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<tr>
<td>RL-10/ADV</td>
<td>1</td>
<td>4.49</td>
<td>251</td>
<td>2104</td>
<td>40</td>
<td>59</td>
<td>90</td>
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<tr>
<td>IOC/ADV</td>
<td>2</td>
<td>4.24</td>
<td>255</td>
<td>2083</td>
<td>29</td>
<td>58</td>
<td>82</td>
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<tr>
<td>ADV</td>
<td>3</td>
<td>5.35</td>
<td>251</td>
<td>2078</td>
<td>52</td>
<td>55</td>
<td>68</td>
</tr>
<tr>
<td>RL-10</td>
<td>4</td>
<td>1.3</td>
<td>70</td>
<td>2213</td>
<td>14</td>
<td>66</td>
<td>54</td>
</tr>
<tr>
<td>IOC</td>
<td>5</td>
<td>2.73</td>
<td>143</td>
<td>2122</td>
<td>25</td>
<td>62</td>
<td>65</td>
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</table>

<table>
<thead>
<tr>
<th>RELATIVE SCORE (10 IS BEST)</th>
</tr>
</thead>
<tbody>
<tr>
<td>RL-10/ADV</td>
</tr>
<tr>
<td>IOC/ADV</td>
</tr>
<tr>
<td>ADV</td>
</tr>
<tr>
<td>RL-10</td>
</tr>
<tr>
<td>IOC</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>ROI (PV)</th>
<th>BENEFITS (PV)</th>
<th>DDT&amp;E + PROD (PV)</th>
<th>LCC (PV)</th>
<th>PEAK FUNDING</th>
<th>ENGINE COST/FLT (PV)</th>
<th>PAYBACK NO. MISSIONS</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.1</td>
<td>8.4</td>
<td>2.8</td>
<td>9.6</td>
<td>3.5</td>
<td>9.3</td>
<td>6</td>
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<td>6.6</td>
<td>8.9</td>
<td>2.7</td>
<td>9.7</td>
<td>4.8</td>
<td>9.4</td>
<td>6.6</td>
</tr>
<tr>
<td>8.5</td>
<td>10.0</td>
<td>2.8</td>
<td>10.0</td>
<td>2.7</td>
<td>10.0</td>
<td>7.9</td>
</tr>
<tr>
<td>10.0</td>
<td>3.0</td>
<td>10.0</td>
<td>9.1</td>
<td>10.0</td>
<td>8.3</td>
<td>10.0</td>
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<td>7.0</td>
<td>4.9</td>
<td>5.6</td>
<td>9.3</td>
<td>5.6</td>
<td>8.9</td>
<td>8.3</td>
</tr>
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</table>
The RL-10 derivatives represent existing technology with ongoing product improvement providing an OTV engine at minimum DDT&E cost. Based on the Revision 8 low mission model, the return on investment (ROI) was the highest with a 54 mission payback. The current engine is flight proven without a single mission failure. However, this technology has the highest Life Cycle Cost (LCC), limited growth and the lowest benefits for a future OTV. Alternatively, an advanced engine has the lowest LCC, lowest cost per flight (CPF), and has the greatest benefits over all the competition. The payback period is 68 missions. The front end program cost is high to achieve this technology and could incur schedule risk depending on the level of technology investment and accomplishment prior to ATP.

An OTV engine using low risk technology improvement provides an attractive alternative to the near term technology, low DDT&E approach. It provides the long term benefits of an advanced engine cycle at low risk. Growth capability is retained while keeping LCC, ROI and DDT&E competitive. The payback period is only 11 missions greater than the RL 10 derivative IIB/IIIB and slightly shorter than the advanced technology engine. While the IOC engine has a high cost per flight, it has a 72% improvement in benefits compared to the RL 10-IIB. We recommend this initial operational capability approach because it offers the OTV program a high performance, low front end cost engine with planned growth potential. This provides the opportunity to improve efficiency, performance, cost per flight, and capacity for future OTV delivery, planetary, and manned missions.

Thrust Level vs. Perigee Burns -- Figure 4.3-2 shows the propellant required to perform the 20 klb delivery mission as a function of OTV main engine thrust level and number of perigee burns for Rocketdyne (a) and Pratt and Whitney (b) engine data. Note that optimum thrust level decreases as the number of perigee burns increases. The relative mission cost of multiple burns shown in Figure 4.3-3 was estimated based on: the indicated optimum thrust level; propellant use at $1500/pound; more frequent engine changeout as thrust is decreased; and increased operational cost and higher mission loss cost as mission duration increases with the number of perigee burns. The net effect is less than $1M per flight regardless of the number of perigee burns. Thus multiple burns yields only a small savings that would limit the growth of OTV for planetary and lunar missions and that would increase mission complexity. Any decrease in propellant delivery cost, which is anticipated in the event that ET propellant scavenging proves feasible, will reduce or eliminate this savings. We elected to size thrust level based on a single perigee burn, selecting a higher thrust that would allow for growth and reduce velocity losses for the planetary and lunar missions. These considerations resulted in recommending a thrust level of 7500 pounds for the OTV main engine.
Figure 4.3-2 Cryo OTV Thrust Trade

Figure 4.3-3 Multiple Burn Cost Trade
Engine Life - The optimum engine life was determined based on the cost of maintenance and engine life development and testing (assumed at $3M/hr). Engine replacement cost for depot level maintenance was assumed in this analysis with one overhaul over the engine's useful life. The Revision 8 mission model was used and the LCC reflects engine replacements beginning in 1995 at an average cost of $10.93M. The results shown in Figure 4.3-4 indicate an optimum MTBO of 7.5 hours (low) with small savings after 5 hrs. While engine life is sensitive to the number of missions, the effect of the number of units on engine recurring cost was not considered.

Figure 4.3-4 Optimum Engine Life
Recommended Cryo Engine Requirements - The requirements for our recommended state-of-the-art (SOA) liquid hydrogen/liquid oxygen OTV engine are given in Table 4.3-2. They were derived from our system analysis of the current engine designs. The Isp is based on an economic analysis of a 20,000 lbm payload, 2-engine stage, and a single perigee burn. The dimensions were based on the engine optimization done for both the Pratt & Whitney RL-10 and the Rocketdyne engine. The major driver for engine dimensions is their affect on the aerobrake diameter and consequently its mass. The engine exit diameter effects the spacing between engines and gimbal requirements with attendant impact on stage length, aerobrake diameter, and engine doors. Engine stowed length directly effects both the stage length and aerobrake diameter. Autogenous pressurization was selected for the space-based OTV because of the potential complication of filling an OTV in low-g with a noncondensible gas such as helium present in the tank. This was the driver for the Tank Head Idle (THI) which can eliminate the need for prepressurization prior to engine start. The NPSH values where based on current engine designs and were not optimized. The THI inlet pressure was based on the conditions at which the propellant is stored. Discrete throttling capable of meeting Rev. 8 mission model requirements was selected because continuous throttling will complicate engine development and increase costs. Aerobrake requirements establish the time available to retract nozzles or to eject a failed nozzle and close protective doors. The development cost was based on discussions with the engine manufacturers, affordable engine technology, 5 hr life, and engine changeout as a complete unit. The cost represents our estimate based on data provided by Aerojet and Rocketdyne.

Table 4.3-2 Recommended Engine Requirements

<table>
<thead>
<tr>
<th>REQUIREMENT</th>
<th>RATIONALE</th>
</tr>
</thead>
<tbody>
<tr>
<td>PERFORMANCE</td>
<td>2.475 sec, 6:1 hr</td>
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<tr>
<td>THRUST</td>
<td>7500 lbf</td>
</tr>
<tr>
<td>MASS</td>
<td>280-300 lbm</td>
</tr>
<tr>
<td>DIMENSIONS</td>
<td>50&quot;</td>
</tr>
<tr>
<td>DIAMETER</td>
<td>&lt; 60&quot; stowed &lt; 120&quot; extended</td>
</tr>
<tr>
<td>LENGTH</td>
<td>50&quot;</td>
</tr>
<tr>
<td>PRESSURIZATION AND CHILDDOWN</td>
<td>60/80 psia</td>
</tr>
<tr>
<td>THROTTLING</td>
<td>STEP THROTTLING, 50 sec</td>
</tr>
<tr>
<td>AEROBRACKET IMPACTS</td>
<td>LAST Firing 1 hr before aero-maneuver, firing 10 min after exit atmosphere</td>
</tr>
<tr>
<td>DEVELOPMENT COST</td>
<td>$175M, 60 mos</td>
</tr>
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</table>
Storable Engine Selection - We compared the pump fed OMS-E engine with one that uses the technology being developed by AFRPL (SLR-132) for use on the storable OTV. Figure 4.3-5 shows the propellant mass required to deliver various payload masses to GEO for stages built around these engines. The OMS based OTV uses a 6000 pound thrust, 334 second specific impulse engine reflecting an increase in nozzle expansion ratio over the STS OMS engine. The SLR-132 based OTV used an optimized 7500 pound thrust engine delivering a specific impulse of 344.1 seconds. Both engines are able to meet the ground-based driver mission. The XLR-132 type engine is recommended, however, because it provides superior performance, it can meet the required IOC, and it provides a clear path to the space-based storable OTV where higher performance is extremely important.

Figure 4.3-5  Ground-Based Storable Engine Selection
4.4 GENERAL ARRANGEMENT TRADES & ANALYSES

A series of trade studies were run to optimize the general arrangement of the OTV configurations. Effort was concentrated on the cryogenic configuration, and the results adapted, where applicable to the storable configurations. The subjects addressed include: 1 vs 1 1/2 stage; 1 vs 2 engines; ground to space commonality; packaging for transport in the cargo bay; and arrangement for space-base assembly and maintenance.

1 VS 1 1/2 STAGE - Parametric evaluation reported in paragraph 4.1 suggested that 1 1/2 stage configurations could have a performance advantage over one stage configurations. As a consequence, a more detailed configuration study reflecting the selected 4-tank, aeroassisted concept was run to establish the merit of conceptual variations using drop tanks. Two drop tank configurations were compared with a reference single stage configuration. This reference held 84,000 pounds of propellant packaged in four near spherical, side-by-side tanks. Figure 4.4-1 shows two potential drop tank arrangements. In the first comparison case, four drop tanks are packaged around a downsized set of four fixed side-by-side tanks. In the second case, the drop tanks are added in tandem with the fixed set of side-by-side tanks. Weights were established and propellant capacity adjusted until the GEO performance capability of the drop tank configurations equaled the capability of the original single stage configuration. The results are indicated in the figure. Both drop tank configurations require greater total propellant usage than the single stage configuration. This result is opposite to the preliminary parametric conclusion reached in Paragraph 4.1, and indicates that, for the selected baseline concept, practical considerations of attachment hardware and aeroassist layout outweigh the theoretical benefit of the drop tank approach. Combining this greater propellant requirement with the cost of the expendable drop tanks makes it clear that the single stage configuration is superior. No further effort was expended on drop tank configurations.

Figure 4.4-1 Cryo Drop Tank Trade
1 VS 2 ENGINES -- It was established that man-rated space-based configurations should have two engines to assure single failure tolerance. This study addressed the feasibility of using two engines on ground-based aft cargo carrier configurations to make evolution to the ultimate space-based configuration more straightforward. This study was conducted using the rather large RL10 derivative configurations. Figure 4.4-2 shows that in order to gimbale through the worst case CG, the two engines must be lowered to the point where the aerobrake extends beyond the maximum allowable ACC envelope. The maximum permissible gimbale angle was established to be 20°. The maximum envelope is based on the special purpose ACC design with a spherical dome extended 7 inches longer than the general purpose ACC design. The engines do not clear the ACC envelope except when gimballed to the full outboard position, and do not leave adequate room for installation of the aerobrake. It was concluded that a single engine configuration is preferred for flight in the ACC, and that two engine commonality through the program is, as a consequent, not practical.

Figure 4.4-2 2-Engine Ground-Based Cryo Packaging
GROUND TO SPACE COMMONALITY — The potential structural commonality between ground and space-based OTV was investigated. The basis for this study was that the ground-based configuration should have one engine and be constrained to fit within the confines of the Aft Cargo Carrier, while the space-based configuration would have two engines. Figure 4.4-3 summarizes a study to determine how much of the ground-based vehicle structure could be used on the space-based vehicle. It was found that only the original center support truss and the structural part of the avionics ring could be counted as truly common. Plumbing attached to the original truss could also be designed to be common. The lower truss and its split plumbing, larger tanks, larger aerobrake and aerobrake supports are all new. Final engine selection and tank size do not affect this result. It was concluded that the space-based structure should be optimized for the space-based application, rather than be compromised to maintain the little commonality that is possible with the ACC constrained single engine configuration.

Figure 4.4-3 Ground-Based to Space-Based Cryo Structure Evolution
CARGO BAY PACKAGING — Both the ground-based ACC OTV and the space-based OTV configurations must be accommodated in the cargo bay. In the case of the ground-based ACC OTV, it must be returned to earth in the cargo bay. In the space-based case, it must be delivered to space in the cargo bay and must be returnable to the earth for major maintenance. In both of these cases transport in segments may be acceptable.

In the case of the ground-based ACC OTV, Mr. Larry Edwards of NASA Headquarters, has conceived an efficient approach that was incorporated in the recommended vehicle design. Figure 4.4-4 illustrates the approach. The bulk of the configuration (the primary structure, LOX tanks, avionics, propulsion and attitude control) is configured in a unit that can be stowed longitudinally in the cargo bay. Keel fittings tie the forward OTV structure to the cargo bay, and the forward LOX tank frames fold partway back and are braced (Section A-A) to provide a torsional load path to the cargo bay longerons. Only the hydrogen tanks must be removed from the flight configuration. They are evacuated after flight, removed from the OTV configuration and stowed fore and aft in the cargo bay. These tanks, which weigh approximately 250 pounds each, provide their own torsional strength, and require the support fittings shown in section B-B. The aerobrake is not retrieved for reuse as the material used, while flexible during ascent, is not anticipated to be flexible after use. The structural ASE required to support this retrieval approach is modest and easily stowed in the bay during the ascent portion of the mission.

Figure 4.4-4 Ground-Based Cryo ASE
Figure 4.4-5 shows the initial delivery of the disassembled space-based OTV to the space-base. As indicated, all subsystems will fit into the orbiter bay, and delivery will require the equivalent of two shuttle flights volume. Since the dry weight of the OTV is on the order of 8000 pounds, it is not advocated that delivery be made in only two flights. Rather, system delivery should be manifested across a larger number of STS flights to achieve full utilization of both Shuttle volume and weight carrying capability.

Figure 4.4-5 Cryo Space-Based OTV Delivery
SPACE ASSEMBLY AND MAINTENANCE — The OTV design has been adjusted to provide for both assembly and maintenance by robotic devices as a primary mode, backed up by remotely operated manipulators and EVA as a contingency mode. Figure 4.4-6 shows the major provisions for grapple fixtures, cradle interfaces and space crane interfaces. These fixtures provide the ability to initially assemble the OTV at the space base, and to perform major parts replacement for maintenance operations. Care has been taken to assure that sufficient space is provided in the vehicle layout to enable these operations with RMS, robotics or space-suited astronauts, as applicable. This requirement has been instrumental in the selection of the open configuration selected for the space-based OTV. Further amplification of this approach is shown in Figure 4.4-7. An octagonal avionics ring was placed at the forward end of the space-based OTVs, providing unobstructed access to all avionics assemblies. The figure shows the locations of these assemblies on the avionics ring. Each replaceable assembly is mounted using RMS type modules that allow removal and replacement with a Module Servicing Tool, which is adaptable to either robotic, RMS or EVA operation. Amplification of assembly and servicing operations is provided in Volume IV.

Figure 4.4-6 Cryo Space-Based OTV Subsystem Servicing Locations
4.5 REDUNDANCY/MAN-RATING

The OTV is to be operated in proximity to the manned Shuttle system from its inception, and is eventually expected to operate in conjunction with the Space Station and to carry men to high orbit. Systems and subsystems must be designed to meet associated safety requirements. In the case of proximity operations, it is necessary to meet the requirements imposed by NASA's safety policy as delineated in NHB 1700.7A. Those systems in use during proximity operations were made dual fault tolerant with respect to credible hardware failures and operator errors. The policy selected for in-flight safety was derived during the course of the study based on the cost of implementing increasingly comprehensive failure policies as illustrated in Figure 4.5-1. Cost increases dramatically with little improvement in system reliability as the most complex policies are implemented. After consideration of these data, NASA direction was to implement a fail safe return policy, where safe return of the crew could be assured in the face of a single failure. This policy must be implemented for manned missions, and may be implemented as much earlier as programmatic considerations indicate to be advantageous.
Figure 4.5-1 Redundancy Trade

The fail safe return philosophy is at a point on Figure 4.5-1 that is intermediate between 'fail safe with rescue' and 'fail-op fail-safe'. The system reliability allocation associated with this philosophy was calculated at 0.994 for a 51 hour space mission and 0.946 for a 480 hour space mission. These overall system reliability requirements led to the redundancy levels incorporated in the subsystem designs.

Table 4.5-1 summarizes the results of the various options considered for man-rating the propulsion system. The results show basic trends and ranges, however, they were completed at different levels of maturity in definition of OTV concepts.
Table 4.5-1 Propulsion System Man-Rating Trade

<table>
<thead>
<tr>
<th>OPTION</th>
<th>CGS (REF. COST)</th>
<th>HWY MASS (Lbm)</th>
<th>ISP (SEC)</th>
<th>MAINTENANCE</th>
<th>MAN-RATING &amp; RELIABILITY</th>
<th>REMARKS</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>REF</td>
<td>REF</td>
<td>REF</td>
<td>SIMPLIST</td>
<td>HOT FAIL-SAFE</td>
<td>SIMPLE VEHICLE DESIGN AND AERODYNAMIC INTERFACE</td>
</tr>
<tr>
<td>2</td>
<td>1.05-1.1</td>
<td>2.6</td>
<td>REF</td>
<td>COMPLICATED</td>
<td>FAIL SAFE EXCEPT FOR THRUST chamber and nozzle</td>
<td>ENGINE DEVELOPMENT TESTING CONCERNS SIMPLE VEHICLE INTERFACES</td>
</tr>
<tr>
<td>3</td>
<td>1.1-1.2</td>
<td>1.2</td>
<td>1200</td>
<td>RCS 400 SEC</td>
<td>COMPLICATED RCS CONDITING SYSTEM</td>
<td>LOW THRUST GEO DEORBIT TRANSPORT LIFE CONCERNS</td>
</tr>
<tr>
<td>4</td>
<td>0.95</td>
<td>0.95</td>
<td>900</td>
<td>REF</td>
<td>THRUST RECONFIGURATION</td>
<td>COMPLICATES DESIGN/DEVELOPMENT MISS 1 &amp; 2 ENGINES FEED SYSTEM</td>
</tr>
<tr>
<td>5</td>
<td>1.2</td>
<td>1.2</td>
<td>450</td>
<td>? TO 2 SEC</td>
<td>2 ENGINES TO MAINTAIN SMALLER VOL. PER ENGINE</td>
<td>COMPLEX CONTROLS LARGER GIMBAL ANGLE</td>
</tr>
<tr>
<td>6</td>
<td>0.90</td>
<td>0.90</td>
<td>750</td>
<td>-2.5 TO 2.7 SEC</td>
<td>3 ENGINES TO MAINTAIN SMALLEST VOL. PER ENGINE</td>
<td>MORE COMPLEX CONTROLS GIMBAL ANGLE SMALLER LARGE AERODYNAMIC DOORS</td>
</tr>
</tbody>
</table>

**ASUMPTIONS:**
- FAIL SAFE - ONE ENGINE OUT MINIMUM REQUIREMENTS FOR MAN-RATING

Option 1 was used as a reference since it represents the minimum unmanned propulsion requirement and high performance.

Option 2 was to back-up the most active component in the engine, the turbopump assembly (TPA). This also included valves and ignitor, but excluded the injector, nozzle, and thrust chamber and might not be considered totally fail safe. It is an approach similar to the Apollo program which had a different requirement.

Option 3 was to use an independent RCS back-up. The common RCS would provide back-up for de-orbit from GEO. The propellant margin would be carried as dry mass on the manned mission only, and depended upon the Isp difference, stage/capsule mass, and mission (GEO or Lunar).

Option 4 considered using a second engine for the manned mission while retaining the benefit of the single engine performance for unmanned missions. Sizing the vehicle to accept 1 or 2 engines resulted in a dry mass penalty canceling any benefit.
Option 5 used a two engine vehicle optimized for the GEO delivery missions. This provided fail safe return of the stage for all missions. The vehicle and payload are immediately returned in the case of a failure at perigee. The subsequent burns do not have significant velocity losses for half thrust, therefore, the mission could be completed and the stage returned. For a manned mission, an engine failure would always abort the mission. The engine performance depends upon the area ratio, but in any case there was some performance penalty over a single engine for the two sets of engine data used.

Option 6 investigated adding more than 2 engines. For a single engine out capability, the reliability was lower. For two engine out capability, the stage mass increased without an offsetting increase in performance.

The conclusion was to provide compete engine back-up for the manned GEO and Lunar missions. The back-up TPA is an attractive option because of the design simplification, but the question would be: Is a single thrust chamber, injector, and nozzle fail safe? The RCS back-up requires a large propellant margin and a higher mission loss rate for unmanned missions. The two engine stage was selected because it was fail safe and minimized the performance penalty.

The avionics and power equipment used in the ground and space-based OTV is summarized in the Figure 4.5-2. The component redundancy levels are indicated. The level required for the short duration unmanned ground-based missions is somewhat less than that deemed necessary for the manned space-based vehicles. In the space-based vehicles, we found that the redundancy required by man-rating (a fail safe return philosophy) was somewhat greater than the redundancy suggested by mission 'lost cost' considerations. We elected to incorporate man-rating redundancy in all space-based configurations as indicated in the space-based column of the chart, since our analyses indicated it was not economically desirable to maintain two different avionic configurations in the space-based program. Details on the selection of these subsystems are presented in Vol. II, Book 3.
<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Component</th>
<th>Ground Based</th>
<th>Space Based</th>
</tr>
</thead>
<tbody>
<tr>
<td>Guidance</td>
<td>Star Scanner</td>
<td>1</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>Star Tracker</td>
<td>N/A</td>
<td>2</td>
</tr>
<tr>
<td>NAVIGATION AND CONTROL</td>
<td>IMU</td>
<td>1</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>GPS Receiver</td>
<td>1</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>GPS Antenna - Low Alt</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>GPS Antenna - High Alt</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>Flight Controller</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>Executive Computer</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td>DATA MANAGEMENT</td>
<td>Condition Monitor</td>
<td>N/A</td>
<td>1</td>
</tr>
<tr>
<td>TELEMETRY AND COMMAND</td>
<td>Command &amp; Data Handling</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>TLM Power Supply</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>Deploy Timer</td>
<td>2</td>
<td>N/A</td>
</tr>
<tr>
<td>COMMUNICATIONS AND TRACKING</td>
<td>STDN/TDRS Xponder</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>20W RF Power Amp</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>S-BAND RF System</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td>ELECTRIC POWER SYSTEM</td>
<td>Fuel Cell (FC)</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>FC Radiators</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>EM Reactants Tanks</td>
<td>1 SET</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>FC Water Storage</td>
<td>1</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>Power Control &amp; Distribution</td>
<td>2</td>
<td>2</td>
</tr>
</tbody>
</table>

Figure 4.5-2 Avionics and Power System Redundancy
### 4.6 SUBSYSTEM SELECTION SUMMARY

A summary of the subsystem trade studies the results and resulting selections is given in the following pages. The detailed evaluations leading to these results are included in Volume II, Book 3.

**AEROASSIST** — The key mission requirement driving aerobrake design for geostationary missions is the weight to be retrieved. The Revision 8 mission model requires return of an empty OTV, return with a 4500 pound manned servicer, or return with a 7500 pound manned capsule. Table 4.6-1 shows aerotherm parameters associated with all the return cases for the ground based OTV using a 38 foot aerobrake. These ground based cases show the trend towards higher temperatures and higher loading as return weight increases. While the 38 foot brake is adequate for the ground based vehicle, we elected to use a 40 foot aerobrake in order to achieve increased design margins. The two space based cases in the table show a minimum sized (38 foot diameter) brake with the early unmanned servicer return mission, and a larger aerobrake with the manned capsule. We elected to use the larger aerobrake throughout the space based program because: We prefer the simplicity of basing only one brake design at the space station; we prefer the increased design margins associated with the larger brake; we like the growth potential for heavier lunar return missions and potentially heavier manned capsules.

#### Table 4.6-1 — OTV Aerobrake Definition

<table>
<thead>
<tr>
<th>CONFIGURATION</th>
<th>W/CpA</th>
<th>BRAKE DIAMETER (FT)</th>
<th>RETURN WEIGHT LB</th>
<th>qMAX* (BTU/FT²SEC)</th>
<th>T* MAX (OF)</th>
<th>DES LOAD (PSF)</th>
</tr>
</thead>
<tbody>
<tr>
<td>GROUND BASED DELIVERY (RETURN EMPTY)</td>
<td>3.20</td>
<td>38</td>
<td>5,700</td>
<td>15.2</td>
<td>2150</td>
<td>22 16</td>
</tr>
<tr>
<td>GROUND BASED W/UNMANNED SERVICER</td>
<td>6.13</td>
<td>38</td>
<td>10,970</td>
<td>22.8</td>
<td>2460</td>
<td>35 26</td>
</tr>
<tr>
<td>GROUND BASED W/MANNED SERVICER</td>
<td>8.05</td>
<td>38</td>
<td>14,400</td>
<td>25.1</td>
<td>2555</td>
<td>47 35</td>
</tr>
<tr>
<td>EARLY SPACE BASED W/SERVICER</td>
<td>6.44</td>
<td>38</td>
<td>11,530</td>
<td>23.4</td>
<td>2485</td>
<td>36 27</td>
</tr>
<tr>
<td>GROWTH SPACE BASED W/MANNED CAPSULE</td>
<td>6.00</td>
<td>44</td>
<td>15,670</td>
<td>21.4</td>
<td>2390</td>
<td>34 26</td>
</tr>
</tbody>
</table>

**MAN SORTIE** — 7,500 lbs CAPSULE, 14 1/2' W x 15' L

* FLEX TPS
PROPULSION -- The major conclusions reached in the propulsion subsystem areas are summarized in Table 4.6-2. The sensitivities shown indicate the factors that drove the decisions. A near term version of the advanced engine was selected to minimize development cost while providing a straight forward evolutionary path to more advanced capability should greater requirements evolve. The man-rating decision, as discussed in section 4.5, was driven by the fail-safe-return man-rating requirement established. Engine thrust, payload mass and number of perigee burns are interrelated as discussed in Section 4.3, and result in the selection reached. The engine throttling requirement is established by the 0.1g payload acceleration mission requirement coupled with the impact of the other system decisions indicated. Engine life reflected a programmatic trade between development cost and frequency of operational replacement. The transition of reaction control system selection is driven by the near term need for economy in the initial ground-based program, and the severe mission requirements imposed by the space-based manned missions.

Table 4.6-2  Propulsion System Conclusions/Recommendations

<table>
<thead>
<tr>
<th>TRADE/ANALYSIS</th>
<th>CONCLUSION</th>
<th>SENSITIVITY</th>
</tr>
</thead>
<tbody>
<tr>
<td>MAIN ENGINE</td>
<td>IOC ENGINE</td>
<td>MISSION MODEL &amp; ENGINE DDT&amp;E vs PERFORMANCE</td>
</tr>
<tr>
<td></td>
<td>&gt;475 sec. 5 HRS LIFE, DDT&amp;E $175 M</td>
<td>ADVANCE ENGINE PROGRAM SHOULD DEVELOP PROTOTYPE ENGINE</td>
</tr>
<tr>
<td>MAN-RATING</td>
<td>2 ENGINES</td>
<td>MANNED SAFETY REQUIREMENTS</td>
</tr>
<tr>
<td>ENGINE THRUST</td>
<td>15000 lbf TOTAL, 2500 LBF/EA SINGLE PERIGEE BURN</td>
<td>NUMBER PERIGEE BURNS, PAYLOAD MASS, &amp; NUMBER OF ENGINES</td>
</tr>
<tr>
<td>ENGINE THROTTLING</td>
<td>STEP THROTTLING 7.5K ENGINE TO 3.2K</td>
<td>0.1 G LEVEL, PAYLOAD MASS, NUMBER LOW-G MISSIONS, NUMBER OF ENGINES &amp; ENGINE PERFORMANCE</td>
</tr>
<tr>
<td>ENGINE LIFE</td>
<td>5 HRS &amp; ORU</td>
<td>REPLACEMENT ON-ORBIT COST &amp; MISSION MODEL SIZE</td>
</tr>
<tr>
<td>REACTION CONTROL</td>
<td>HYDRAZINE ON GB COMMON GH2/GO2 ON SB</td>
<td>MISSION MODEL SIZE, GH2/GO2 DDT&amp;E, &amp; DEVELOPMENT OF THRUSTER &amp; COMMON FEED TECHNOLOGY</td>
</tr>
</tbody>
</table>

4-37
STRUCTURES -- Configuration and structural trades conducted in this study are described in detail in Volume II, Book 3, paragraph 2.4. Those trades dealing with selection of general arrangement are summarized in paragraph 4.4 of this volume. Other structures trades are summarized in Table 4.6-3.

After a thorough evaluation of available composite and metallic materials, composites were selected for all primary and secondary structural elements, with the exception of propellant tankage. Graphite epoxy was selected for structures below 300 degrees Fahrenheit because of its light weight and ease of fabrication. Graphite polymide was selected for structure above 300 degrees Fahrenheit, which is required to support the aerobrake thermal shield. This material is able to retain strength at a temperature of 600 degrees Fahrenheit, which in turn establishes thickness requirements on insulation. We selected 2090 aluminum/lithium alloy for cryogenic tanks. This material is expected to display the excellent low temperature and weldability characteristics of the 2219 alloy used for the external tank, while providing significantly lighter weight. We selected 15(V)-3(Cr)-3(Al)-3(Sn) titanium for storable propellant tankage. This is a new alloy that will require further testing but it displays encouraging initial results relative to forming and repairing welding when compared to 6AL4V titanium.

The OTV configurations developed provide adequate protection against the anticipated meteoroid environment. A 0.006 aluminum meteoroid bumper with MLI tank thermal insulation serving as a particle catcher was found adequate for the space based vehicle. The shorter duration ground based missions resulted in a vehicle that was adequately protected with the MLI alone. Subsequent analyses conducted in the extension study indicate that a bumper is required on the ground based vehicle, and a thicker bumper is required on the space based vehicle to provide added protection against the debris environment being defined for space station design. This refinement is not reflected in this volume.

Additional studies were made to establish the appropriate umbilical arrangement and structural interface with the Aft Cargo Carrier. The top level results of these studies are indicated in Table 4.6-3, and more detail is provided in Volume II, Book 3 as indicated.
### Table 4.6-3 Structural Design Trade Summary

<table>
<thead>
<tr>
<th>STRUCTURAL TRADE</th>
<th>KEY ISSUES</th>
<th>RECOMMENDATION</th>
<th>REF</th>
</tr>
</thead>
<tbody>
<tr>
<td>COMPOSITE SELECTION</td>
<td>◦ OPERATING TEMPERATURE ◦ STRUCTURAL CHARACTERISTICS ◦ FABRICATION CONCERNS</td>
<td>◦ USE GRAPHITE POLYIMIDE FOR AEROBRAKE ◦ USE GRAPHITE EPOXY FOR BASIC STRU.</td>
<td>VOL. II BK. 3 PAR. 2.4.4</td>
</tr>
<tr>
<td>METAL SELECTION</td>
<td>◦ LOW TEMPERATURE STRENGTH ◦ TOUGHNESS ◦ PROPPELLANT COMPATIBILITY ◦ FABRICATION CONCERNS</td>
<td>◦ USE 2090 AL/LI FOR CYRO TANKS ◦ USE 15-3-3-3 TITANIUM FOR STORABLE PROPPELLANT TANKS</td>
<td>VOL. II BK. 3 PAR. 2.4.5</td>
</tr>
<tr>
<td>METEOROID PROTECTION</td>
<td>◦ METEOROID ENVIRONMENT ◦ PROTECTION CRITERIA ◦ WEIGHT &amp; VOLUME ◦ FABRICATION</td>
<td>◦ USE NO PROTECTION ON GND BASED (SUBSEQUENTLY CHANGED FOR DEBRIS) ◦ USE ALUMINUM BUMPER &amp; MLI FOR CATCHER ON SPACE BASED</td>
<td>VOL. II BK. 3 PAR. 2.4.10</td>
</tr>
<tr>
<td>UMBILICAL LOCATION</td>
<td>◦ INSPECTABILITY ◦ USE AT UMBILICAL (ICD80900000025) ◦ ACC CRUCIFORM CROSSBEAM (DMR26AP00231) ◦ STRUCTURAL INTEGRITY AND PHYSICAL FIT</td>
<td>◦ PENETRATE ACC SKIRT TO PLATES AT CRUCIFORM &amp; LOX TANKS ◦ SEPARATE FLUID &amp; ELECTRICAL UMBILICALS</td>
<td>VOL. II BK. 3 PAR. 2.4.3</td>
</tr>
<tr>
<td>ACC BEAM STIFFNESS</td>
<td>◦ MAXIMIZE PAYLOAD ◦ OPTIMUM ACC BEAM (GROUND BASED ACC/OTV)</td>
<td>◦ 25.5 INCH DEEP PARALLEL CRUCIFORM BEAMS SAVES 18# IN OTV, 110# IN ACC RELATIVE TO ORIGINAL TAPERED BEAMS</td>
<td>VOL. II BK. 3 PAR. 2.4.1</td>
</tr>
<tr>
<td>ACC STRUCTURAL ATTACHMENT</td>
<td>◦ ACC/OTV STRUCTURAL I/F ◦ RESTRAINTS AT 4 ATTACHMENTS ◦ WEIGHT &amp; DEFLECTION</td>
<td>◦ USE 10 DOF RESTRAINT (TRADED WITH A BEP) ◦ ADD LATERAL RESTRAINT AT LH2 TANK ◦ SAVES 75# &amp; REDUCES DEFL. 2&quot;</td>
<td>VOL. II BK. 3 PAR. 2.4.2</td>
</tr>
</tbody>
</table>

4-39
AVIONICS — Figure 5.6-4 summarizes trade studies performed for all major functional elements in each of the five avionics subsystems: 1) GN&C, 2) DMS, 3) C&T, 4) T&C, and 5) EPS. No unusual results were obtained as a result of these trades. Basic technology advances in disciplines supporting avionics hardware such as microelectronics, opto-electronics, semiconductors, and computer architecture will ensure a continued growth in capability and reliability while maintaining relatively low cost between the present and the OTV Phase B/C/D period. The inertial guidance system for the ground-based OTV could make use of the ring laser gyro technology should schedules push the IOC farther out. The use of redundant, propellant grade reactant fuel cells was selected to reduce logistics and maintenance costs, among other factors, while offering sufficient reliability for the DRMs. Individual electronic subsystems, such as memories, are equipped with built-in battery back-up power. While not mandatory, GPS and TDRS improvements would enhance the OTV program. Addition of an aft (upward) looking antenna on GPS would significantly improve gain margins in obtaining GPS updates at GEO altitude. While TDRS coupled with ground coverage provides adequate OTV command capability, use of a third TDRS and increase of its azimuth steering angle to +45° would significantly improve TDRS coverage in the absence of support from ground stations.

Table 4.6-4 Avionics Trade Summary

<table>
<thead>
<tr>
<th>OTV RF COMMUNICATIONS ALTERNATIVES</th>
<th>BALL ELECTRICALLY SWITCHED STEERABLE ANTENNA</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>20W RF POWER AMP PREFERRED</td>
</tr>
<tr>
<td>GN&amp;C STATE VECTOR UPDATE</td>
<td>GPS IS THE PREFERRED OPTION FOR ALL STATE VECTOR UPDATES</td>
</tr>
<tr>
<td>MICROPROCESSOR / MICROCOMPUTERS</td>
<td>FAIRCHILD 9450 PREFERRED ARCHITECTURE</td>
</tr>
<tr>
<td>CENTRALIZED vs. DISTRIBUTED</td>
<td>DISTRIBUTED, NETWORK ARCHITECTURE PREFERRED</td>
</tr>
<tr>
<td>COMPUTER DATA MANAGEMENT</td>
<td>DELCO MAGIC V PREFERRED EXECUTIVE COMPUTER</td>
</tr>
<tr>
<td>ON-BOARD vs. GROUND CHECKOUT</td>
<td>ON-BOARD CHECKOUT PREFERRED</td>
</tr>
<tr>
<td>BUILT-IN vs. MULTIPLE UNIT AVIONICS</td>
<td>BUILT-IN, LAYERED FAULT-TOLERANCE</td>
</tr>
<tr>
<td>BLACK BOX REDUNDANCES</td>
<td>APPROACH REDUCES BOX REDUNDANCIES</td>
</tr>
<tr>
<td>ELECTRO-OPTICAL NAVIGATION SENSORS</td>
<td>SOLID STATE STAR TRACKER PREFERRED (GB USES EARLIER STAR SCANNER TECHNOLOGY)</td>
</tr>
<tr>
<td>ELECTRICAL POWER GENERATION TECHNOLOGY</td>
<td>FUEL CELL PREFERRED</td>
</tr>
<tr>
<td>GYRO TECHNOLOGY</td>
<td>RING LASER KYRO PREFERRED OVERALL</td>
</tr>
<tr>
<td></td>
<td>DRIRU SUITABLE FOR NEAR TERM USE</td>
</tr>
</tbody>
</table>

4-40
5.0 CONCEPT SELECTION

5.1 HIGH POTENTIAL CRYOGENIC SELECTION

An initial concept selection was made at contract midterm to accommodate the 'nominal' Rev. 7 mission model. Subsequent to this selection, MSFC produced a Rev. 8 mission model and directed that development recommendations be justified by the 'low' version of this model. We found that while the 'nominal' Rev. 7 model suggested these OTV development steps (a ground-based vehicle, an initial space-based vehicle, and a growth space-based vehicle), the 'low' Rev. 8 model could be accommodated by the first two of these steps.

Figure 5.1-1 shows the family of cryogenic stages we recommended to capture the Rev. 7 nominal mission model. The ground-based stage is sized at 45,000 pounds propellant capacity to fully utilize STS payload capability when launched in the aft cargo carrier. It would be used to perform single and multiple delivery missions until the initial space-based configuration was to be introduced. We recommended that this stage employ a single 7500 pound thrust advanced technology engine. The configuration is tightly packaged to fit assembled in the aft cargo carrier, and uses a foldable 40 foot diameter fabric covered aerobrake. The aerobrake is designed to support empty stage return at a maximum surface pressure of 23 psf.

Figure 5.1-1 Cryo Configuration Summary
The initial space-based configuration is derived from the ground-based stage. Its 55,000 pound propellant load was selected to support the driving 20,000 pound delivery mission. It utilizes two engines of the same type, and most of the same avionics components, as the ground-based vehicle. We believe that the general arrangement must be opened up to facilitate maintenance in space. Mission duration is increased to 10 days to support the unmanned servicing mission. A 44 foot aerobrake is required to protect the open configuration, and its 35 psf design pressure supports return with the unmanned servicer.

A growth stage would have been required to support the Rev. 7 manned GEO mission and the larger lunar missions. Both the initial and the growth stages would have been maintained at the Space Station from IOC of the growth stage throughout space-based operations. The growth stage's 81,000 pound propellant load was selected to support the driving Manned Lunar Sortie in a two stage configuration. This is slightly larger than the 75,000 pound load that would have been required to perform the Revision 7 Manned GEO Sortie, but our preliminary programmatic trades indicate the selection of the slightly larger stage would have been cost effective. The mission duration of the growth stage would have been up to 24 days as required by the 14,000 pound Revision 7 Manned GEO Sortie. A 44 foot aerobrake designed for a 63 psf peak pressure enable return with a manned capsule. The two-stage configuration would have been required to support the Manned Lunar Sortie, where 65,000 pounds of payload is delivered in conjunction with a 15,000 pound roundtrip manned sortie.

As previously noted, only the ground-based stage and the initial space-based stage are required to perform the 'low' Revision 8 Mission Model. All three configurations are described in Section 6.1.

5.2 HIGH POTENTIAL STORABLE SELECTION

Figure 5.2-1 is a pictorial presentation of the complete storable OTV family of high potential stages that was selected to perform the 'nominal' Rev. 7 OTV mission model. This selection was not updated to meet the requirements of the "low" Rev. 8 model because it was programmatically demonstrated (including operational and space-basing impacts) that the storable concepts were less desirable than the cryogenic concepts, even with the low use rates involved in the "low" Rev. 8 model. This family of stages will perform the missions identified in the Rev. 7 model with the exception of the heavy lunar missions in the post 2006 timeframe. We proposed to capture those missions using a low technology cryo perigee stage. Two configurations for the early ground-based OTV are defined on the left of the figure; one carried aloft in the ACC and the other configured to fit in the Orbiter payload bay. Both are sized to take advantage of the total lift capability of the STS in the early 1990s and are outfitted to deliver unmanned single or multiple payloads, as identified in the mission model, to GEO operating as a perigee stage. The space-based family is built around three stages with propellant capacities carefully selected to most efficiently perform the broad range of identified missions. The 53,000 lb capacity stage is the workhorse configuration which has application in all GEO missions. Operating as a perigee stage it is the GEO delivery vehicle for single and multiple
6.0 SELECTED CONCEPTS DEFINITION

6.1 HIGH POTENTIAL CONCEPT DEFINITION — CRYO

6.1.1 INITIALLY GROUND BASED CRYO

6.1.1.1 General Arrangement (Ground Based Cryo) — The overall concept of our selected ground-based cryogenic OTV is shown in Figure 6.1.1.1-1, and a more detailed layout in Figure 6.1.1.1-2. The four tank, single advanced technology engine configuration uses the volume and weight efficient principles suggested by Larry Edwards (NASA Headquarters) to fit easily into the Aft Cargo Carrier (ACC). The 40 foot diameter aerobrake folds forward while stowed in the ACC. It is discarded after flight and not stowed in the orbiter bay for retrieval. The aluminum/lithium propellant tanks are designed by engine inlet pressure requirements. Their thinnest gauges are .018 in. for the LO₂ tank and .014 in. for the LH₂ tank. The tanks are insulated with multi-layer insulation. The hydrogen tanks are removed on orbit after mission completion and, with the core system (LO₂ tanks, structure, avionics, propulsion), are stowed in the orbiter bay for retrieval. The propulsion and avionics subsystems are mounted on the central truss, and reflect essentially a single string design. The major exception is redundancy in those systems that require dual fault tolerance while in the vicinity of the Orbiter. The structure is of lightweight graphite epoxy. The propellant load was selected to enable full utilization of projected STS lift capability on GEO delivery missions.

6.1.1.2 Subsystem Summary Description (Ground-Based Cryo)

6.1.1.2.1 Aeroassist (Ground Based-Cryo) — The overall layout of the ground-based cryo OTV aeroassist device is shown in Figures 6.1.1.1-1 and -2. Details of the construction of its surface insulation and the parameters influencing its design are shown in Figure 6.1.1.2.1-1. The basic shape of the aeroassist device is the 70 degree blunted cone proven on the Mars Viking lander. The 40 foot diameter device is designed to retrieve a nearly empty OTV from geostationary transfer orbit. Its center of gravity is offset to cause it to trim out at a 0.12 lift/drag ratio. This has been shown adequate to provide trajectory control when used with a roll modulation control technique. The 40-foot diameter was selected to provide adequate shielding of the OTV from the aerodynamic wake, for trim angles up to an L/D of 0.20. This size, the weight of the OTV at reentry and the physical properties of the aerodynamic surface establish the temperatures and heat fluxes shown in Figure 6.1.1.2.1-1.

The outer portion of the shield folds forward to fit within the Aft Cargo Carrier, and is constructed of the flexible, multilayer material shown in Figure 6.1.1.2.1 backed by graphite polyimide ribs. These ribs can tolerate temperatures up to 600 degrees Fahrenheit, which establishes the thickness of the insulation shown in the figure. The multilayer Flexible Surface Insulation (FSI) outer layer is a woven Micalon (silicon carbide) fabric which can tolerate high heating rates without becoming brittle. A three dimensional woven structure between the inner and outer surfaces is filled with a ceramic felt insulation. The inner layer is NEXTEL 312, which has superior mechanical properties, impregnated with an RTV gas sealer.
Figure 6.1.1.1-1 Ground Based Cryogenic OTV Concept
payloads. For unmanned roundtrip servicing missions, the 53,000 lb stage is combined with the 25,000 lb stage to form a two stage vehicle. For the demanding requirements of the manned trips to GEO, the 53,000 lb stage is mated to the 90,000 lb stage to form another two-stage configuration. The 53,000 lb stage will be fitted with aerobrakes appropriately sized for the size and weight of the body being returned from GEO. When only the stage is returning, as from delivery missions, only a 25-foot diameter brake is required. When returning from delivery of multiple payloads, the multiple payload carrier returns with the stage and therefore, the required brake size is 32 feet in diameter. Bringing the manned capsule back from the manned servicing mission is the most demanding mission for the 53,000 lb stage and requires a 41-foot diameter brake. The 90,000 lb stage is sized for the first stage application on the manned servicing vehicle; however, it will be the primary vehicle for performing the planetary missions in the mission model. Some of the less demanding planetary missions can be performed by the 53,000 lb and the ground-based stages. The identified application for the 25,000 lb stage in the current mission model is for the second stage of the two-stage unmanned servicing vehicle.

Figure 5.2-1 Storable Configuration Summary
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</tbody>
</table>

**Figure 6.1.1.2.1-1** Ground-Based Cryo Aeroshield

Preceding page blank not filmed.
The central section of the aeroassist device is rigid and is covered with the same type of ceramic tiles used on the Orbiter. The temperatures associated with the low ballistic coefficient of this configuration are low enough to permit use of the Flexible Surface Insulation (FSI), but the current design uses Orbiter tiles instead. A door is provided in the central heat shield so the two step engine nozzle can be deployed through it. Figure 6.1.1.1-2 shows an FSI door that is rolled back from the opening, but a rigid door has been selected to maintain the blunted 70 degree cone basic geometry.

6.1.1.2.2 Propulsion (Ground Based Cryo) - The propulsion characteristics are shown in Table 6.1.1.2.2-1 for the main propulsion and Table 6.1.1.2.2-2 for the reaction control system. The main engine is a single 7500 lb thrust expander cycle, which can be available in 1993 with an accelerated development program. Current advanced engine research funding is scheduled to demonstrate the required technology by 1990. These research engines would then reduce the risk of a development program which is estimated to take 5-7 years. To provide an evolutionary path from ground-based to space-based, we recommend accelerating this program and the development of a new engine for the ground-based OTV. This could be accomplished with a phased program, i.e., a lower technology engine initially that has the capability to evolve to a space-based design. The engine Isp would be 475 (minimum). The engine can provide propulsive settling of propellants by operating at tank head idle (THI—without rotation of turbo machinery). It can operate with saturated propellants at pumped idle (PHI) thrust level and low NPSH at full thrust level. Capacity for both hydrogen and oxygen autogeneous tank pressurization is provided by the engine. The nominal inflight tank operating pressures are 26 psia and 21 psia for LO2 and LH2, respectively. This allows for the negative acceleration head associated with the siphon feed.

The advanced engine has a goal of 300 -500 firings and a time between overhaul of 10 - 20 hours. However, the initial engine would be qualified for only about 5 hrs. The engine development time is 60 months from start of development to the first operational engine. Dual engine installations were also evaluated for the ACC ground-based cryogenic OTV. Layout studies performed at MAF indicated that additional length would be required for the ACC in order to accommodate 2 RL10-IIB or RL10-III engines with the capability to gimbal through the center of gravity and provide a fail safe engine capability. This additional length could not be obtained. This study should be revisited for the small advanced engine, including impacts to cargo bay return.

Propellants are stored in a four tank configuration, two liquid oxygen and two liquid hydrogen. The tanks are manifolded together in a parallel flow configuration so that propellants will be depleted simultaneously from each of the two tanks. In order to deplete the parallel tanks and both propellants simultaneously, a propellant utilization system is included. This system consists of propellant utilization probes that provide continuous liquid level data during main engine firings and discrete point level sensors that provide data to allow the cancellation of cumulative errors in the continuous mode when the liquid level passes the discrete point sensor. The data from each tank is input to the stage computer. The computer outputs signals to either
Table 6.1.1.2.2-1 Ground-Based Cyrogenic MPS Summary

- **ENGINE**
  - SINGLE ENGINE, 7.5K THRUST, $I_s = 475$ SEC, EXPANDER CYCLE

- **PROPELLANT DISTRIBUTION**
  - DUAL TANK PARALLEL FEED START TRAP

- **PRESSURIZATION**
  - AUTOCENGUS FROM ENGINE FOR PUMPED IDLE AND FULL THRUST - NOT REQUIRED FOR TANK HEAD IDLE

- **ENGINE FEATURES**
  - TANK HEAD IDLE (THI) CONDITIONING AND SETTLING, PUMP HEAD IDLE (PHI) FOR LOW THRUST APPLICATIONS
  - 5 HR LIFE

- **VENT**
  - GROUND/ASCENT
  - "O"G (TVS)

- **VALVE ACTUATION**
  - HELIUM, STORED ON STAGE

- **PROPELLANT UTILIZATION**
  - TANK TO TANK AND MR CONTROL

- **CARGO BAY RETRIEVAL**
  - SEPARATION OF LH$_2$ TANKS

- **THERMAL PROTECTION**
  - H$_2$ - 1/2" SOFI 1/2" MLI (25 LAYERS DAK)
  - O$_2$ - 1/2" MLI (25 LAYERS DAK)

- **STS PROXIMITY OPERATIONS**
  - TWO FAULT TOLERANT

- **REdundancy**
  - SINGLE FAILURE TOLERANT EXCEPT FOR ENGINE
Table 6.1.1.2.2-2  Ground-Based RCS Summary

- PROPELLANT
  - Hydrazine ($N_2H_4$)

- ROCKET ENGINE MODULE
  - 30 LB, 7 ENGINES PER MODULE  \( \text{Isp} = 230 \text{ SEC} \)
  - 14 THRUSTERS SCARFED INTO AEROBRAKE
  - 3 DOF and +X TRANSLATION
  - FAIL OPERATIONAL

- PROPELLANT SUPPLY
  - THREE 24" DIAMETER TANKS
  - POSITIVE EXPULSION  \( 400 \text{ LBS OF HYDRAZINE MAXIMUM} \)
  - 400 PSI 2:1 BLOWDOWN

- SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR STS PROXIMITY OPERATIONS
the tank propellant utilization valves to keep liquid levels in each tank the same or to the engine to shift mixture ratio to assure simultaneous depletion of usable propellants. Refillable traps are included in the outlet of each tank to provide liquid propellants for the chilldown until the remaining propellants are settled over the tank outlet by the engine thrust.

The tank vent system consists of both ground vents for loading and low gravity vent systems for flight operations. The flight system uses a mixer and a thermodynamic vent heat exchanger to minimize the operation impacts for in flight venting and reduce propellant thermal stratification. The mixer and heat exchanger similar to the STS Centaur are mounted inside the tanks. The controls and magnetically coupled drive motors are on the outside so that they can be serviced without entering the tanks.

High pressure helium is stored at 3000 psi in a composite overwrapped vessel for MPS engine purges and valve actuation. The stage propellant system valves are also pneumatically activated.

The ground-based cryogenic OTV must be disassembled for return to the ground in the orbiter cargo bay because of its size. The design includes the removal of the LH₂ tanks to provide for sufficient clearance in the cargo bay for the remainder of the stage.

The residual propellants (up to 1.5%) will be burned and dumped in a nonoptimum burn after the perigee raising burn that follows the aeropass maneuver. During that burn, the remaining propellant quantity will be determined to an accuracy of 0.25% so that nonoptimum trajectory burn times can be calculated. The first burn of this maneuver will utilize the MPS engine to consume some portion of the residuals. The second burn will start with the MPS engine and finish with an RCS vernier burn during which the remaining propellant, approximately 260 lbs, will be dumped through 2.5" dump valves in the MPS plumbing system.

This complex propulsive dumping maneuver is required to dump the residual propellants without freezing the residual hydrogen. If the hydrogen were dumped nonpropulsively about 70% of the residuals could freeze when the triple point pressure for hydrogen (1.0 psia) was reached. LO₂ is not as prone to freezing because it has a triple point pressure of about 0.022 psia. Before we selected this propulsive mode of operation we looked at several alternatives which are discussed in Volume II, Book 3.

The thermal protection system is 0.5 inches of SOFI and 0.5 double aluminized Kapton multilayer insulation (MLI) on the hydrogen tank and 0.5 inch MLI on the oxygen tank. Both insulation systems are purged on the ground with low dew point nitrogen to eliminate moisture contamination.

The system is two fault tolerant for inadvertent RCS and main engine ignition for proximity operations near the orbiter.
The system is single failure tolerant except for the engine itself as shown in the MPS schematic, Figure 6.1.1.2.2-1. Figure 4.1.2.2-2 shows the legend that identifies the components shown in the propulsion schematics. The pneumatic system is not shown in Figure 6.1.1.2.2-1.

The reaction control system (RCS) uses hydrazine monopropellant pressurized by nitrogen gas operating in a blowdown mode from 400 psi. Fourteen (14) thrusters provide 3 degree of freedom operation and +X translation. The thrusters are 30 lbf each and are clustered with seven (7) thrusters in each module. The thrusters provide an Isp of 230 seconds. The propellant is stored in three 24-inch diameter tanks, each having a usable propellant capacity of about 133 lbs of hydrazine at a 2:1 blowdown. The RCS is two fault tolerant for proximity operations as shown in Figure 6.1.1.2.2-3.

6.1.1.2.3 Structures & Packaging (Ground-Based Cryo) - The configuration, shown in Figure 6.1.1.1-2, consists of two 132 in. diameter spherical LH₂ tanks with cone ends and two 93 in. diameter spherical LO₂ tanks with cone ends and one advanced design engine that generates 7500 lbs of thrust. The engine and the lower support for the four tanks is provided by a central core truss that also provides the interface at four points with the ACC. Upper ends of the tanks are linked together and tie to the upper part of the truss at the LO₂ tanks. The folding aerobrake attaches at the engine end of the core truss. The brake is folded while the vehicle is in the ACC and is deployed by springs after leaving the ACC. Interface with the ACC is on the end opposite the aerobrake and engine. These points also interface with the payload adapter. Umbilical provisions with ACC are also opposite the aerobrake. Avionics are installed on the center core truss. The vehicle has been designed to be partially disassembled so that it can be returned to earth in the cargo bay of the orbiter after jettisoning the aerobrake.

The cryogenic tanks are of fusion welded construction and are made in two halves from 2090 aluminum lithium alloy. Minimum membrane thickness is .014. If problems are uncovered during testing of the 2090 alloy or in developing forming in two halves, the backup alloy would be 2219 aluminum with backup processing to be four gores per head with machined conical caps. If difficulties are encountered in handling .014 thick tanks, membrane thickness would be increased to what is required for handling. The basic air frame truss and tank support struts are graphite epoxy. The aerobrake support structure is designed to operate at 600°F. The center section is made up of a hexcel honeycomb with graphite polyimide skin, covered with shuttle FRCI-20-12 tiles. The outer portion is a flexible surface insulation composed of a Nicalon outer layer and sealed Nextel inner layer separated by Q-felt insulation. It is supported by graphite polyimide ribs that are hinged to permit stowage for ascent in the Aft Cargo Carrier. The structural airborne support equipment (ASE) considerations were shown in Figure 4.4-4. To stow the OTV in the orbiter cargo bay, the two LH₂ tanks are removed and stowed - one forward and one aft of the OTV.
Figure 6.1.1.2.2-1 Ground Based Cryogenic Propulsion Schematic
The LH₂ tanks are supported in the orbiter bay at three points. A fitting is attached to the LH₂ tank that interfaces with the orbiter bay keel fitting as shown in Figure 4.4-4. Two trunnion fittings pick up the LH₂ tank support and interface with the orbiter longeron fittings. The LO₂ tanks remain installed in the OTV airframe. It is designed to interface with the orbiter keel fittings at the lower axle end of the LO₂ tanks. The airframe is attached to the orbiter longerons as shown in Figure 4.4-4, section A-A. The forward hydrogen tank supports are folded back and braced to the central truss by stowage members that comprise a portion of the ASE, stowage members are added to brace the OTV LH₂ truss for in flight loads. All structural ASE will be aluminum and will be stowed in the orbiter payload bay and attached to the OTV by EVA at retrieval rendezvous.

Avionics (Ground-Based Cyro) - The cryogenic ground-based, ACC delivered, OTV avionics, Figure 6.1.1.2.4-1, is a modular design that supports technology insertion as well as redundancy enhancement. A significant feature is its distributed computer architecture with a flexible executive operating system that facilitates performance enhancement and permits affordable software development. The design is single fault tolerant through internal component redundancy for mission success and two fault tolerant for critical operations in the vicinity of the Orbiter. An avionics component list and physical description is presented in Table 6.1.1.2.4-1.

Guidance, Navigation and Control (GN&C) - The GN&C hardware consists of the following:

a. Strapdown Inertial Measurement Unit (IMU)
b. Solid State Star Scanner
c. GPS Receiver/Processor and Hi and Low-Altitude Antennas
d. Majority Vote Flight Controller
A detailed description of these elements is presented in Reference 7.

6.1.1.2.4.2 Data Management - The OTV data management subsystem is configured in a distributed architecture that includes two Executive Computers (dual-CPU type) as shown in Figure 6.1.1.2.4-1, each with large shareable mass memories and local memories. Key functional areas under Executive Computer software control are the Executive Operating System, attitude, guidance and navigation management, sequence control, power management, and test and checkout. The Executive and all of the other intelligent avionics subsystems are interconnected via a global network bus. This global network can support a throughput of from 10 to 20 Mbps via fiber optic cable. The network structure permits each subsystem to access the bus using an intelligent, standard protocol interface.

6.1.1.2.4.3 Telemetry and Command (T&C) - The telemetry and command subsystem is designed around a basic SCI Data Acquisition and Control System (DACS) having a single control and I/O interface unit. The central unit consists of an 80C86 CMOS microprocessor-based system with local RAM (32K) and ROM (8K) for conducting telemetry and command processing independent of the executive computer. Command decoding and authentication, time tagging and command override services are provided.

Figure 6.1.1.2.4-1 Block Diagram of the ground-based, ACC delivered, cryogenic configuration
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<th>Size (in)</th>
<th>Total Qty Wt (lb)</th>
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Table 6.1.1.2.4-1 OTV Avionics Equipment List - Ground-Based ACC, Cryogenic Configuration (Sheet 2 of 2)

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<td><strong>446</strong></td>
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</table>
6.1.1.2.4.4 Communication and Tracking (C&T) - The C&T subsystem provides both direct and relay communication with the ground. Communication with the Orbiter is either direct or through a ground station. The C&T subsystem operates at S-band and is compatible with STDN/TDRSS and SGLS depending upon the specific mission. Provisions have been incorporated for redundant transponders, RF power amplifiers and COMSEC equipment. Two electronically switched steerable array antennas provide hemispheric coverage. Each antenna includes a redundant microprocessor and redundant switching power divider. The other major components are inherently redundant, i.e., 145 passive elements with associated power drivers. Each antenna also includes an integrated preamplifier to facilitate parallel operation of two receivers (for fault tolerant reception) with minimal RF distribution losses. The direct/relay feature provides maximum flexibility from low earth orbit to GEO in terms of coverage and link margins for the various OTV missions. Relay C&T via TDRSS provides the primary tracking and communications for OTV operations below 10,000Km altitude. Direct C&T is the primary mode for higher OTV altitudes, with TDRSS as a backup where coverage is available. The heart of the C&T subsystem is a dual mode TDRSS/STDN transponder and 20 watt RF amplifier (such as the existing Motorola packages) combined with the Ball Aerospace ESSA. This combination provides the flexibility in spatial coverage and the necessary link margins for the various OTV missions.

6.1.1.2.4.5 Electrical Power Subsystem (EPS) - The OTV Electrical Power Subsystem, Figure 6.1.1.2.4-2, consists of redundant fuel cells, vehicle cabling, power distribution and control, reactants, plumbing, and radiators. Power is distributed through redundant buses to the OTV subsystems. The Power Control and Distribution Assembly (PCDA) contains motor driven switches and relays needed to provide load control and fault protection circuitry. The PCDA also interfaces the command and data systems where commands are received from the OTV data bus, and health and status are passed to the data management subsystem. Each of the OTV fuel cells is sized to deliver 1.2 KW peak which includes 20% design margin. A high current density design for the fuel cells was selected to minimize weight and volume for short and medium duration missions. The fuel cells are also sized to provide coarse bus voltage regulation (28 ± 4 VDC) during worst case operation at the end of a five year life. This eliminates the requirement for active power conditioning. An active coolant loop and radiator system are used to reject fuel cell waste heat. One 35 square foot radiator is sized to reject the fuel cell waste heat. Reactants are taken from the main propellant system. Redundant fuel cells and plumbing allow the EPS to meet system reliability requirements without battery backup. There is no safety issue associated with this type of a fuel cell application because it is an extension of the STS design. System power up is also simplified because fuel cell initialization consists of warming the catalysts to operating temperature and supplying reactants.
6.1.1.2.5 Thermal Control (Ground-Based Cyro) – This configuration utilizes a fuel cell power system. The fuel cell thermal control system (TCS) is sized for an OTV continuous flight power requirement of 1.2 KW and a nominal 76-hour OTV flight duration. The fuel cell TCS requires 35 sq-ft² radiator area to dissipate the fuel cell waste heat effectively. The fuel cell radiator weight is 35 lb (Table 6.1.1.2.4-1). The radiators face outboard (maximum view to space) and are mounted to the oxygen tanks with low conductivity mounts which are blanketed from a sun flux environment. A one layer minimum thermal blanket is fixed to the back of the radiator facing the oxygen tank; the cryo side of this blanket has a low emissivity.

The avionics packages are located on the structural trusses between the cryo tanks. The avionics are passively cooled and mounted on pallets for effective heat sinking energy distribution. Small thermostatically controlled heaters and thermal blankets may be required for certain components to supplement the passive thermal control system. Component surface finishes (i.e., painted or polished) and mounting techniques shall be specified at a later time in the OTV design development.

The payload/OTV interface must be made nearly adiabatic. To accomplish this, 25 to 50 layers of insulation blanket (double aluminized Kapton MLI), is located at the interface.
The cryo tanks require a significant amount of insulation to prevent excessive boiloff. 0.25 inch of SOFI with 0.5 inch of MLI (25 layers of double aluminized Kapton) are used for the H2 tanks. The layer of SOFI is used to eliminate the need for an MLI purge system. The temperature of the outer side of the SOFI is warm enough not to freeze any component of the dry nitrogen atmosphere provided in the Aft Cargo Carrier. The SOFI is not required for the oxygen tanks. The main propellant feedline insulation is 2 layers of gold foil.

The OTV reaction control system (RCS) requires thermal protection for the RCS tank, feedlines, and propulsion modules. The RCS tank has an MLI blanket (10 to 25 layers) and strip heaters. The feedlines contain hydrazine (freezing point of 35°F) and requires low power (approximately 25 watt) strip heaters and one or two layers of thermally insulating blankets. The RCS modules will be maintained at sufficiently high temperatures by "thermal pulsing" techniques (i.e., periodic module firings).

The helium tank used in the pneumatic system requires heater tape for adequate thermal control to maintain proper pressurization.

Engine nozzle heating effects are not considered a problem for this configuration.
6.1.1.3 System Weight Summary - Ground-Based Cryo

Total flight vehicle weight for the ground-based cryogenic configuration is presented in Table 6.1.1.3-1. Dry weight, nonpropulsive fluids and usable propellant are summarized. Dry weight is categorized according to the groupings requested by MSFC, and the individual items include a 15% contingency (assuming that all equipment can be considered to be new in this time frame). Table 6.1.1.3-2 shows a detailed dry weight breakdown within each group, including the contingency weight assigned.

Table 6.1.1.3-1 Stage Weight Summary - Ground-Based Cryo 45K Propellant Load

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<td>4. Propulsion Less Engine</td>
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<tr>
<td>5. Main Engine</td>
<td>313</td>
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<tr>
<td>6. Reaction Control System</td>
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<tr>
<td>7. Guidance, Navigation, Control</td>
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<td>8. Communications &amp; Data Handling</td>
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<td>9. Electrical Power</td>
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<td>11. Aerobrake</td>
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</table>

Dry Weight Total 4916

12. Fluids

Reactants, Coolants & Residuals
Residual - FU (LH₂) 96
Residual - OX (LO₂) 579
FC Coolant 10
Pressurants - He & GN₂ 24
Hydrazine - ACS 400

Inert Weight Total 6025

Usable Main Propellants
Fu-LH₂ (Incl. FPR) 6332
OX-LO₂ (Incl. FPR) 37993

Ignition Weight Total 50350

Mass Fraction

\[
\frac{44325 \text{ (Main Prop Incl FPR)}}{50350 \text{ (Ignition Weight)}} = 0.88
\]
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Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity
Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo
45K Propellant Capacity (Continued)

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Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity (Continued)

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<tr>
<td></td>
<td>TPS Quilt (Flex)</td>
<td>568</td>
</tr>
<tr>
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<td>Contingency</td>
<td>112</td>
</tr>
<tr>
<td>11.2</td>
<td>Doors &amp; Mechanism</td>
<td></td>
</tr>
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<td>Doors</td>
<td>101</td>
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<td>15</td>
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<tr>
<td>11.3</td>
<td>Support Structure</td>
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<td></td>
<td>Ribs and Struts</td>
<td>302</td>
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<td></td>
<td>Contingency</td>
<td>45</td>
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<td></td>
<td><strong>Group 11 Total</strong></td>
<td><strong>1320</strong></td>
</tr>
<tr>
<td>15</td>
<td>Propellants</td>
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<td>15.1</td>
<td>Main</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Usable - LH2 (incl. FPR)</td>
<td>6332</td>
</tr>
<tr>
<td></td>
<td>Usable - LO2 (incl FPR)</td>
<td>37993</td>
</tr>
<tr>
<td></td>
<td>Residual - LH2</td>
<td>96</td>
</tr>
<tr>
<td></td>
<td>Residual - LO2</td>
<td>579</td>
</tr>
<tr>
<td></td>
<td>Press. Pneum. (He)</td>
<td>10</td>
</tr>
<tr>
<td>15.2</td>
<td>F.C. Coolant &amp; Reactants</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Coolant</td>
<td>10</td>
</tr>
<tr>
<td>15.3</td>
<td>ACS</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Hydrazine</td>
<td>400</td>
</tr>
<tr>
<td></td>
<td>Pressurant - GH2</td>
<td>14</td>
</tr>
<tr>
<td></td>
<td><strong>Group 15 Total</strong></td>
<td><strong>45434</strong></td>
</tr>
</tbody>
</table>
6.1.1.4 Performance on Model Missions: Ground-based Cryo - The following is a summary of the ground rules and assumptions used in the performance analyses contained herein. This description applies not only to the data presented in section 6.1.1.4 but to the analyses in section 6.2.1.4 as well.

The delta v's used for ground-based geosynchronous delivery missions were as shown in Table 6.1.1.4-1:

Table 6.1.1.4-1 Ground-Based ACC OTV GEO Delivery Delta Vs

<table>
<thead>
<tr>
<th>BURN</th>
<th>PURPOSE</th>
<th>PLANE CHANGE (DEG)</th>
<th>PROPULSIVE DELTA-V (FPS)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Shuttle MECO to 86.4 x 140.0 nmi</td>
<td>0.00</td>
<td>248.7</td>
</tr>
<tr>
<td>2</td>
<td>86.4 x 140 nmi to 140 x 140 nmi</td>
<td>0.00</td>
<td>96.0</td>
</tr>
<tr>
<td>3</td>
<td>140 x 140 nmi to 140 x 19322.9 nmi</td>
<td>2.19</td>
<td>8073.8</td>
</tr>
<tr>
<td>4</td>
<td>140 x 19322.9 to 19322.9 nmi circ</td>
<td>26.31</td>
<td>5855.8</td>
</tr>
<tr>
<td>5</td>
<td>19322.9 circ to 45 x 19322.9 nmi</td>
<td>28.50</td>
<td>6059.7</td>
</tr>
<tr>
<td>6</td>
<td>Aeropass maneuver to 2.0 x 149 nmi</td>
<td>0.00</td>
<td>0.0</td>
</tr>
<tr>
<td>7</td>
<td>2.0 x 140 to 140 nmi circ</td>
<td>0.00</td>
<td>535.0</td>
</tr>
</tbody>
</table>

For ground-based cargo bay OTV missions, the GEO mission delta v's are the same as above except that the first two burns are omitted.

The above ideal, impulsive delta-v's. Gravity induced velocity losses were added to the initial perigee burns as a function of the burn time involved. boiloff was accounted for at the rate of 2.8 lbs/hr.

The delta v's used for planetary missions were derived from a hypothetical launch geometry which minimizes the OTV delta-v penalty incurred due to precessing of the Shuttle orbit while the OTV is away. No attempt was made to research actual launch window geometries and there was assumed to be no plane change required to get from the Shuttle orbit to the departure hyperbola at launch time. Since each planetary mission has a unique delta-v budget, we have not listed the planetary delta-v's in tabular form. More information on planetary mission analysis methodology is contained in Reference 8.

For some OTV configurations on some planetary missions it was necessary to add an expendable kick stage (EKS) to the payload. For such cases, the specific orbital energy (or C3) at which the OTV shuts down and the kick stage takes over was chosen so as to minimize the gross weight of the OTV + EKS + payload. In all cases where an EKS was used, they were sized by assuming a mass fraction of 0.95 and an Isp of 310 seconds.

Table 6.1.1.4-2 summarizes the propellant load required to accomplish each of the model missions that are to be performed by the ground-based cryogenic Orbital Transfer Vehicle. Figure 6.1.1.4-1 presents a parametric summary of the performance capability of this vehicle.
Table 6.1.1.4-2 Performance Analysis for Required Missions
Ground-Based Cryogenic ACC, 45K OTV

Isp = 475 Sec

REV. 8

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP (lb)</th>
<th>P/L DN (lb)</th>
<th>OTV PROPELLANT (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>GEOSYNCHRONOUS MISSIONS</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>13006</td>
<td>12017</td>
<td>0</td>
<td>37485</td>
</tr>
<tr>
<td>18912</td>
<td>12000</td>
<td>2000</td>
<td>40488</td>
</tr>
<tr>
<td>19031</td>
<td>12000</td>
<td>0</td>
<td>37428</td>
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<tr>
<td>(Reflight)</td>
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<td></td>
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</tr>
<tr>
<td>19031</td>
<td>12000</td>
<td>0</td>
<td>37428</td>
</tr>
<tr>
<td>PLANETARY MISSIONS</td>
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<td></td>
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<tr>
<td>17075</td>
<td>5000</td>
<td>0</td>
<td>29320</td>
</tr>
<tr>
<td>17081</td>
<td>4079</td>
<td>0</td>
<td>14437</td>
</tr>
<tr>
<td>17084</td>
<td>4410</td>
<td>0</td>
<td>37287</td>
</tr>
</tbody>
</table>

EKS MASS (lb)

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>19031</td>
<td>8268</td>
</tr>
<tr>
<td>17081</td>
<td>4636</td>
</tr>
<tr>
<td>17084</td>
<td>0</td>
</tr>
</tbody>
</table>

Figure 6.1.1.4-1 Ground Based 45Klb Cryo OTV Performance Capability
6.1.2 SPACE-BASED CRYO FAMILY

6.1.2.1 General Arrangement (Space-Based Cryo) - The space-based cryogenic family of OTV uses two basic stage designs in three configurations. The first configuration provides an initial space-based capability to perform GEO and planetary delivery missions and unmanned GEO servicing missions. This single stage concept is illustrated in Figure 6.1.2.1-1. This vehicle is derived from our ground based concept, but there are several important differences. Propellant capacity has been increased to 55,000 lb. to enable a 20,000 lb GEO payload delivery capability. Minimum tank gauges have been reduced to .010 in. on the LO₂ tank and .012 on the LH₂ tank, reflecting lower tank pressure requirements. Meteoroid shielding has been added to the tanks. The general arrangement has been opened up to permit servicing at the space station, when necessary, by a space suited astronaut. Redundancy, including two main engines, has been added to increase mission reliability. Avionics units have been mounted on an avionics ring at the forward end of the vehicle to simplify space-based maintenance. The aeroshield is designed to withstand a peak pressure of 35 psf, enabling retrieval of the unmanned servicing spacecraft. A more detailed layout of this stage is shown in Figure 6.1.2.1-4.

Figure 6.1.2.1-1 Initial Space-Based Cryo OTV
The general arrangement of our selected growth space-based OTV is shown in Figure 6.1.2.1-2. This configuration is not required to support the Rev. 8 low mission model. It is required to support the Manned Lunar Sortie mission in the Rev. 8 nominal model, and is capable of supporting the Manned GEO Sortie mission in the older Rev. 7 mission model. In most respects, this vehicle is identical to the initial space-based OTV. The basic structure is identical. The level of subsystem redundancy is the same. Since the electrical power subsystem and reaction control subsystem are fed from the main propellant tanks, no subsystem changes are required to accommodate different mission durations. Design variation does result from changes in propellant load and heating environment resulting from the delivery and retrieval of heavier spacecraft. Tank size is increased to accommodate an 81,000 lb propellant load. This is large enough to perform the largest lunar missions in a two stage configuration without excessive compromise in meeting the manned GEO sortie mission requirement. Since this vehicle is readily capable of performing the Rev. 7 manned GEO sortie (a 14000 retrieval payload), the aerobrake was sized to be compatible with this capability. In this case, the peak design pressure of the aerobrake is increased to 63 psf. This results in an increase in TPS thickness and aerobrake structural strength. A more detailed layout of this stage is shown in Figure 6.1.2.1-5.
The ultimate spacebased OTV capability requirement, encountered in the nominal Rev 8 mission model, is to perform the Manned Lunar Sortie mission. This requires delivery of 80,000 lb to and return of 15,000 lb from low lunar orbit on a single OTV mission. The 81,000 lb propellant capacity stage was sized to accomplish this mission with the two stage configuration shown in Figure 6.1.2.1-3. No design changes are required to implement this configuration, other than development of an appropriate interstage structure. It is anticipated that the increased severity of the lunar reentry conditions can be accommodated with the same aeroshield design by using a two pass aeromanuever.

Figure 6.1.2.1-3  Cryogenic Lunar Logistics Vehicle
6.1.2.2 Subsystem Summary Description (Space-Based Cryo)

6.1.2.2.1 Aeroassist (Space-Based Cryo) - The overall layout of the space based cryo configurations were shown in Figures 6.1.2.1-1 through -5. The aeroassist devices used with these configurations are similar in concept to the one used on the ground-based configuration discussed in Paragraph 6.1.1.2.1. The diameter is increased to 44 feet to protect the larger OTV stage and payloads to be retrieved. The total weights to be retrieved are heavier, so design surface pressure and heating is greater, resulting in thicker surface insulation. Figure 6.1.2.2.1-1 summarizes the aeroshield parameters.

<table>
<thead>
<tr>
<th>CONFIGURATION</th>
<th>W/C'D'A</th>
<th>BRAKE DIAMETER (FT)</th>
<th>TPS</th>
<th>q_MAX (BTU/FT²/SEC)</th>
<th>q_MAX (BTU/FT²/°F)</th>
<th>T_MAX (°F)</th>
<th>TPS THICKNESS (IN.)</th>
<th>W/DRAKE</th>
<th>W/RETURN</th>
<th>DESIGN LOAD (PSF)</th>
</tr>
</thead>
<tbody>
<tr>
<td>MANNED CAPSULE</td>
<td>6.0</td>
<td>44</td>
<td>FSI</td>
<td>21.4</td>
<td>3050</td>
<td>2390</td>
<td>0.38</td>
<td>0.10</td>
<td>34</td>
<td>26</td>
</tr>
<tr>
<td>GROWTH CAPSULE</td>
<td>9.9</td>
<td>44</td>
<td>FSI</td>
<td>25.6</td>
<td>3680</td>
<td>2600</td>
<td>0.43</td>
<td>0.06</td>
<td>63</td>
<td>48</td>
</tr>
</tbody>
</table>

* 14,000 lbs, 14½' W x 23' L (A growth capability)

Figure 6.1.2.2.1-1 Space-Based Cryo Aeroshield
The initial capability aeroshield shown in the figure is mounted on the 55,000 lb propellant capacity stage which is designed to return the unmanned servicing spacecraft from GEO to low orbit. The growth capability aeroshield is placed on the 81,000 lb OTV, and is designed to return the 14,000 lb manned capsule to LEO. Both of the configurations have sufficiently high ballistic coefficients, and associated reentry heating, that it is necessary to use the rigid surface insulation tiles on the central portion of the aeroshield. Likewise, the use of flexible surface insulation for the engine door is not feasible. It should be noted that the 7500 lb manned capsule of the Rev. 8 mission model can be returned with the initial capability aeroshield. The growth capability aeroshield would not have to be introduced until the manned lunar sortie is encountered.

These configurations employ two main engines to achieve fail-safe-return man rating. As a consequence, the engine doors are larger than those used on the single engined ground-based configuration as well as being constructed using rigid surface insulation. The resulting rigid doors are opened by lifting them from their openings and rotating them as shown in Figure 4.2.2.1-1. The weights of these aeroshields are summarized in paragraph 4.2.3 along with the remainder of vehicle subsystem weights.

6.1.2.2.2 Propulsion (Space-Based Cryo) - The propulsion characteristics of the space-based cryogenic stages are shown in Table 6.1.2.2.2-1 and the reaction control characteristics are shown in Table 6.1.2.2.2-2. The MPS uses two 7500 lb advanced expander cycle engines. Studies show that the development of an engine for OTV is cost effective. Technology development continues with three main engine contractors through funding from NASA/LeRC. Advanced expander cycle concepts all use higher chamber pressure and expansion ratios to obtain performance levels from 475 to 487 seconds, depending on engine contractor performance predictions and the level of technology incorporated in the expander cycle (ranges from hydrogen expander to dual propellant expander). Rocketdyne and Pratt Whitney point designs are in the 7500 and 15,000 lb thrust class, whereas Aerojet is working in the 3,000 lb thrust class. Throttle ratios possible are 10:1 for the expander cycle engines with an ultimate goal of up to 30:1, but with the current Rev. 8 mission model we have selected 50% step throttling as cost effective. All engines have THI and PHI capabilities and autogenous pressurization capability. Optional valve/actuator control is provided by high pressure GH2 and GO2. The cycle life varies between 300 starts and 10 hours of life up to 500 starts and 20 hours of life as a design goal. Increasing life beyond the ground-based 5 hrs must be based on mission model and cost.

Lower tank pressures are used on the space-based vehicle because the propellants will be maintained saturated at 1 atm and the engine interface is below the tank outlet. Nominal operating pressures are 18 - 19 psia and 17 - 18 psia in the LO2 and LH2 tank, respectively.
Table 6.1.2.2-1 Space-Based Cryogenic MPS Summary

<table>
<thead>
<tr>
<th>Component</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>ENGINE</td>
<td>TWO ENGINES, 7500 LB THRUST, EXPANDER CYCLE Isp - 475 SEC</td>
</tr>
<tr>
<td>PROPELLANT DISTRIBUTION</td>
<td>DUAL TANK, PARALLEL FEED, TOTAL ACQUISITION DEVICE</td>
</tr>
<tr>
<td>PRESSURIZATION</td>
<td>AUTOGENEOUS FROM MPS ENGINE FOR PUMPED IDLE AND FULL THRUST--NOT REQUIRED FOR TANK HEAD IDLE</td>
</tr>
<tr>
<td>VENT</td>
<td>TVS HEAT EXCHANGERS AND NON-PROPULSIVE VENTS FOR BOTH PROPELLANTS FOR FLIGHT OPS</td>
</tr>
<tr>
<td>VALVE ACTUATION</td>
<td>HIGH PRESSURE GASEOUS HYDROGEN AND OXYGEN</td>
</tr>
<tr>
<td>PROPELLANT UTILIZATION</td>
<td>TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL</td>
</tr>
<tr>
<td>THERMAL CONTROL</td>
<td>H&lt;sub&gt;2&lt;/sub&gt; 1&quot; MLI (50 LAYERS)</td>
</tr>
<tr>
<td></td>
<td>O&lt;sub&gt;2&lt;/sub&gt; 1&quot; MLI (50 LAYERS)</td>
</tr>
<tr>
<td>PROXIMITY OPERATION</td>
<td>TWO FAULT TOLERANT ISOLATION</td>
</tr>
<tr>
<td>REDUNDANCY</td>
<td>FAIL SAFE</td>
</tr>
<tr>
<td>MAINTENANCE</td>
<td>COMPONENT MODULARITY</td>
</tr>
<tr>
<td></td>
<td>ENGINE REPLACED AS UNIT - 5 HR LIFE</td>
</tr>
</tbody>
</table>
Table 6.1.2.2-2  Space-Based Cryogenic RCS Summary

- PROPELLANT
  - GO₂/CH₂

- ROCKET ENGINE MODULE
  - NEW DESIGN BASED ON TECHNOLOGY STUDIES, 100 LBF,
  - 14 THRUSTERS (7 THRUSTERS PER MODULE)
  - THRUSTER ISₚ = 400 SEC, ISₚₚ = 378 SEC WITH CONDITIONER LOSSES
  - 3 DOF CONTROL AND +X TRANSLATION
  - FAIL OPERATIONAL

- PROPELLANT SUPPLY
  - COMMON STORAGE WITH MPS TANKS, CONDITIONED BY GAS GEN/HEAT EXCHANGER
    - ASSY TO 1,000 psi
  - REGULATED TO 300 psi FOR THRUSTERS AND PNEUMATICS.
  - 320 LBₚₚ of GO₂/CH₂ AT A MR 4:1

- SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS
Propellants are stored and delivered to the engines using the same general tankage arrangements and components defined previously for the ground-based vehicle. However, for the space-based vehicle, system components and tankage will be modularized for ease of replacement on orbit. For instance, the hydrogen tank has a connector on the side that contains the propellant, pressurization, vent, thermodynamic vent, electrical power and control, and instrumentation interfaces. All connections are made simultaneous within one connector to assure ease of tank replacement. The MPS schematic is shown in Figure 6.1.2.2.2-1. A total acquisition system is included to provide for onorbit detanking as a contingency for an aborted mission.

Figure 6.1.2.2.2-1 Space-Based LH2/LO2 Schematic
The RCS uses gaseous oxygen and hydrogen thrusters to provide thrust for attitude and translational maneuvers. The propellants are stored in the main propulsion tanks and accumulators are charged from the GGS. The accumulator pressure can vary from 1000 to 300 psi between charges and the pressure is regulated to 300 psi for delivery to the GO2/GH2 thrusters. The system will store 12,000 lb-sec of total impulse between recharges. The thruster technology will be new, however, it is based upon LeRC's GO2/GH2 thruster development which occurred in the early 1970 time period and demonstrated Isp in the 400 second range. This technology is also now being studied by ALRC and LeRC for the Space Station. To provide gas for recharging the accumulators between main engine burns a gas generator, pump and heat exchanger are provided to condition the propellants. This conditioner is designed to run off optimum performance at a mixture ratio of 1.0. This reduces gas generator temperatures to about 1500°F and allows flexibility in conditioning the system for start and a simplified control system. This lower efficiency operation degrades the system Isp from 400 to 378 seconds. The RCS schematic is shown in Figure 6.1.2.2.2-2.

Figure 6.1.2.2.2-2 Space-Based GH2/GO2 RCS Schematic
6.1.2.2.3 Structures and Packaging (Space-Based Cryo) - The space-based cryo configurations shown in Figures 6.1.2.1-4 and -5 use two spherical LH$_2$ tanks, two spherical LO$_2$ tanks and two 7500 lb thrust engines. A central truss provides the backbone of the stage with engine and aerobrake mounted on one end of the truss and the avionics ring and payload adapter attached on the other. The two space-based cryo stage sizes are implemented by simply changing tanks. The smaller 55,000 lb capacity tanks will use spacers to fill the gaps between them and the structure that was designed to accommodate up to the 81,000 lb tank size.

The OTV is delivered to space in the orbiter cargo bay (Figures 6.1.2.2.3-1 and -2). For orbiter cargo bay delivery, tanks are removed from the central core and aerobrake is removed. The aerobrake is unfolded in space. Grappling fixtures have been provided to allow use of an RMS to handle the tanks and truss.
The aerobrake is mounted in the payload bay on an aerobrake deployment assist mechanism (ADAM). The ADAM (Figure 6.1.2.2.3-3) consists of a central shaft with 4 hinged, telescoping arms and slotted guide plates. It is designed to be either returned to earth after deploying the aerobrake or stored in the space hangar with a collapsed aerobrake. The aerobrake is deployed by extending the telescoping arms up and out, which pulls the spring assisted struts through the guide plates and allows the hinged rib to unfold. After being fully extended, the aerobrake is removed from the ADAM, and mounted to the OTV support structure.

The aerobrake structure consists of lower support ring, upper interface ring, hinged ribs and spring assisted, collapsible struts. The center core is composed of graphite polyimide honeycomb with ceramic foam tiles and a quilted outer edge of nicalon, ceramic felt and sealed Nextel.

The release mechanism, (Figure 6.1.2.2.3-4) consisting of a release handle, 12 latch pins and a connecting kevlar rope, is mounted to the OTV support ring and facilitates the attachment and removal of the aerobrake. The release mechanism, when engaged, retracts the 12 latch pins simultaneously via kevlar rope and frees the aerobrake from the OTV. Figure 6.1.2.2.3-5 shows the aerobrake rib deflection, which is considered to be acceptable.
HINGED TELESCOPING ARM

DEPLOYED

FLEX TPS BLANKET

SLOTTED GUIDE PLATES (ASE)

FOLDED

TELESCOPING CHANNEL

15.0'

6.5'

RIGID TILES

18.0'

22.0'

107.

Figure 6.1.2.2.3-3 Space-Based Aerobrake

AERObRAKE INTERFACE RING (13" 6' DIA)

6" STROKE RELEASES ALL 12 LATCHES

END EFFECTOR

'KEVLAR' ROPE (CONTINUED BETWEEN LATCHES)

LATCH (12 PLACES)

AEROBRACE INTERFACE RING

Figure 6.1.2.2.3-4 Aerobrake Release Mechanism

6-38
The cryogenic tanks are of fusion welded construction and are made in two halves from 2090 aluminum lithium alloy. LH2 tank membrane is a minimum of .012 thick. If problems are uncovered during testing of the 2090 alloy or in developing forming in two halves, the back up alloy would be 2219 aluminum with back up processing to be four gores per head with machined concical caps. If difficulties are encountered in handling .012 thick tanks, membrane thickness would be increased as required. The basic airframe truss is graphite epoxy.

The vehicles are equipped with crane and cradle provisions for handling at the Space Station. In addition major components such as aerobrakes and tanks have grapple provisions for component changeout at the Space Station.

6.1.2.2.4 Avionics (Space Based Cryo) - Avionics for the space-based, cryogenic configuration of the OTV is modular in design and similar to the ground based configuration. The space-based configuration has a distributed computer architecture with a flexible executive operating system that facilitates performance enhancement and permits affordable software development. Because of longer mission times, this design has greater fault tolerance features. It retains the two fault tolerant feature for critical operations in the vicinity of the Orbiter. System block diagram is shown in Figure 6.1.2.2.4-1, equipment list in Table 6.1.2.2.4-1.
6.1.2.2.4.1 Guidance, Navigation and Control (GN&C) - The GN&C hardware consists of the following:

a. Dual Redundant Ring Laser Gyro (RLG) Inertial Measurement Unit(s) (IMU)
b. Dual Star Trackers
c. GPS Receiver/Processor & Hi and Low-altitude Antennas
d. Dual Majority Vote Flight Controllers

Two RLG IMUs were selected for the space-based configuration rather than the Teledyne DRIRU II unit because of the longer mission and performance capability of the cryogenic OTV. Each IMU includes three (3) ring laser gyros (RLGs) and three (3) pendulous mass accelerometers and required computers and power supplies. A star tracker was selected instead of a scanner to take advantage of increased sensitivity of trackers and to minimize required maneuvers. Details of the selected GN&C hardware are presented in Reference 7.

6.1.2.2.4.2 Data Management (DM) - The OTV Data Management subsystem is the same as in Section 6.1.1.2.4.2.

6.1.2.2.4.3 Telemetry and Command (T&C) - The T&C subsystem is the same as in Section 6.1.1.2.4.3.

6.1.2.2.4.4 Communication and Tracking (C&T) - The C&T subsystem is the same as in Section 6.1.1.2.4.4.
<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Equipment</th>
<th>Weight (lb)</th>
<th>Power (W)</th>
<th>Size (in)</th>
<th>Total Qty Wt (lb)</th>
<th>Power (W)</th>
<th>Max</th>
<th>Avg</th>
</tr>
</thead>
<tbody>
<tr>
<td>GN&amp;C</td>
<td>Star Scanner</td>
<td>11</td>
<td>10</td>
<td>7x7x20</td>
<td>2</td>
<td>22</td>
<td>20</td>
<td>10</td>
</tr>
<tr>
<td></td>
<td>IMU</td>
<td>24</td>
<td>40</td>
<td>8x8x12</td>
<td>2</td>
<td>48</td>
<td>80</td>
<td>80</td>
</tr>
<tr>
<td></td>
<td>GPS Receiver</td>
<td>20</td>
<td>30</td>
<td>8x8x9</td>
<td>1</td>
<td>20</td>
<td>30</td>
<td>10</td>
</tr>
<tr>
<td></td>
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### Table 6.1.2.2.4-1 OTV Avionics Equipment List - Space-Based Configuration (Sheet 2 of 2)

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<th>Equipment</th>
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<th>Power (w)</th>
<th>Size (in)</th>
<th>Qty</th>
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<th>Power (w)</th>
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6.1.2.2.4.5 **Electrical Power Subsystem (EPS)** - The Electrical Power subsystem is essentially the same as in Section 6.1.1.2.4.5 sized for a 1.7 Kw peak output which includes a 20% design margin for each fuel cell. The configuration is shown in Figure 6.1.2.2.4-2. Two 25 sq ft radiators reject fuel cell waste heat. Since the EPS reactant is supplied from the main propellant tanks, the OTV has an inherent ability to support longer duration missions without requiring design changes. End-of-mission fuel cell reactant tankage is not required as propellant tank purge is not required while the fuel cells are operating during space-based operations.
6.1.2.2.5 Thermal Control (Space-Based Cryo) - These vehicles have essentially the identical thermal control system (TCS) as the ground-based cryo configuration except for mission duration. The avionics are mounted circumferentially and outboard on the avionics ring located at the payload/OTV interface. The outboard side of the ring is painted with a low alpha over epsilon paint. The avionics are housed in MMS-type boxes. The avionics components are mounted to the skirt in a manner which allows component waste heat to travel freely to the skirt. The location of the avionics on the ring will allow for the component waste heat to be evenly distributed among all the avionics. This reduces supplemental heater power requirements.

The fuel cell TCS is sized for a nominal 25-day OTV flight duration which requires two 25-ft² radiators to dissipate fuel cell waste heat. The radiators are located on the avionics ring simplifying the cooling loop system and reducing its weight. The two radiators are mounted on opposite sides of the vehicle to accommodate long duration fixed OTV orientation with respect to the sun vector, thus preventing fuel cell overheating.

All H₂ and O₂ cryo tanks are insulated with 1.0 inch (50 layers) of LI. The main propellant feedline insulation consists of 2 layers of gold foil.

Meteoroid protection is provided for propellant tankage with a stand-off thin wall aluminum bumper outside the multi-layer insulation. The MLI functions as a catcher for meteoroid impact particles as well as an insulation.
6.1.2.3 System Weight Summary - Space-Based Cryo

The flight vehicle weight for the initial space-based cryogenic configuration (55K lb propellant capacity) is presented in Table 6.1.2.3-1. Dry weight, non propulsive fluids and usable propellants are summarized. Dry weight is categorized according to the groupings requested by MSFC, and the individual items include a 15% contingency. Table 6.1.2.3-2 shows a detailed dry weight breakdown within each group, including the contingency weight assigned.

Table 6.1.2.3-3 and 6.1.2.3-4 show the equivalent information for the growth space-based cryogenic configuration (81K lb propellant capacity).

Table 6.1.2.3-1 Stage Weight Summary - Space-Based Cryo 55K Propellant Load

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<td>4. Propulsion Less Engine</td>
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<td>5. Main Engine</td>
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<tr>
<td>6. Reaction Control System</td>
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<td>7. Guidance, Navigation, Control</td>
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<td>8. Communications &amp; Data Handling</td>
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<td>9. Electrical Power</td>
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<td>10. Thermal Control System</td>
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<td>11. Aerobrake</td>
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Dry Weight Total 7364

12. Fluids
Reactants, Coolants & Residuals
Residual - FU (LH2)  118
Residual - OX (LO2)  707
FC Coolant  15

Inert Weight Total 8204

Usable Main Propellants
FU-LH2 (Incl. FPR)  7739
OX-LO2 (Incl. FPR)  46436

Ignition Weight Total 62379

Mass Fraction

\[
\frac{54175 \text{ (Main Prop Incl FPR)}}{62379 \text{ (Ignition Weight)}} = 0.87
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Table 6.1.2.3-2 Detailed Dry Weight Breakdown - Initial Space-Based Cryo 55Klb Propellant Capacity (Continued)

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Dry Weight Total: 7650

| 12. Fluids                               |             |
| Reactants, Coolants & Residuals         |             |
| Residual - FU (LH₂)                     | 174         |
| Residual - OX (LO₂)                     | 1041        |
| FC Coolant                              | 15          |

Inert Weight Total: 8880

Usable Main Propellants
|             |             |
| FU-LH₂ (Incl. FPR)                     | 11397       |
| OX-LO₂ (Incl. FPR)                     | 68388       |

Ignition Weight Total: 88665

Mass Fraction

\[
\frac{79785 \text{ (Main Prop Incl FPR)}}{88665 \text{ (Ignition Weight)}} = 0.90
\]
## Table 6.1.2.3-4: Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity

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Group 2 Total: 1821

Group 3 Total: 835

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Table 6.1.2.3-4  Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity (Continued)

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<td>68388</td>
</tr>
<tr>
<td></td>
<td>Residual - FU LH2</td>
<td>174</td>
</tr>
<tr>
<td></td>
<td>Residual - OX LO2</td>
<td>1041</td>
</tr>
<tr>
<td>15.2</td>
<td>FC Coolant</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Coolant</td>
<td>15</td>
</tr>
<tr>
<td></td>
<td>Group 15 Total</td>
<td>81015</td>
</tr>
</tbody>
</table>
6.1.2.4 Performance on Model Missions: Space Based Cryo

The following is a summary of the groundrules and assumptions used in the performance analyses contained herein. This description applies not only to the data presented in section 6.1.2.4 but to the analyses in section 6.2.2.4 as well.

For space based OTV GEO missions, the delta v's used are shown in Table 6.1.2.4-1.

Table 6.1.2.4-1 Space-Based OTV GEO Delivery Delta V's

<table>
<thead>
<tr>
<th>BURN</th>
<th>PURPOSE (orbit dimensions in nmi)</th>
<th>PLANE CHANGE (DEG)</th>
<th>PROPULSIVE DELTA-V (FPS)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>270 circ. to 270 x 19322.9</td>
<td>2.26</td>
<td>7856.4</td>
</tr>
<tr>
<td>2</td>
<td>270 x 19322.9 to 19322.9 circ.</td>
<td>26.24</td>
<td>5855.8</td>
</tr>
<tr>
<td>3</td>
<td>19322.9 circ. to 45 x 19322.9</td>
<td>28.50</td>
<td>6049.7</td>
</tr>
<tr>
<td>-</td>
<td>Aeropass maneuver to 32.9 x 270</td>
<td>0.00</td>
<td>0.0</td>
</tr>
<tr>
<td>4</td>
<td>32.9 x 270 to 270</td>
<td>0.00</td>
<td>535.0</td>
</tr>
</tbody>
</table>

The above are ideal, impulsive delta-v's. Gravity induced velocity losses were added to the initial perigee burns as a function of the burn time involved. Boiloff was accounted for at the rate of 2.8 lbs/hr.

Lunar mission OTV delta-v's are shown in Table 6.1.2.4-2

Table 6.1.2.4-2 SPACE BASED OTV LUNAR DELIVERY DELTA V's

<table>
<thead>
<tr>
<th>BURN</th>
<th>PURPOSE (orbits dimensions in nmi)</th>
<th>PROPULSIVE DELTA-V (FPS)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>270 circ. to trans-lunar inject</td>
<td>11350.0</td>
</tr>
<tr>
<td>2</td>
<td>outbound midcourse correction</td>
<td>150.0</td>
</tr>
<tr>
<td>3</td>
<td>inject into 70 nmi circ. lunar orbit</td>
<td>2870.0</td>
</tr>
<tr>
<td>4</td>
<td>70 nmi circ. lunar to trans-earth</td>
<td>2870.0</td>
</tr>
<tr>
<td>5</td>
<td>return leg midcourse correction</td>
<td>150.0</td>
</tr>
<tr>
<td>-</td>
<td>aeropass maneuver</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>circularize with space station</td>
<td>535.0</td>
</tr>
</tbody>
</table>

The delta v's used for planetary missions were derived from a hypothetical launch geometry which minimizes the OTV delta-v penalty incurred due to precession of the Space Station orbit while the OTV is away. No attempt was made to research actual launch window geometries and there was assumed to be no plane change required to get from the Space Station orbit to the departure hyperbola at launch time. Since each planetary mission has a unique delta-v budget, we have not listed the planetary delta-v's in tabular form. More information on planetary mission analysis methodology is contained in Reference 8.
For some OTV configurations on some planetary missions it was necessary to add an expendable kick stage (EKS) to the payload. For such cases, the specific orbital energy (or C3) at which the OTV shuts down and the kick stage takes over was chosen to minimize the gross weight of the OTV + EKS + payload. In all cases where an EKS was used, they were sized by assuming a mass fraction of 0.95 and an Isp of 310 seconds.

Table 6.1.2.4-3 presents the propellant loads required to perform each of the low model missions that can be captured by the initial space based cryogenic OTV. Table 6.1.2.4-4 presents the equivalent data for the two stage version of the growth space based OTV. These data were used in the programmatic trade studies discussed in Volume III. Figure 6.1.2.4-1 presents a summary of the performance capability of the initial space based OTV recommended for development as a result of programmatic evaluations of the Rev 8 Low Mission Model.

![Figure 6.1.2.4-1 Space-Based 55Klb OTV Performance Capability](image)

\[ \text{Legend} \]
- 0 Degree Turn
- + 10 Degree Turn
- . 20 Degree Turn
- A 30 Degree Turn
- X 40 Degree Turn
- V 50 Degree Turn

\[ \text{Figure 6.1.2.4-1 Space-Based 55Klb OTV Performance Capability} \]
Table 6.1.2.4-3  Performance analysis for Required Missions  
Cryogenic, Growth Space-Based, 55K OTV

Isp = 475 sec

<table>
<thead>
<tr>
<th>Mission</th>
<th>P/L Up (lb)</th>
<th>P/L Dn (lb)</th>
<th>OTV Propellant (lb)</th>
<th>EKS Mass</th>
</tr>
</thead>
<tbody>
<tr>
<td>Geosynchronous Missions</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>13006</td>
<td>12017</td>
<td>0</td>
<td>40596</td>
<td>50179</td>
</tr>
<tr>
<td>13700</td>
<td>20000</td>
<td>0</td>
<td>53099</td>
<td>30014</td>
</tr>
<tr>
<td>18073</td>
<td>12000</td>
<td>0</td>
<td>40571</td>
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<tr>
<td>18040</td>
<td>20000</td>
<td>0</td>
<td>53099</td>
<td>32228</td>
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<tr>
<td>18722</td>
<td>20000</td>
<td>0</td>
<td>53099</td>
<td>0</td>
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<td>18912</td>
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<td>2000</td>
<td>43749</td>
<td>15188</td>
</tr>
<tr>
<td>10100</td>
<td>20000</td>
<td>0</td>
<td>53099</td>
<td>10458</td>
</tr>
<tr>
<td>13002</td>
<td>7000</td>
<td>4510</td>
<td>36017</td>
<td>0</td>
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<tr>
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<td>7500</td>
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<tr>
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<td>12000</td>
<td>0</td>
<td>40571</td>
<td>0</td>
</tr>
<tr>
<td>19035</td>
<td>20000</td>
<td>0</td>
<td>53099</td>
<td>0</td>
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<tr>
<td>Lunar Missions</td>
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<td>17200</td>
<td>5000</td>
<td>0</td>
<td>29743</td>
<td>15188</td>
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<td>17202</td>
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<td>10458</td>
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<td>Planetary Missions</td>
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<td>10458</td>
<td>0</td>
</tr>
<tr>
<td>17084</td>
<td>4410</td>
<td>0</td>
<td>32228</td>
<td>0</td>
</tr>
</tbody>
</table>
Table 6.1.2.4-4  Performance Analysis for Growth Lunar Missions  
Cryogenic, Two Stage, Space-Based, 81K OTV

Isp  =  475 sec

OTV Propellant (lb)

<table>
<thead>
<tr>
<th>Mission</th>
<th>P/L Up(lb)</th>
<th>P/L Dn(lb)</th>
<th>Stage 1</th>
<th>Stage 2</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lunar Missions</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>17203</td>
<td>80000</td>
<td>15000</td>
<td>81915</td>
<td>81915</td>
<td>163830</td>
</tr>
<tr>
<td>17204</td>
<td>80000</td>
<td>0</td>
<td>77325</td>
<td>77325</td>
<td>154650</td>
</tr>
<tr>
<td>17205</td>
<td>80000</td>
<td>10000</td>
<td>80884</td>
<td>80884</td>
<td>161768</td>
</tr>
</tbody>
</table>

6.2.1 GROUND BASED STORABLE

6.2.1.1 General Arrangement (Ground Based Storable) - Two ground based storable configurations have been defined. The first is a perigee kick stage packaged to be carried aloft in the Aft Cargo Carrier (Figure 6.2.1.1-1). The second is an identical capability stage packaged to be carried in the Orbiter cargo bay (Figure 6.2.1.1-2).
ACC -- The general arrangement of the ACC configured storable OTV was selected to maximize commonality with the space based configuration described in paragraph 6.2.2. The propellant tanks all use the same diameter tooling, and the concept of extending the engine through the heat shield, enabling payload retrieval, is retained. The stage is built around a subsystem module and an airframe truss. The subsystem module provides mounting space for the avionics components and support for the main propellant tanks. The airframe truss supports the tanks laterally and provides the attachment for the main engines and the aerobrake. Both the subsystem module and the airframe truss will be constructed of graphite epoxy composite materials to minimize weight. The main tanks are sized to contain 37,300 pounds of storable propellants and will be constructed of 15-3-3-3 titanium. The tank size selected is adequate to perform the mission model in 1993 and 1994 within the projected 72,000 lb lift capability of the Space Transportation System (STS) in that timeframe. The main propulsion system engines are two XLR-132 engines scaled up to 7500 pounds thrust. Extendable exit cones will be provided to an exit ratio of 600 to 1. The nozzles will be extended through the aerobrake while firing and retracted inside the brake contour during the aerobraking maneuver. The aerobrake is 23 feet in diameter and will be constructed of a multilayer fabric material of Nicalon, Q-felt, and Nextel sealed with RTV. The fabric will be supported on a graphite polyimide frame or honeycomb. The center section will contain the door through which the engine nozzles will extend and retract. The physical dimensions and weight of the storable OTV require the aerobrake to be no more than 23 feet in diameter. As shown in Figure 6.2.1.1-1, this allows the fully deployed aerobrake to fit within the dimensions of the Aft Cargo Carrier. No deployment mechanisms will be
required. At the end of the mission the outer torus of the aerobrake will be jettisoned and the remainder of the OTV will be installed in the Orbiter bay for return to earth. No further disassembly of the OTV is required to fit within the envelope of the P/L bay. More details of this concept are shown in Figure 6.2.1.1-3.

CARGO BAY -- We have designed a minimum length OTV (Figure 6.2.1.1-2) that will fit within the envelope of the STS P/L Bay and still leave adequate space for the longest payloads identified in the Mission Model. Commonality with subsequent space based designs has been sacrificed to obtain short stage length. Fuel and oxidizer tanks are different diameters to achieve equal, short tank length, and the aerobrake has been mounted on the forward end of the stage, enabling use of four short engines tucked into the corners between tanks. The overall length has been held to approximately 13.5 feet which leaves 46.5 feet for payload and ASE. The four main propulsion propellant tanks, sized for 37,300 pounds of storable propellant, are of 15-3-3-3 titanium alloy and supported in a truss and skin structure of graphite epoxy. The subsystem equipment is fitted into the quadrants between the tanks. The main propulsion system will use the XLR-132 engines with 3750 pounds thrust. Fixed nozzles with an expansion ratio of 400 to 1 were selected to minimize length. A 23 foot diameter deployable aerobrake was selected using the same multilayer fabric material selected for the ACC OTV aerobrake design over graphite polyimide support structure. The center support structure is honeycomb and the outer portion is rib construction. Since the aerobrake will be mounted on the forward end of the stage, the payload interface, instead of the engines, will penetrate the heat shield. Thus this perigee stage will have no capability to retrieve payloads in the aerobraked operational mode. No payload retrieval is required in this mode. At the end of the mission, the outer portion of the brake will be discarded and the remainder of the OTV will be installed in the Orbiter P/L bay for return to earth.

6.2.1.2 Subsystem Summary Description (Ground-Based Storable)

6.2.1.2.1 Aeroassist (Ground-Based Storable) -- The overall layouts of the ground-based storable configurations were shown in Figures 6.2.1.1-1 through -3. The aeroshield devices used with these configurations are similar in concept to those used on the cryogenic configurations discussed in Section 6.1. The diameter has been reduced to 23 feet as the more compact storable stages are more easily protected from the aerodynamic wake and afterbody re-circulation. This smaller diameter results in a higher ballistic coefficient than that experienced on the ground based cryogenic vehicle. The resulting surface insulation temperature is higher, and its thickness is correspondingly greater. The resulting aeroassist parameters are summarized in Table 6.2.1.2.1-1. Note that the stage configured to be carried aloft in the Orbiter cargo bay has its engines on the opposite end from the aeroshield, and no door is needed to permit their use. Note also that the ACC version is small enough to fit within the ACC envelope without being folded. We have maintained the concept of RSI on the central portion of the brake and FSI on the outboard portion.

<table>
<thead>
<tr>
<th>W/CDA</th>
<th>Brake Dia (ft)</th>
<th>Tmax (sec)</th>
<th>Qmax (Btu/ft²)</th>
<th>Tmax (°F)</th>
<th>Thick (In.)</th>
<th>MBr</th>
<th>Design Load psf</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.2</td>
<td>23</td>
<td>FSI</td>
<td>15.7</td>
<td>3700</td>
<td>2570</td>
<td>0.43</td>
<td>0.14</td>
</tr>
<tr>
<td></td>
<td>RSI</td>
<td>18.8</td>
<td>4440</td>
<td>2360</td>
<td>0.43</td>
<td>35</td>
<td>28</td>
</tr>
</tbody>
</table>

6-59
6.2.1.2.2 Propulsion (Ground-Based Storable) - The propulsion characteristics are shown in Table 6.2.1.2.2-1 and the RCS characteristics are shown in Table 6.2.1.2.2-2. Two main engines provide 7500 lbs of thrust each and are based on technology (XLR-132) currently under development at AFRPL. The engine is a gas generator cycle that uses oxidizer cooling and provides a specific impulse of 345.7 seconds at a chamber pressure of 1500 psia and expansion ratio of 600:1. The engine weighs 253 lbs with a two position nozzle, and can be gimbaled up to ±16° to track the center of gravity for engine out operations.

The XLR-132 technology is currently being developed by AFRPL. The expendable engine can be available in 1989 and the reusable engine in 1992 under the current planning and funding schedules. As currently designed, the 3750 lbf engine has a life of 1020 seconds and a capability for 10 starts. Based on discussions with AFRPL, the life is expected to be extended to 5 hours with a possible performance reduction. Reusable XLR-132 engine studies are planned to begin in mid 1985.

The propellants, MMH and N₂O₄ are stored in a four tank configuration which can be returned to the ground in the shuttle payload bay without disassembly. The tanks are manifolded together in parallel flow configuration so that each propellant will be depleted simultaneously. A propellant utilization system is included to assure that the usable propellants in each tank can be depleted with a minimum residual propellant weight. This system consists of propellant utilization probes that provide both a continuous output of liquid level and discrete points to update propellant usage data during engine firings to cancel cumulative errors periodically as the propellant is consumed. The outputs of the computer can be used to vary the consumption from individual tanks to maintain liquid levels within acceptable limits, or to vary the engine mixture ratio to achieve simultaneous depletion of usable propellants. Propellant start traps are also provided.

The pressurization gas is helium stored in high pressure vessels and regulated by electronic pressure regulators. The MMH tank pressure is lower than the N₂O₄ tank because of the lower engine interface pressure; 17 psia for MMH vs 37 psia for N₂O₄. This required an additional regulator at the MMH tank. These are shown in Figure 6.2.1.2.2-1. The nominal flight operating pressure for MMH is 20 psia and for N₂O₂ is 50 psia. This assumes 3 psi delta-P for frictional losses and allows for a 10 psi rise in the N₂O₄ tank during coast because of its higher vapor pressure. The storage system also provides helium for engine valve actuation, main engine turbine spin-up, and purge of oxidizer and fuel pump seals.
<table>
<thead>
<tr>
<th>Category</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>ENGINE</td>
<td>TWO ENGINES, 7500 LBS THRUST EACH, GAS GENERATOR CYCLE Isp = 345.7 SEC, AREA RATIO 600:1</td>
</tr>
<tr>
<td>PROPELLANT DISTRIBUTION</td>
<td>DUAL TANK, PARALLEL FEED, START TRAP</td>
</tr>
<tr>
<td>PRESSURIZATION</td>
<td>HELIUM STORED GAS, REGULATOR CONTROLLED TO PROPELLANT TANKS</td>
</tr>
<tr>
<td>VENT</td>
<td>GROUND VENT ONLY—NONE DURING ASCENT AND FLIGHT. REDUCE PRESSURE PRIOR TO RETURN IN CARGO BAY</td>
</tr>
<tr>
<td>VALVE ACTUATION</td>
<td>ELECTRICAL AND HELIUM, COMMON WITH STAGE PRESSURIZATION SUPPLY</td>
</tr>
<tr>
<td>PROPELLANT UTILIZATION</td>
<td>TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL</td>
</tr>
<tr>
<td>CARGO BAY RETRIEVAL</td>
<td>RETURN STAGE IN CARGO BAY AND DETANK RESIDUALS ON THE GROUND</td>
</tr>
<tr>
<td>THERMAL PROTECTION</td>
<td>HEATER BLANKETS AND MULTILAYER INSULATION</td>
</tr>
<tr>
<td>PROXIMITY OPERATIONS</td>
<td>TWO FAULT TOLERANT</td>
</tr>
<tr>
<td>REDUNDANCY</td>
<td>FAIL SAFE</td>
</tr>
</tbody>
</table>

Table 6.2.1.2.2-1 Ground-Based Storable MPS Summary
Figure 6.2.1.2.2-1 Ground-Based N₂O₄/MMH Schematic

Table 6.2.1.2.2-2 Ground-Based Storable RCS Summary

- **PROPELLANT**
  - HYDRAZINE (N₂H₄) ISP = 230 SEC

- **ROCKET ENGINE MODULE**
  - 30 LB, 7 ENGINES PER MODULE
  - 14 THRUSTERS
  - 3 DOF and -X TRANSLATION
  - FAIL OPERATIONAL

- **PROPELLANT SUPPLY**
  - 24" DIAMETER TANK
  - POSITIVE EXPULSION
  - 400 PSI 2.3:1 BLOWDOWN
  - 150 LBS OF HYDRAZINE MAXIMUM

- **SAFETY**
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS
The vent system has a vent and relief valve on each tank which provide over pressure relief capability and the capability to depressurize the ullage to acceptable values prior to stowage of the OTV in the cargo bay.

The stage will be returned to the ground in the shuttle cargo bay. Propellants will not be dumped to space; unless future analysis shows the combined stage/propellant weight exceeds 32,000 lbs, which is the shuttle gross payload landing capability; the center of gravity is not within the shuttle center of gravity limits; or if the propellant slosh magnitude is not acceptable to the orbiter's autopilot system during reentry. At present the ground-based stage does not exceed these requirements. Residual propellants will be detanked on the ground.

The thermal protection system includes heaters, and multilayer insulation to maintain propellant temperature within acceptable limits.

For proximity operations near the shuttle, the system is dual fault tolerant and for flight operations, the system is fail safe as shown in Figure 6.2.1.2-1.

The reaction control is the same as that use for the ground based cryo (Figure 6.1.1.2.2-3). It uses hydrazine monopropellant pressurized by nitrogen gas operating in a blowdown mode from 400 psi. Fourteen thrusters provide three degrees of freedom and +X translation. The thrusters provide an ISP of 230 seconds. The propellant supply is one 24-inch diameter titanium tank, having a usable propellant capacity of 150 lbs of hydrazine at a 2.3:1 blowdown ratio. The RCS is two fault tolerant for proximity operations.

6.2.1.2.3 Structures and Mechanisms (Ground-Based Storable) - The ACC configuration shown in Figures 6.2.1.1-1 and -3 consists of two 68 in. diameter cylindrical MMH tanks with .75 ellipse lower heads and two 68 in. diameter cylindrical N2O4 oxidizer tanks with .75 ellipse lower heads. Two 7500 lb thrust engines are mounted on a center core truss arrangement. Tanks are supported off conical shaped forward heads by a cross beam arrangement. The 23 ft diameter aerobrake is mounted on the engine support by graphite polyimide struts.

The vehicle is delivered to space in the ACC. With only a 23 ft brake required, a fixed Viking shaped aerobrake can be used. Interface with the ACC and payload adapter is at the crossbeam end of the vehicle. Umbilical provisions with the ACC are also on that end. Avionics are installed on the outside surfaces of the forward crossbeam to provide ready access for replacement.

The vehicle will be returned to earth in the cargo bay of the orbiter after jettisoning the fabric aerobrake. A grapple fixture is provided on the OTV to interface with the orbiter RMS.

The storable tanks are of fusion welded construction of heat treated 15-3-3-3 titanium. If problems are uncovered during the alloy test program, the back up alloy would be 6AL-4V titanium. The conical heads are made in five pieces with a machined cone shaped cap for tank pick up and four formed
and chem milled gores. The elliptical tank head is of similar construction but with a formed center piece. Minimum tank membrane gage is 0.017 inches. The tank barrel is made in two pieces. Main structural crossbeam and engine thrust beam are fabricated from graphite epoxy. Lower tank support beams are of graphite polyimide construction. Aerobrake has a center core of graphite polyimide honeycomb covered with ceramic foam tiles. The remainder of the aerobrake is Nextel, nicalon, ceramic felt, and RTV construction. Generally, similar construction is used on the cargo bay configuration shown in Figure 6.2.1.1-2.

Airborne support equipment (ASE) considerations for the ACC OTV are shown in Figure 6.2.1.2.3-1. The ground-based storable ACC OTV will be supported in the orbiter bay for retrieval at five locations - three forward, two aft. The upper forward mounts will consist of two trunnion and scuff plate fittings mounted to the OTV crossbeam. The trunnion and scuff plate fittings will be of aluminum alloy and located to mate with the orbiter sill longeron bridge fitting. The lower forward mount will consist of one aluminum alloy base plate and trunnion fitting attached also to the OTV crossbeam and mating with an orbiter bay keel fitting. The upper aft mounts will consist of two trunnions and scuff plates attached to the OTV aerobrake structure and mating at the outboard ends with orbiter bay sill longeron bridge fittings. The trunnion and scuff plate fittings will be of aluminum alloy and located to mate with the orbiter sill longeron fittings. The two aft fittings are to be carried up in orbiter payload bay and attached to OTV at rendezvous.

Figure 6.2.1.2.3-1 Ground-Based Storable OTV ASE

NOTE: THIS SAME APPROACH WOULD BE USED TO TRANSPORT THE SPACE BASED STORABLE OTV TO THE SPACE STATION
The payload bay configured storable OTV concept will be mounted for ascent and descent using the same ASE. The required design has not been finalized, but a cradle will be required. The cradle will pick up longeron and keel fittings at the rear of the cargo bay and cantilever the perigee stage, apogee stage and payload stack. For very long payloads, it is likely that longeron fittings on the payload would be beneficial.

6.2.1.2.4 Avionics (Ground-Based Storable) - The ground-based storable avionics configuration (Figure 6.2.1.2.4-1) is essentially identical to the ground-based cryo configuration (Section 6.1.1.2.4). Fuel cell reactant tankage must be added since it cannot be main tank fed as it was in the cryo configuration. Table 6.2.1.2.4-1 reflects the subsystem and system unit and total values for power and weight for this configuration. A more detailed description of this system is provided in Reference 7.

Figure 6.2.1.2.4-1 Block Diagram of the Ground-Based Storable Configuration
6.2.1.2.4.1 Guidance, Navigation and Control - The Guidance, Navigation and Control subsystem is essentially the same as in Section 6.1.1.2.4.1. The engine controller is deleted as it is not required for the storable engine.

6.2.1.2.4.2 Data Management - The Data Management subsystem is the same as in Section 6.1.1.2.4.2.

Table 6.2.1.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 1 of 2)

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Equipment</th>
<th>Weight (lb)</th>
<th>Power (w)</th>
<th>Size (in) H W L</th>
<th>Qty</th>
<th>Wt (lb)</th>
<th>Max</th>
<th>Avg</th>
</tr>
</thead>
<tbody>
<tr>
<td>GN&amp;C</td>
<td>Star Scanner</td>
<td>12</td>
<td>7</td>
<td>7x 7x20</td>
<td>1</td>
<td>12</td>
<td>7</td>
<td>5</td>
</tr>
<tr>
<td></td>
<td>IMU</td>
<td>37</td>
<td>25</td>
<td>6x12x16</td>
<td>1</td>
<td>37</td>
<td>25</td>
<td>25</td>
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<tr>
<td></td>
<td>GPS Receiver</td>
<td>45</td>
<td>35</td>
<td>8x12x16</td>
<td>1</td>
<td>45</td>
<td>35</td>
<td>15</td>
</tr>
<tr>
<td></td>
<td>GPS Antenna-Low Alt</td>
<td>5</td>
<td></td>
<td>6x 6x10</td>
<td>2</td>
<td>10</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>GPS Antenna-Hi Alt</td>
<td>5</td>
<td></td>
<td>18x18x26</td>
<td>1</td>
<td>5</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Flight Controller</td>
<td>45</td>
<td>120</td>
<td>8x 8x16</td>
<td>1</td>
<td>45</td>
<td>120</td>
<td>120</td>
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<tr>
<td></td>
<td><strong>Subsystem Total</strong></td>
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<td>10</td>
<td>60</td>
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<td>20</td>
<td>120</td>
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<tr>
<td></td>
<td>&amp; Mass Memory</td>
<td></td>
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<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td><strong>Subsystem Total</strong></td>
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<td>120</td>
<td></td>
<td></td>
<td>85</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Telemetry and Command</td>
<td>Command &amp; Data</td>
<td>15</td>
<td>35</td>
<td>6x 8x10</td>
<td>1</td>
<td>15</td>
<td>35</td>
<td>22</td>
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<tr>
<td></td>
<td>Handling</td>
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<td></td>
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<td></td>
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<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>TLM Power Supply</td>
<td>7</td>
<td>10</td>
<td>4x 7x 7</td>
<td>1</td>
<td>7</td>
<td>10</td>
<td>5</td>
</tr>
<tr>
<td></td>
<td>Deploy Timer</td>
<td>6</td>
<td>6</td>
<td>3x 4x 7</td>
<td>2</td>
<td>12</td>
<td>12</td>
<td>4</td>
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6-67
Table 6.2.1.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 2 of 2)

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Equipment</th>
<th>Weight (lb)</th>
<th>Power (w)</th>
<th>Size (in)</th>
<th>Total Qty</th>
<th>Wt (lb)</th>
<th>Max</th>
<th>Avg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Communications and Tracking</td>
<td>STDN/TDRS Xponder</td>
<td>16</td>
<td>55</td>
<td>6x 6x14</td>
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<td>16</td>
<td>55</td>
<td>55</td>
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<tr>
<td></td>
<td>20w RF Power Amp</td>
<td>6</td>
<td>125</td>
<td>3x 6x10</td>
<td>1</td>
<td>6</td>
<td>125</td>
<td>40</td>
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<tr>
<td></td>
<td>S-Band RF System</td>
<td>90</td>
<td>30</td>
<td>2</td>
<td>180</td>
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<td>EPS</td>
<td>Fuel Cell (FC)</td>
<td>35</td>
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<td>11x12x12</td>
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<td></td>
</tr>
<tr>
<td></td>
<td>FC Radiators</td>
<td>35</td>
<td></td>
<td>35ft²x2&quot;</td>
<td>1</td>
<td>35</td>
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</tr>
<tr>
<td></td>
<td>FC Reactants</td>
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<tr>
<td></td>
<td>FC Reactant Tank</td>
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<td></td>
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<tr>
<td></td>
<td>LH₂ &amp; Plumb</td>
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<td></td>
<td></td>
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<td></td>
</tr>
<tr>
<td></td>
<td>FC Reactant Tank</td>
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<td></td>
<td>38</td>
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<td></td>
</tr>
<tr>
<td></td>
<td>LO₂ &amp; Plumb</td>
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</tr>
<tr>
<td></td>
<td>FC Coolant</td>
<td>10</td>
<td></td>
<td></td>
<td></td>
<td>10</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>FC H₂O Tank</td>
<td>13</td>
<td></td>
<td></td>
<td></td>
<td>13</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Power Control &amp;</td>
<td>27</td>
<td>10</td>
<td>6x 8x12</td>
<td>2</td>
<td>54</td>
<td>20</td>
<td>20</td>
</tr>
<tr>
<td></td>
<td>&amp; Distribution</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
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<td></td>
</tr>
<tr>
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<td>Heaters</td>
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<td></td>
<td></td>
<td>50</td>
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<td>50</td>
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<td></td>
<td>Engine Power</td>
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<td></td>
<td></td>
<td></td>
<td>200</td>
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<tr>
<td></td>
<td>Subsystem Total</td>
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<td></td>
<td></td>
<td>297</td>
<td>270</td>
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<td></td>
<td></td>
<td>707</td>
<td>874</td>
<td>476</td>
</tr>
</tbody>
</table>
6.2.1.2.4.3 Telemetry and Command - The Telemetry and Command subsystem is the same as in Section 6.1.1.2.4.3.

6.2.1.2.4.4 Communications and Tracking - The Communications and Tracking subsystem is the same as in Section 6.1.1.2.4.4.

6.2.1.2.4.5 Electrical Power - The fuel cell reactant tanks are added to this configuration as reactants cannot be taken from the propellant system. The EPS design (Figure 6.2.1.2.4-2) is essentially the same as in Section 6.1.1.2.4.5. Fuel cells are sized for 1.1 kw including a 20% design margin.

Figure 6.2.1.2.4-2 EPS Configuration for Ground-Based Storable Configuration
6.1.1.2.5 **Thermal Control (Ground-Based Storable)** - This configuration utilizes a fuel cell power system. The fuel cell thermal control system is sized for an OTV continuous flight power requirement of 1.1 KW and a 2 day flight duration. The fuel cell thermal control system requires 35 sq ft of radiator area to dissipate the fuel cell waste heat effectively. The radiator faces outboard and is mounted on one of the oxidizer tanks.

The avionics are to be passively cooled with the components mounted to high mass heat sink structures which face outboard and have a low alpha/epsilon paint. Some components may require thermostatically controlled heater strips. The avionics bay structure protects the avionics and thermal blankets on all or most of the avionics components since they have no direct view to space. Component surface finishes and mounting techniques shall be defined as the OTV design develops.

The payload/OTV interface is made nearly adiabatic. Approximately 25 to 50 layers of insulation blanket (double aluminized Kapton MLI) is located at the interface.

The OTV reaction control system (RCS) requires thermal protection for the RCS tank, feedlines, and propulsion modules. The RCS tank has an MLI blanket (10 layers) and strip heaters. The feedlines contain hydrazine (freezing point of 35°F) and require low power strip heaters (approximately 25 watts) and one or two layers of thermally insulating blankets. The RCS modules and feedlines will be maintained at sufficiently high temperatures by "thermal pulsing" techniques (i.e., periodic module firing).

The N₂O₄ and MMH tanks shall be insulated with two layers of Kapton thermal blankets. Thermostatically controlled strip heaters are used to maintain sufficiently high propellant tank temperatures. The short flight duration should enable the propellant system capacitance to maintain acceptable propellant temperatures with a minimal supplemental heater power requirement.

The helium pressurant tank requires MLI, since the tank is of composite construction. Thermostatically controlled strip heaters are necessary to ensure adequate helium temperature/pressures.

The heating effects of the engine nozzles are of no concern.

6.2.1.3 **System Weight Summary - Ground-Based Storable** - Total Flight vehicle weight for the ground-based storable configuration is presented in Table 6.2.1.3-1 for the ACC concept, in Table 6.2.1.3-2 for the cargo bay concept. Dry weight is developed in detail. Propulsive and nonpropulsive fluid loadings are summarized. The breakdown has not been rearranged into MSFC's suggested format as this approach was deleted from contention at an earlier stage in the study.
Table 6.2.2.3-1 System Weight Summary (Ground-Based Storable)

Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - ACC Configured

<table>
<thead>
<tr>
<th>Description</th>
<th>Weight (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Structure</strong></td>
<td></td>
</tr>
<tr>
<td>Basic Airframe Truss</td>
<td>120</td>
</tr>
<tr>
<td>OX Tank</td>
<td>196</td>
</tr>
<tr>
<td>Tank (2)</td>
<td>158</td>
</tr>
<tr>
<td>Fwd &amp; Aft Attach</td>
<td>38</td>
</tr>
<tr>
<td>Fu Tank</td>
<td>96</td>
</tr>
<tr>
<td>Tank (2)</td>
<td>60</td>
</tr>
<tr>
<td>Fwd &amp; Aft Attach</td>
<td>36</td>
</tr>
<tr>
<td>Engine Truss &amp; Attch</td>
<td>101</td>
</tr>
<tr>
<td>Engine Truss</td>
<td>67</td>
</tr>
<tr>
<td>Hardpoints - Engine Attch</td>
<td>30</td>
</tr>
<tr>
<td>Attach - Engine Actuators</td>
<td>4</td>
</tr>
<tr>
<td>Subsystem Module</td>
<td>208</td>
</tr>
<tr>
<td>PIDA Grapple Fixture</td>
<td>20</td>
</tr>
<tr>
<td>RMS Grapple Fixture &amp; Struts</td>
<td>30</td>
</tr>
<tr>
<td>Orbiter Pickup Trunnion (3)</td>
<td>15</td>
</tr>
<tr>
<td>P/L or ACC Attch</td>
<td>40</td>
</tr>
<tr>
<td><strong>Aerobrake Assy - 23 ft</strong></td>
<td>538</td>
</tr>
<tr>
<td><strong>Environmental Control</strong></td>
<td>117</td>
</tr>
<tr>
<td><strong>Meteoroid Protection</strong></td>
<td>N/A</td>
</tr>
<tr>
<td><strong>Thermal Control/Protection</strong></td>
<td>117</td>
</tr>
<tr>
<td>Ox Tanks - MLI &amp; Heater Tape</td>
<td>41</td>
</tr>
<tr>
<td>Fu Tanks - MLI &amp; Heater Tape</td>
<td>35</td>
</tr>
<tr>
<td>Engine Truss - MLI</td>
<td>1</td>
</tr>
<tr>
<td>Hydrazine Tank - MLI &amp; Heater Tape</td>
<td>11</td>
</tr>
<tr>
<td>He Tanks (2) - MLI</td>
<td>1</td>
</tr>
<tr>
<td>Engine Compartment - MLI</td>
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</tr>
<tr>
<td>Propellant Lines, Components, MLI, Heater Tape, Coatings, etc.</td>
<td>23</td>
</tr>
<tr>
<td>Description</td>
<td>Weight (lb)</td>
</tr>
<tr>
<td>-------------------------------------------------</td>
<td>-------------</td>
</tr>
<tr>
<td><strong>Main Propulsion System</strong></td>
<td></td>
</tr>
<tr>
<td>Engine (2): XLR-132 7500# Thrust Ea</td>
<td>506</td>
</tr>
<tr>
<td>Propellant Distribution System</td>
<td>194</td>
</tr>
<tr>
<td>Pressurization/Pneumatic System</td>
<td>170</td>
</tr>
<tr>
<td>ACS Common Feed</td>
<td>N/A</td>
</tr>
<tr>
<td>PU/Acquisition System</td>
<td>N/A</td>
</tr>
<tr>
<td>Instrumentation</td>
<td>5</td>
</tr>
<tr>
<td>Actuators (4) – Electrical</td>
<td>64</td>
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<tr>
<td><strong>Orientation Control</strong></td>
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<tr>
<td>ACS Subsystem - Hydrazine</td>
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<tr>
<td>Rocket Engine Modules (2)</td>
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<tr>
<td>Mounting - REMS</td>
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<td>Hydrazine Tank (1)</td>
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<td>Mounting - Tank</td>
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<td>Propel. Distribution &amp; Pressurization</td>
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<td><strong>Electrical Power</strong></td>
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<td>Fuel Cell System</td>
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<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
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<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
<td>38</td>
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<tr>
<td>Radiator System</td>
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<tr>
<td>Residual H₂O System</td>
<td>13</td>
</tr>
<tr>
<td>Wire Harness, Connectors, etc.</td>
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</tr>
<tr>
<td>Mounting Provisions</td>
<td>38</td>
</tr>
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</table>
Table 6.2.1.3-1 Weight Statement  
Space-Based Storable  
37.3K Propellant Load  
Pge Stage - ACC Configured  
(Continued)

<table>
<thead>
<tr>
<th>Description</th>
<th>Weight (lb)</th>
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<tr>
<td>Avionics</td>
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<td>Avionics (Equipment List)</td>
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</tr>
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<td>Mounting Provisions</td>
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<td>Dry Weight</td>
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<td>Contingency (15%)</td>
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<td>Total Dry Weight</td>
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</tr>
<tr>
<td>Main Propellant (MR 2:1 Ox Wt to Fu Wt)</td>
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<tr>
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<td>24866</td>
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<tr>
<td>MMH</td>
<td>12434</td>
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<td>ACS Propellant &amp; Pressurant</td>
<td>157</td>
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<td>N2H4</td>
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<tr>
<td>N2</td>
<td>7</td>
</tr>
<tr>
<td>FC Reactant &amp; Coolant</td>
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<tr>
<td>Reactant</td>
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<tr>
<td>Coolant</td>
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<td>Total Loaded Weight</td>
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</tbody>
</table>

\[
\lambda = \frac{37300 \text{ (Main Propellants)}}{41361 \text{ (Ignition Weight)}} = 0.902
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<th>Description</th>
<th>Weight (lb)</th>
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<tbody>
<tr>
<td>Structure</td>
<td>1340</td>
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<td>Basic Airframe Truss</td>
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</tr>
<tr>
<td>Ox Tank</td>
<td>250</td>
</tr>
<tr>
<td>Tank (2)</td>
<td>180</td>
</tr>
<tr>
<td>Fwd &amp; Aft Attach</td>
<td>70</td>
</tr>
<tr>
<td>Fu Tank</td>
<td>208</td>
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<tr>
<td>Tank (2)</td>
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</tr>
<tr>
<td>Fwd &amp; Aft Attach</td>
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<tr>
<td>Engine Truss &amp; Attach</td>
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<tr>
<td>Engine Truss</td>
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<td>Hardpoints &amp; Struts</td>
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<tr>
<td>Attach - Engine Actuators</td>
<td>8</td>
</tr>
<tr>
<td>PIDA Grapple Fixture</td>
<td>20</td>
</tr>
<tr>
<td>RMS Grapple Fixture &amp; Struts</td>
<td>30</td>
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<tr>
<td>Orbiter Pickup Trunnion (3)</td>
<td>15</td>
</tr>
<tr>
<td>P/L Attach</td>
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<td>Aerobrake Assy - 23 ft</td>
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</tr>
<tr>
<td>Environmental Control</td>
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<tr>
<td>Meteoroid Protection</td>
<td>N/A</td>
</tr>
<tr>
<td>Thermal Control/Protection</td>
<td>117</td>
</tr>
<tr>
<td>Ox Tanks - MLI &amp; Heater Tape</td>
<td>41</td>
</tr>
<tr>
<td>Fu Tanks - MLI &amp; Heater Tape</td>
<td>35</td>
</tr>
<tr>
<td>Engine Truss - MLI</td>
<td>1</td>
</tr>
<tr>
<td>Hydrazine Tank - MLI &amp; Heater Tape</td>
<td>11</td>
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<tr>
<td>He Tanks (2) - MLI</td>
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<tr>
<td>Engine Compartment - MLI</td>
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<td>Propellant Lines, Components, MLI, Heater Tape, Coatings, etc.</td>
<td>23</td>
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</table>
Table 6.2.1.3-2 Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - P/L Bay Configured
(Continued)

<table>
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<th>Weight (lb)</th>
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<tr>
<td>Main Propulsion System</td>
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<tr>
<td>Engine (4): XLR-132 3750# Thrust Ea</td>
<td>456</td>
</tr>
<tr>
<td>Propellant Distribution System</td>
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<tr>
<td>Pressurization/Pneumatic System</td>
<td>170</td>
</tr>
<tr>
<td>ACS Common Feed</td>
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</tr>
<tr>
<td>PU/Acquisition System</td>
<td>N/A</td>
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<tr>
<td>Instrumentation</td>
<td>5</td>
</tr>
<tr>
<td>Actuators (8) - Electrical</td>
<td>128</td>
</tr>
<tr>
<td>Orientation Control</td>
<td>116</td>
</tr>
<tr>
<td>ACS Subsystem - Hydrazine</td>
<td></td>
</tr>
<tr>
<td>Rocket Engine Modules (2)</td>
<td>32</td>
</tr>
<tr>
<td>Mounting - REMS</td>
<td>3</td>
</tr>
<tr>
<td>Hydrazine Tank (1)</td>
<td>41</td>
</tr>
<tr>
<td>Mounting - Tank</td>
<td>6</td>
</tr>
<tr>
<td>Propel. Distribution &amp; Pressurization</td>
<td>34</td>
</tr>
<tr>
<td>Electrical Power</td>
<td></td>
</tr>
<tr>
<td>Fuel Cell System</td>
<td>70</td>
</tr>
<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
<td>42</td>
</tr>
<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
<td>38</td>
</tr>
<tr>
<td>Radiator System</td>
<td>35</td>
</tr>
<tr>
<td>Residual H2O System</td>
<td>13</td>
</tr>
<tr>
<td>Wire Harness, Connectors, etc.</td>
<td>116</td>
</tr>
<tr>
<td>Mounting Provisions</td>
<td>39</td>
</tr>
</tbody>
</table>
Table 6.2.1.3-2 Weight Statement
Space-Based Storable
37.3K Propellant Load
Pge Stage - P/L Bay Configured
(Continued)

<table>
<thead>
<tr>
<th>Description</th>
<th>Weight (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Avionics</td>
<td>456</td>
</tr>
<tr>
<td>Avionics (Equipment List)</td>
<td>410</td>
</tr>
<tr>
<td>Mounting Provisions</td>
<td>46</td>
</tr>
<tr>
<td><strong>Dry Weight</strong></td>
<td><strong>3906</strong></td>
</tr>
<tr>
<td>Contingency (15%)</td>
<td>586</td>
</tr>
<tr>
<td><strong>Total Dry Weight</strong></td>
<td><strong>4492</strong></td>
</tr>
<tr>
<td>Main Propellants (MR 2:1 Ox Wt to Fu Wt)</td>
<td>37300</td>
</tr>
<tr>
<td>N2O4</td>
<td>24866</td>
</tr>
<tr>
<td>MMH</td>
<td>12434</td>
</tr>
<tr>
<td>Pressurant (He) - MPS</td>
<td>13</td>
</tr>
<tr>
<td>ACS Propellant &amp; Pressurant</td>
<td>157</td>
</tr>
<tr>
<td>N2H4</td>
<td>150</td>
</tr>
<tr>
<td>N2</td>
<td>7</td>
</tr>
<tr>
<td>FC Reactant &amp; Coolant</td>
<td>22</td>
</tr>
<tr>
<td>Reactant</td>
<td>12</td>
</tr>
<tr>
<td>Coolant</td>
<td>10</td>
</tr>
<tr>
<td><strong>Total Loaded Weight</strong></td>
<td><strong>41984</strong></td>
</tr>
</tbody>
</table>

\[
\lambda = \frac{37300 \text{ (Main Propellants)}}{41984 \text{ (Ignition Weight)}} = 0.888
\]
6.2.1.4 Performance on Model Missions: Ground-Based Storables - The ground-based storable stage is used as a perigee stage, requiring the use of an additional expendable kick stage to perform some part of the mission, such as circularizing the payload at geosynchronous altitude. In such cases, the EKS was assumed to have a mass fraction of 0.95, an Isp of 310 seconds, and was sized to be just large enough to perform the mission at hand, i.e. "custom fit" to each particular mission. For this reason the performance curve graphs for the GB ACC storable perigee stage is qualitatively different than those for the cryogenic stages which performed their missions "solo". Table 6.2.1.4-1 summarizes the propellant load required and the gross weight of this stage on each of the Rev. 8 model missions on which it will be used. Gross weight includes the weight of the required kick stage. Figure 6.2.1.4-1 summarizes the performance of this stage to high circular orbits assuming full utilization of the Shuttle launch capability (72,000 lb to 140 nmi when launched east from KSC), and the use of expendable apogee kick stages.

Table 6.2.1.4-1
Performance Analysis for Required Missions Storable, Ground-Based ACC, 37.3K Perigee Stage
(Rev. 7 Missions)

Isp = 345.7 sec.

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP (lb)</th>
<th>P/L DN (lb)</th>
<th>OTV PROP.(lb)</th>
<th>OTV + EKS + P/L (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>19036</td>
<td>6000</td>
<td>0</td>
<td>18914</td>
<td>36780</td>
</tr>
<tr>
<td>19031</td>
<td>8025*</td>
<td>0</td>
<td>23560</td>
<td>45751</td>
</tr>
<tr>
<td>19031</td>
<td>9317*</td>
<td>0</td>
<td>26546</td>
<td>51498</td>
</tr>
<tr>
<td>18724</td>
<td>10000</td>
<td>0</td>
<td>28133</td>
<td>54543</td>
</tr>
<tr>
<td>18064</td>
<td>10163</td>
<td>0</td>
<td>28512</td>
<td>55271</td>
</tr>
<tr>
<td>13006</td>
<td>12017</td>
<td>0</td>
<td>32851</td>
<td>63571</td>
</tr>
</tbody>
</table>

*Early Year DoD Equivalent Payloads Projected by MMA
6.2.2 **Space-Based Storable Family**

6.2.2.1 **General Arrangement (Space-Based Storable)** - The space-based storable family of vehicles uses one perigee stage configuration and two two-stage configurations to meet HEO and planetary mission requirements (Figures 6.2.2.1-1 to -3). These configurations use 3 stage sizes, further details of which are shown in Figures 6.2.2.1-4 to -6. Capture of the large lunar missions requires use of a low technology cryogenic perigee stage under the 53,000 lb propellant capacity storable stage.

Either evolving from an initially ground based OTV or starting as a space based OTV, our study shows the basic configuration of the storable space-based GEO delivery OTV, shown in Figure 6.2.2.1-1, should be essentially the same. The subsystem module/airframe truss forms an efficient structure to support the main propulsion system, propellant tanks, and avionics and electric power system equipment. Repackaging of the avionics and propulsion components is necessary to accommodate the space based maintenance activities involving a limited number of technicians for a minimum time. Accessibility to the equipment both by astronauts in EVA gear and by robotics has been a consideration in configuring the space-based vehicles. The basic ground-based vehicle and the space-based vehicles have much in common. The avionics components are essentially the same except for added redundancy. The main engines and feed system are the same 7500 lb thrust XLR-132 engines and feed system selected for the ACC ground-based OTV. The main propellant tanks are sized for the more ambitious missions identified for the later years when the space station will be operational. The tank diameter is the same as selected for the ground-based ACC OTV to assure common tooling between the potential ground based program and the space based program, and to enable delivery of
the assembled OTV to the Space Station in the Orbiter payload bay. The OTV configuration selected for delivery of payloads to GEO is shown in Figure 6.2.2.1-1. The vehicle will operate as a perigee stage and an expendable AKM will be provided to insert the payloads into GEO-synchronous orbit. The aerobrake is constructed of the same materials as the ground based ACC brake. The size has been increased commensurate with the return weight of the vehicle. Although shown with a 25 ft diameter brake, this vehicle will be capable of being fitted with a 32 ft diameter brake when required to return with additional equipment (such as the multiple payload carrier) after delivering multiple payloads.

Figure 6.2.1.1-1 Space-Based GEO Delivery Vehicle
The unmanned servicing missions to GEO will be performed with a two stage vehicle (Figure 6.2.2.1-2) made up of the GEO delivery vehicle, described earlier, as the first stage and a smaller propellant capacity stage as the second stage. The first stage will perform the perigee burn, separate from the second stage and payload, coast out to GEO altitude, return for the aerobrake maneuver and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the payload into GEO orbit. The stage will stay in the vicinity of the servicer through the duration of the mission and then perform the deorbit burn to bring the servicer back to the Space Station. The smaller second stage is configured similar to the first stage but with smaller tanks and shorter airframe structure. The subsystem module, engines and feed system, avionics equipment, and electric power system are the same for both stages. Tanks for the EPS fuel cell reactants will be larger for the second stage because of the longer duration of the stage two mission. The diameter of the tanks for the second stage are the same as the first stage tanks in order that the tooling for the domes can be common. By welding the domes of the fuel tanks together with no barrel section and adding a short barrel section in the oxidizer tanks, the propellant capacity for the second stage is approximately 25,400 pounds. The aerobrake for the second stage is sized at 32 ft in diameter to bring the unmanned servicer back through the aeromanuever. Construction of the brake is the same for both stages.

Figure 6.2.2.1-2  Space-Based Unmanned Servicing Vehicle
The manned servicing missions to GEO will be performed with a two stage vehicle (Figure 6.2.2.1-3) made up of the GEO delivery vehicle, slightly reoutfitted, as the second stage and a larger propellant capacity stage as the first stage. The first stage will perform the perigee burn, separate from the second stage and manned capsule, coast out to GEO altitude, return for the aerobrake maneuver and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the manned payload into GEO orbit. The stage will stay in the vicinity of the manned capsule through the duration of the servicing mission and then perform the deorbit maneuvers to bring the servicer back to the Space Station. The larger first stage is configured for maximum commonality with the second stage. As for the small stage for the unmanned servicing vehicle, the major difference is in the tank length and the airframe truss. The tanks are lengthened, but retain the same diameter for tooling commonality, to provide capacity for 90,000 lb of propellant. Because of the mass of the complete vehicle/payload stack, two additional engines have been added. The four engine arrangement uses the same 7500 pound thrust XLR-132 engine but will require a different feed system. The 53,000 pound propellant capacity second stage is the same basic stage as the GEO delivery vehicle and the first stage of the unmanned servicing vehicle except with a larger diameter aerobrake and larger capacity fuel cell reactant tanks. The fuel cell reactant tanks are sized for support of the 24 day manned mission. The aerobrake is increased to 41 ft in diameter because of the weight of the returning stage and manned capsule.
ORIGINAL PAGE IS
OF POOR QUALITY
6.2.2.2 Subsystem Summary Description (Space Based Storable)

6.2.2.2.1 Aeroassist (Space-Based Storable) - The overall layouts of the space based storable configurations are shown in Figures 6.2.1.1-1 through -6. The aeroshield devices used with these configurations are similar in concept to those used on the cryogenic configurations discussed in Section 6.1. Three different aeroshield diameters have been established to protect the several vehicles to be returned. Table 6.2.2.2.1-1 shows the return situations involved. Empty stages of two different sizes will be returned. Three payloads (the multiple payload carrier, the unmanned servicer, and the manned spacecraft) and the appropriate stage will be returned. The table shows the aeroshield diameter required to perform each of these returns. In order to simplify the array of aeroshields to be designed and supported in space, we have used the 32 foot aeroshield to perform the functions indicated for the 28 and 30 foot aeroshields as well. The ballistic coefficients, brake loadings temperatures and surface insulation thicknesses associated with these cases is summarized in Table 6.2.2.2.1-1. The aeroshield weights presented in Section 6.2.2.3 reflect these data.
Table 6.2.2.1-1  Space-Based Storable Aeroshield

<table>
<thead>
<tr>
<th>Aeroshield Parameter</th>
<th>W/CDA</th>
<th>Brake Diameter (Ft)</th>
<th>TPS</th>
<th>Q&lt;sub&gt;max&lt;/sub&gt; (Btu/ft&lt;sup&gt;2&lt;/sup&gt;)</th>
<th>Q&lt;sub&gt;max&lt;/sub&gt; (Btu/ft&lt;sup&gt;2&lt;/sup&gt;)</th>
<th>t&lt;sub&gt;max&lt;/sub&gt; (°F)</th>
<th>TPS Thickness (In.)</th>
<th>W&lt;sub&gt;Brake&lt;/sub&gt;</th>
<th>W&lt;sub&gt;Return&lt;/sub&gt;</th>
</tr>
</thead>
<tbody>
<tr>
<td>Return 53 k-lb Stg</td>
<td>7.46</td>
<td>25</td>
<td>FSI</td>
<td>16.7</td>
<td>3950</td>
<td>2670</td>
<td>0.45</td>
<td>0.10</td>
<td>43</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RSI</td>
<td>20.0</td>
<td>4740</td>
<td>2530</td>
<td>0.52</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Return F/L Carrier</td>
<td>7.62</td>
<td>30</td>
<td>FSI</td>
<td>15.8</td>
<td>3700</td>
<td>2590</td>
<td>0.43</td>
<td>0.12</td>
<td>43</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RSI</td>
<td>19.0</td>
<td>4440</td>
<td>2430</td>
<td>0.48</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Return 90 k-lb Stg</td>
<td>7.95</td>
<td>28</td>
<td>FSI</td>
<td>15.8</td>
<td>3750</td>
<td>2590</td>
<td>0.44</td>
<td>0.09</td>
<td>46</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RSI</td>
<td>19.0</td>
<td>4500</td>
<td>2430</td>
<td>0.50</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Return Servicer</td>
<td>8.62</td>
<td>32</td>
<td>FSI</td>
<td>15.9</td>
<td>3750</td>
<td>2600</td>
<td>0.44</td>
<td>0.10</td>
<td>52</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RSI</td>
<td>19.1</td>
<td>4500</td>
<td>2440</td>
<td>0.50</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Manned Return</td>
<td>11.00</td>
<td>41</td>
<td>FSI</td>
<td>16.1</td>
<td>3800</td>
<td>2610</td>
<td>0.45</td>
<td>0.07</td>
<td>68</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RSI</td>
<td>19.3</td>
<td>4560</td>
<td>2460</td>
<td>0.52</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>


6.2.2.2.2 Propulsion (Space-Based Storable) - The propulsion characteristics are shown in Table 6.2.2.2.2-1 and the reaction control characteristics are shown in Table 6.2.2.2.2-2. The main propulsion system for the space-based storable is similar to the ground-based storable with the following additions.

Two to four main engines will be required to fly the three stages that have been defined. The engines will be capable of readily being maintained at the Space Station. The engines will also have provisions for the OTV computer to monitor the health of the engine during flight. The 90K perigee stage will use four 7500 lb thrust engines, the 25K second stage will use two 7500 lb, and the 53K second stage and perigee stage will use two 7500 lb thrust engines.

The propellant distribution system components will be modularized for space maintenance and health monitoring provisions will be added. In addition a total tank propellant acquisition system to allow for filling and draining the tanks at the Space Station will be required. The schematic is shown in Figure 6.2.2.2.2-1.

Table 6.2.2.2.2-1 Space-Based Storable MPS Summary

<table>
<thead>
<tr>
<th>Component</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>ENGINE</td>
<td>TWO TO FOUR ENGINES, 7500 LB THRUST EACH, GAS GENERATOR CYCLE, $I_{sp} = 345.7$ SEC, AREA RATIO 600:1</td>
</tr>
<tr>
<td>PROPELLANT DISTRIBUTION</td>
<td>DUAL TANK PARALLEL FEED, FULL TANK ACQUISITION FOR SERVICING AT SPACE STATION</td>
</tr>
<tr>
<td>PRESSURIZATION</td>
<td>HELIUM STORED GAS, REGULATOR CONTROLLED</td>
</tr>
<tr>
<td>VENT</td>
<td>SPACE STATION VENT ONLY</td>
</tr>
<tr>
<td>VALVE ACTUATION</td>
<td>ELECTRICAL AND HELIUM, COMMON WITH STAGE PRESSURIZATION SUPPLY</td>
</tr>
<tr>
<td>PROPELLANT UTILIZATION</td>
<td>TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL</td>
</tr>
<tr>
<td>THERMAL PROTECTION</td>
<td>HEATER BLANKETS AND MLI</td>
</tr>
<tr>
<td>MAINTENANCE</td>
<td>COMPONENT AND ENGINE MODULARITY</td>
</tr>
</tbody>
</table>
Table 6.2.2.2-2 Space-Based Storable RCS Summary

- **PROPELLANT**
  - N₂O₄/MMH

- **ROCKET ENGINE MODULE**
  - NEW DESIGN BASED ON SHUTTLE TECHNOLOGY $I_{sp} = 280$ SEC
  - THRUST 100 lbf, 14 THRUSTERS (7 THRUSTERS PER MODULE)
  - 3 DOF FOR PERIGEE STAGE, +X TRANSLATION
  - 3 DOF FOR APOGEE STAGE, +X TRANSLATION
  - FAIL OPERATIONAL

- **PROPELLANT SUPPLY**
  - COMMON STORAGE WITH MPS TANKS
  - MPS ENGINE PUMP FEED TO ACCUMULATORS
  - FEED TO ACS ENGINES AT 400 PSI 3:1 BLOWDOWN
  - 20" ID TANKS WITH SCREEN ACQUISITION DEVICE
  - 430 LB N₂O₄/MMH AT A MR OF 1.65:1

- **SAFETY**
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS

Figure 6.2.2.2.2-1 Space Based N₂O₄/MMH Schematic
The MMH tank operating pressure will remain the same as the ACC OTV. However, the goal of the AFRPL XLR-132 design is to reduce the N₂O₄ NPSH by about 15 feet. This corresponds to about a 10 psi N₂O₄ tank pressure reduction.

For the second stage application, propellants are stored in the main tanks and are transferred to the RCS accumulators from the high pressure sections of the main engine turbo pumps during engine burns. The 20 inch diameter accumulators have screen liquid acquisition devices to prevent gas ingestion during expulsion to the thruster system at about 400 psia. The system will provide 60,000 lb·sec of total impulse between recharges from the MPS engine. The thruster design will be based on shuttle RCS technology level, therefore system sizing has been based on a Isp of 280 seconds. For perigee operation the tanks should not require resupply. Initial RCS propellant loading will be done during MPS fill before the accumulators are pressurized to 400 psia.

The RCS system is isolated in a dual fault tolerant manner for proximity operations as shown in the schematic, Figure 6.2.2.2.2-2.

![Figure 6.2.2.2.2-2 Space-Based N₂O₄/MMH RCS Schematic](image-url)
6.2.2.2.3 Structures and Mechanisms (Space-Based Storable) - The space based storable OTV requires three different sized stages (Figures 6.2.1.1-4 to -6) to satisfy the baseline mission requirements. The fleet will consist of a 53K OTV, a 25.4K OTV, and a 90K OTV. All configurations use two 68 in diameter cylindrical MMH tanks with 0.75 ellipse lower heads and two 68 in diameter cylindrical N2O4 oxidizer tanks with 0.75 ellipse lower heads. Engines are mounted on a center core truss arrangement. Tanks are supported off conical shaped forward heads by a crossbeam arrangement. The aerobrake is mounted on the engine support by graphite polyimide struts.

The vehicles are configured to be delivered to space fully assembled (except for aerobrake) in the space shuttle cargo bay. The aerobrakes will also be delivered in the shuttle cargo bay and deployed and attached to the OTV in space.

The avionics are installed on the outer surfaces of the crossbeam to provide ready access for removal/replacement. The storable tanks are of fusion welded construction of heat treated 15-3-3-3 titanium. If problems are encountered during the alloy test program, the back up alloy would be 6AL 4 V titanium. The conical heads are made in five pieces with a machined cone shaped cap for tank pickup and four formed and chem milled gores. The elliptical tank head is made of similar construction but with a formed center piece. Tank barrel is made in two pieces. Minimum tank membrane gage is 0.006. If difficulties are encountered in handling 0.006 thick tanks, membrane thickness will be increased to what is required.

Main structural crossbeam and engine thrust structure are fabricated from graphite epoxy. Lower tank support beams are of graphite polyimide construction. Aerobrakes consist of a center core structure of graphite polyimide honeycomb and ceramic foam tiles. The remainder of the aerobrake is a nicalon, ceramic felt and sealed Nextel layup.

The major difference between the stages are total length, aerobrake diameter, and number of engines. A table of aerobrake diameter and number of engines per vehicle is shown below.

<table>
<thead>
<tr>
<th>Stage</th>
<th>No. Engines</th>
<th>A/B Diameter</th>
</tr>
</thead>
<tbody>
<tr>
<td>25.4K OTV</td>
<td>2</td>
<td>32 ft</td>
</tr>
<tr>
<td>53K OTV</td>
<td>2</td>
<td>25, 32, &amp; 41 ft</td>
</tr>
<tr>
<td>90K OTV</td>
<td>4</td>
<td>32 ft</td>
</tr>
</tbody>
</table>

The vehicles are equipped with crane and cradle provisions for handling at the Space Station. In addition, major components such as aerobrakes and tanks have grapple provisions for component changeout at the Space Station.
6.2.2.2.4 Avionics (Space-Based Storable) - The space-based storable avionics configuration (Figure 6.2.2.2.4-1) is essentially identical to the space-based cryo configuration (Section 6.1.2.2.4). Fuel cell reactant tankage must be added since it cannot be main tank fed as it was in the cryo configuration. Table 6.2.2.2.4-1 reflects the subsystem and system unit and total values for power and weight for this configuration.

6.2.2.2.4.1 Guidance, Navigation and Control - The Guidance, Navigation and Control subsystem is essentially the same as in Section 6.1.1.2.4.1. The engine controller is deleted as it is not required for the storable engine.

6.2.2.2.4.2 Data Management - The Data Management subsystem is the same as in Section 6.1.2.2.4.2.

6.2.2.2.4.3 Telemetry and Command - The Telemetry and Command subsystem is the same as in Section 6.1.2.2.4.3.

6.2.2.2.4.4 Communications and Tracking - The Communications and Tracking subsystem is the same as in Section 6.1.2.2.4.4.

6.2.2.2.4.5 Electrical Power - The fuel cell reactant tanks are added to this configuration as reactants cannot be taken from the propellant system. The EPS design (Figure 6.2.2.2.4-2) is essentially the same as in section 6.1.2.2.4.5. Fuel cells are sized for 1.1KW, which includes a 20% design margin.

Figure 6.2.2.2.4-1 Block Diagram of the Space-Based Storable Configuration
Table 6.2.2.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 1 of 2)

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Equipment</th>
<th>Weight (lb)</th>
<th>Power (w)</th>
<th>Size (in)</th>
<th>Qty</th>
<th>Total Wt (lb)</th>
<th>Power (w)</th>
</tr>
</thead>
<tbody>
<tr>
<td>GN&amp;C</td>
<td>Star Tracker</td>
<td>11</td>
<td>10</td>
<td>7x 7x20</td>
<td>2</td>
<td>22</td>
<td>20</td>
</tr>
<tr>
<td></td>
<td>IMU</td>
<td>24</td>
<td>40</td>
<td>8x 8x12</td>
<td>2</td>
<td>48</td>
<td>80</td>
</tr>
<tr>
<td></td>
<td>GPS Receiver</td>
<td>20</td>
<td>30</td>
<td>8x 8x9</td>
<td>1</td>
<td>20</td>
<td>30</td>
</tr>
<tr>
<td></td>
<td>GPS Antenna-Low Alt</td>
<td>5</td>
<td>6x 6x10</td>
<td>2</td>
<td>10</td>
<td>10</td>
<td></td>
</tr>
<tr>
<td></td>
<td>GPS Antenna-Hi Alt</td>
<td>5</td>
<td>18x18x26</td>
<td>1</td>
<td>5</td>
<td>15</td>
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6-93
Figure 6.2.2.2.4-2  EPS configuration for the space-based, storable configuration.
6.2.2.2.5 Thermal Control (Space Based Storable) - All space-based storable configurations have fuel cell power systems. Separate H₂ and O₂ tanks must carry sufficient fuel to support a 2.10 kW continuous power requirements for the flight. Two 25-ft² radiators are required and should be mounted on opposite sides of the vehicle.

The avionics are housed in modularized boxes mounted to the avionics bay structure. This isolates many of the avionics components and makes more difficult to evenly distribute waste heat among the various avionics components. The design can, however, be passively thermally controlled. The high power avionics is mounted to base plates to allow a high heat conductance to the base plate. The base plate, in turn, has a strong radiation tie to space.

The payload/OTV interface is made nearly adiabatic by approximately 25 to 50 layers of insulation blanket (double aluminized Kapton MLI) located at the interface.

The N₂O₄ and MMH tanks are insulated with two layers of Kapton thermal blankets. These tanks are equipped with heaters (thermostatically controlled) to maintain acceptable propellant temperatures for long flight durations.

The impacts the meteoroid shield has on the OTV thermal control system will be evaluated as the OTV design develops.

The RCS tanks, feedlines, and modules require heaters since thermal pulsing would consume too much fuel. The RCS tank requires a 25 layer MLI blanket. The RCS feedlines will be insulated with one or two layers of thermally insulating blankets.

The heating effects of the engine nozzles are currently of no concern.

The composite helium pressurant tanks will require MLI. Thermostatically controlled strip heaters are necessary to ensure adequate helium temperatures/pressures.

6.2.2.3 System Weight Summary - Space-Based Storable

Total flight vehicle weight for the space-based storable configuration is presented in Tables 6.2.2.3-1 through -4. Table 6.2.2.3-1 presents data relative to the 53K lb perigee stage, while Table 6.2.2.3-2 presents data relative to the 90K lb perigee stage. Tables 6.2.2.3-3 and -4 present data relative to the 25.4K lb and 53K lb apogee stages, respectively. Dry weight is developed in detail. Propulsive and nonpropulsive fluid loadings are summarized. The breakdown has not been rearranged into MSFC's suggested format as this approach was deleted from contention at an earlier stage in the study.
Table 6.2.2.3-1 Weight Statement
Space-Based Storable
53K Propellant Load
Perigee Stage

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<th>Description</th>
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<tr>
<td>Basic Airframe</td>
<td>153</td>
</tr>
<tr>
<td>Ox Tank</td>
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</tr>
<tr>
<td>Tank (2) - Ti</td>
<td>141</td>
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<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
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<tr>
<td>Fu Tank</td>
<td></td>
</tr>
<tr>
<td>Tank (2) - Ti</td>
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</tr>
<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
<td>36</td>
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<td>Engine Truss &amp; Attach</td>
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<td>Attach - Engine Actuators</td>
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<tr>
<td>Subsystem Module</td>
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<tr>
<td>RMS Grapple Fixtures &amp; Struts</td>
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<tr>
<td>OMV I/F Fittings</td>
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<tr>
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<td>Crane I/F Fittings (2)</td>
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<tr>
<td>P/L or ACC Attach (4)</td>
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<td>Aerobrake Assy</td>
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<tr>
<td>Support (ASE) - S.B. Maintenance</td>
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<td>Environmental Control</td>
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</tr>
<tr>
<td>Meteoroid Protection</td>
<td></td>
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<tr>
<td>MPS Tanks</td>
<td>138</td>
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<tr>
<td>ACS Tanks</td>
<td>5</td>
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<tr>
<td>He Tanks</td>
<td>9</td>
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<tr>
<td>FC Reactant Tanks</td>
<td>3</td>
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<tr>
<td>FC Water Tank</td>
<td>2</td>
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<tr>
<td>Thermal Control/Protection</td>
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<tr>
<td>Thermal Control - Heater Tape @ MPS, ACS Tanks</td>
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Table 6.2.2.3-1 Weight Statement
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53K Propellant Load
Perigee Stage
(Continued)

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<td>Pressurization System</td>
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<td>ACS Common Feed</td>
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<tr>
<td>PU/Acquisition System</td>
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<td><strong>Instrumentation</strong></td>
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<td>Actuators (4) - Electrical</td>
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<td><strong>Orientation Control</strong></td>
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<td>Mounting - Tanks</td>
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<td>Propel. Distribution &amp; Pressurization</td>
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<td>Mounting Provisions</td>
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<tr>
<td>Fuel Cell System</td>
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<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
<td>42</td>
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<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
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<tr>
<td>Radiator System</td>
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<td>Residual H2O System</td>
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Table 6.2.2.3-1 Weight Statement
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53K Propellant Load
Perigee Stage
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Space-Based Storable
90K Propellant Load
Perigee Stage

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<td>Tank (2) - Ti</td>
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<tr>
<td><strong>Electrical Power</strong></td>
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<tr>
<td>Fuel Cell System</td>
<td></td>
</tr>
<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
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<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
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<tr>
<td>Radiator System</td>
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<tr>
<td>Residual H2O System</td>
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<td>Wire Harness, Connectors, etc.</td>
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<td>Mounting Provisions</td>
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Table 6.2.2.3-2 Weight Statement
Space-Based Storable
90K Propellant Load
Perigee Stage
(Continued)

<table>
<thead>
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<th>Weight (lb)</th>
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<tbody>
<tr>
<td>Avionics</td>
<td>425</td>
</tr>
<tr>
<td>Avionics (Equipment List)</td>
<td>373</td>
</tr>
<tr>
<td>Mounting Provisions</td>
<td>52</td>
</tr>
<tr>
<td>Dry Weight</td>
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</tr>
<tr>
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</tr>
<tr>
<td>Total Dry Weight</td>
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</tr>
<tr>
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</tr>
<tr>
<td>N204</td>
<td>60000</td>
</tr>
<tr>
<td>MMH</td>
<td>30000</td>
</tr>
<tr>
<td>Pressurant (He) - MPS</td>
<td>39</td>
</tr>
<tr>
<td>ACS Propellant &amp; Pressurant Scavenged from MPS</td>
<td>N/A</td>
</tr>
<tr>
<td>FC Reactant &amp; Coolant</td>
<td>22</td>
</tr>
<tr>
<td>Reactant</td>
<td>12</td>
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<tr>
<td>Coolant</td>
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Table 6.2.2.3-3 Weight Statement
Space-Based Storable
25.4K Propellant Load
Apogee Stage

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<td>Basic Airframe</td>
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<tr>
<td>Ox Tank</td>
<td>106</td>
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<tr>
<td>Tank (2) - Ti</td>
<td>68</td>
</tr>
<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
<td>38</td>
</tr>
<tr>
<td>Fu Tank</td>
<td>74</td>
</tr>
<tr>
<td>Tank (2) - Ti</td>
<td>38</td>
</tr>
<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
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</tr>
<tr>
<td>Engine Truss &amp; Attach</td>
<td>110</td>
</tr>
<tr>
<td>Engine Truss</td>
<td>76</td>
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<tr>
<td>Hardpoints - Engine Attach</td>
<td>30</td>
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<tr>
<td>Attach - Engine Actuators</td>
<td>4</td>
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<tr>
<td>Subsystem Module</td>
<td>208</td>
</tr>
<tr>
<td>RMS Grapple Fixtures &amp; Struts</td>
<td>56</td>
</tr>
<tr>
<td>OMV I/F Fittings (4)</td>
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</tr>
<tr>
<td>Cradle/Orbiter I/F Trunnions (5)</td>
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</tr>
<tr>
<td>Crane I/F Fittings (2)</td>
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</tr>
<tr>
<td>P/L or ACC Attach (4)</td>
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<tr>
<td>Aerobrake Assy</td>
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<td>Support (ASE) - S.B. Maintenance</td>
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<tr>
<td>Environmental Control</td>
<td>204</td>
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<tr>
<td>Meteoroid Protection</td>
<td>108</td>
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<td>MPS Tanks</td>
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<tr>
<td>ACS Tanks</td>
<td>5</td>
</tr>
<tr>
<td>He Tanks</td>
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<tr>
<td>FC Reactant Tanks</td>
<td>7</td>
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<tr>
<td>FC Water Tank</td>
<td>2</td>
</tr>
<tr>
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<td>96</td>
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<tr>
<td>Thermal Control - Heater Tape @ MPS,</td>
<td>59</td>
</tr>
<tr>
<td>ACS Tanks</td>
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Table 6.2.2.3-3 Weight Statement
Space-Based Storable
25.4K Propellant Load
Apogee Stage
(Continued)

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<td>Engine (2): XLR-132 7500# Thrust Ea</td>
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<tr>
<td>Propellant Distribution System</td>
<td>113</td>
</tr>
<tr>
<td>Pressurization System</td>
<td>202</td>
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<tr>
<td>ACS Common Feed</td>
<td>8</td>
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<tr>
<td>PU/Acquisition System</td>
<td>190</td>
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<tr>
<td>Instrumentation</td>
<td>15</td>
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<td>Actuators (4) - Electrical</td>
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<tr>
<td>Orientation Control</td>
<td></td>
</tr>
<tr>
<td>ACS Subsystem : Bi-Prop (MMH &amp; N204)</td>
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<tr>
<td>Rocket Engine Modules (2)</td>
<td>51</td>
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<tr>
<td>Mounting - REMS</td>
<td>5</td>
</tr>
<tr>
<td>ACS Accumulator Tanks (2)</td>
<td>28</td>
</tr>
<tr>
<td>Mounting - Tanks</td>
<td>3</td>
</tr>
<tr>
<td>Propel. Distribution &amp; Pressurization</td>
<td>82</td>
</tr>
<tr>
<td>Mounting Provisions</td>
<td>8</td>
</tr>
<tr>
<td>Electrical Power</td>
<td></td>
</tr>
<tr>
<td>Fuel Cell System</td>
<td>66</td>
</tr>
<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
<td>54</td>
</tr>
<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
<td>44</td>
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<tr>
<td>Radiator System</td>
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<td>Residual H20 System</td>
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Table 6.2.2.3-3 Weight Statement  
Space-Based Storable  
25.4K Propellant Load  
Apogee Stage  
(Continued)

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<tr>
<td>Dry Weight</td>
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<td>N204</td>
<td>16933</td>
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<tr>
<td>MMH</td>
<td>8467</td>
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<td>Pressurant (He) - MPS</td>
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<tr>
<td>ACS Propellant &amp; Pressurant Scavenged from MPS</td>
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<tr>
<td>FC Reactant &amp; Coolant</td>
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</tr>
<tr>
<td>Reactant</td>
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<tr>
<td>Coolant</td>
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Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage

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<td>Ox Tank</td>
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<td>Tank (2) - Ti</td>
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</tr>
<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
<td>38</td>
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<tr>
<td>Fu Tank</td>
<td></td>
</tr>
<tr>
<td>Tank (2) - Ti</td>
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</tr>
<tr>
<td>Aft Struts &amp; Fwd Fittings</td>
<td>36</td>
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<tr>
<td>Engine Truss &amp; Attach</td>
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<td>Engine Truss</td>
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<td>Hardpoints - Engine Attach</td>
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<tr>
<td>Attach - Engine Actuators</td>
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<tr>
<td>Subsystem Module</td>
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<tr>
<td>RMS Grapple Fixtures &amp; Struts</td>
<td>208</td>
</tr>
<tr>
<td>OMV I/F Fittings (4)</td>
<td>56</td>
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<tr>
<td>Cradle/Orbiter I/F Trunnions (5)</td>
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</tr>
<tr>
<td>Crane I/F Fittings (2)</td>
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<tr>
<td>P/L or ACC Attach (4)</td>
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<tr>
<td><strong>Aerobrake Assy</strong></td>
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<td><strong>Support (ASE) - S.B. Maintenance</strong></td>
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<tr>
<td><strong>Environmental Control</strong></td>
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<tr>
<td><strong>Meteoroid Protection</strong></td>
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<td>MPS Tanks</td>
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<td>ACS Tanks</td>
<td>5</td>
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<tr>
<td>He Tanks</td>
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<tr>
<td>FC Reactant Tanks</td>
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<tr>
<td>FC Water Tank</td>
<td>2</td>
</tr>
<tr>
<td><strong>Thermal Control/Protection</strong></td>
<td></td>
</tr>
<tr>
<td>Thermal Control - Heater Tape @ MPS, ACS Tanks</td>
<td>59</td>
</tr>
<tr>
<td>Thermal Protection - MLI, Tape &amp; Coatings @ Engine Truss, Compartment, Prop; Lines, etc.</td>
<td>96</td>
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<td></td>
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6-105
Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage
(Continued)

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<th>Description</th>
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<tbody>
<tr>
<td>Main Propulsion System</td>
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</tr>
<tr>
<td>Engine (2): XLR-132 7500# Thrust Ea</td>
<td>506</td>
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<tr>
<td>Propellant Distribution System</td>
<td>113</td>
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<td>Pressurization System</td>
<td>302</td>
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<tr>
<td>ACS Common Feed</td>
<td>8</td>
</tr>
<tr>
<td>PU/Acquisition System</td>
<td>190</td>
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<tr>
<td>Instrumentation</td>
<td>15</td>
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<td>Actuators (4) - Electrical</td>
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<tr>
<td>Orientation Control</td>
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<tr>
<td>ACS Subsystem - Hydrazine</td>
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<td>Rocket Engine Modules (2)</td>
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<td>Mounting - REMS</td>
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<tr>
<td>ACS Accumulator Tanks (2)</td>
<td>28</td>
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<td>Mounting - Tank</td>
<td>3</td>
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<tr>
<td>Propellant Distribution &amp; Pressurization</td>
<td>82</td>
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<tr>
<td>Mounting Provisions</td>
<td>8</td>
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<tr>
<td>Electrical Power</td>
<td></td>
</tr>
<tr>
<td>Fuel Cell System</td>
<td>66</td>
</tr>
<tr>
<td>Reactant Tank (LH2) &amp; Plumbing</td>
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<tr>
<td>Reactant Tank (LO2) &amp; Plumbing</td>
<td>49</td>
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<tr>
<td>Radiator System</td>
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<tr>
<td>Residual H2O System</td>
<td>13</td>
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<tr>
<td>Wire Harness, Connectors, &amp; etc.</td>
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6-106
Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage
(Continued)

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<td>Avionics (Equipment List)</td>
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</tr>
<tr>
<td>Mounting Provisions</td>
<td>52</td>
</tr>
<tr>
<td><strong>Dry Weight</strong></td>
<td>5009</td>
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<tr>
<td>Contingency (15%)</td>
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<tr>
<td>Total Dry Weight</td>
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</tr>
<tr>
<td><strong>Main Propellants (MR 2:1 Ox Wt to Fu Wt)</strong></td>
<td>53000</td>
</tr>
<tr>
<td>N204</td>
<td>35333</td>
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<tr>
<td>MMH</td>
<td>17667</td>
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<tr>
<td><strong>Pressurant (He) - MPS</strong></td>
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<tr>
<td>ACS Propellant &amp; Pressurant Scavenged from MPS</td>
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</tr>
<tr>
<td><strong>FC Reactant &amp; Coolant</strong></td>
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<td>H2</td>
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<td>O2</td>
<td>328</td>
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<td>Coolant</td>
<td>10</td>
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<td><strong>Total Loaded Weight</strong></td>
<td>59166</td>
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</table>
6.2.2.4 Performance on Model Missions: Space-Based Storables

The following is a summary of the assumptions used in performance analyses which were unique to storable OTV configurations. Several of the storable configurations were envisioned as perigee stages, requiring the use of an additional expendable kick stage to perform some part of the mission, such as circularizing the payload at geosynchronous altitude. In such cases, the EKS was assumed to have a mass fraction of 0.95, an Isp of 310 seconds, and was sized to be just large enough to perform the mission at hand, i.e. “custom fit” to each particular mission. For this reason the performance curves for the SB 53K, and SB 90K storable perigee stages are qualitatively different than those for the cryogenic stages which performed their missions “solo”.

Table 6.2.2.4-1 summarizes the propellant load required and gross weight of the 53K lb storable perigee stage on each of the Rev. 7 model missions on which it will be used in conjunction with an expendable apogee stage. Table 6.2.2.4-2 shows propellant and gross weight of the two stage configuration comprised of the 53K lb perigee stage and the 25K lb reusable apogee stage. Table 6.2.2.4-3 presents data analogous to Table 6.2.2.4-1 for the 90K lb perigee stage. Table 6.2.2.4-4 presents data equivalent to Table 6.2.2.4-2 for the 90K lb perigee stage/53K lb apogee stage combination. Figures 6.2.2.4-1 and-2 summarize the performance of the perigee stages to high circular orbits assuming the use of expendable apogee kick stages.

![Figure 6.2.2.4-1 Space-Based 53Klb Storable OTV Performance](image.png)

Figure 6.2.2.4-1 Space-Based 53Klb Storable OTV Performance
Table 6.2.2.4-1 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage, Used with Expendable Apogee Stage
(Rev. 7 Missions)

<table>
<thead>
<tr>
<th>Mission</th>
<th>P/L Up</th>
<th>P/L DN</th>
<th>OTV Propellant</th>
<th>Gross Wt.</th>
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<td>89268</td>
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<td>13660*</td>
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<td>46939</td>
<td>89268</td>
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<th>OTV Propellant</th>
<th>Gross Wt.</th>
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*DoD Equivalent Payload
Growth Projected by MMA

**Planetary program analyzed here was extracted from a preliminary Rev 8 planetary model supplied by MSFC on 25 Jan 1985 and not redone due to selection of cryo OTV
Table 6.2.2.4-2 Performance Analysis for Required Missions
Storable, Space-Based, 25K Apogee Stage, Used with 53K Perigee Stage
(Rev. 7 Missions)

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>GROSS WT.</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV + EKS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>+ PAYLOAD</td>
</tr>
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</table>

Geosynchronous Missions

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
<td>OTV + EKS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>+ PAYLOAD</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
<td>OTV + EKS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>+ PAYLOAD</td>
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<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
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<td>OTV + EKS</td>
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<tr>
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<td></td>
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<td>+ PAYLOAD</td>
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Table 6.2.2.4-3 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage, Used with Expendable Apogee Stage

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
<td>OTV + EKS</td>
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<td></td>
<td></td>
<td>+ PAYLOAD</td>
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Lunar Missions

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<thead>
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<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
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<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
<td>OTV + EKS</td>
</tr>
<tr>
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<td>+ PAYLOAD</td>
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Planetary Missions

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<thead>
<tr>
<th>MISSION</th>
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<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV PROPELLANT</td>
<td>OTV + EKS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>+ PAYLOAD</td>
<td></td>
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</tbody>
</table>

**Planetary program analyzed here was extracted from a preliminary Rev 8 planetary model supplied by MSFC on 25 Jan 1985 and not redone due to selection of cryo OTV**
Table 6.2.2.4-4 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage
(Rev. 7 Missions)

<table>
<thead>
<tr>
<th>MISSION</th>
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<th>OTV PROPELLANT</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>OTV + EKS</td>
<td>+ PAYLOAD</td>
<td></td>
</tr>
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</table>

Geosynchronous Missions

<table>
<thead>
<tr>
<th>MISSION</th>
<th>P/L UP</th>
<th>P/L DN</th>
<th>OTV PROPELLANT</th>
<th>OTV PROPELLANT</th>
<th>GROSS WT.</th>
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</thead>
<tbody>
<tr>
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<td>62879</td>
<td>34671</td>
<td>97550</td>
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<tr>
<td>15006</td>
<td>14000</td>
<td>14000</td>
<td>82354</td>
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<td>129144</td>
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Lunar Missions

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<th>P/L DN</th>
<th>OTV PROPELLANT</th>
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<th>GROSS WT.</th>
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<td>15000</td>
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<td>39066</td>
<td>196355</td>
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<tr>
<td>17204</td>
<td>80000</td>
<td>0</td>
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<td>80000</td>
<td>10000</td>
<td>154500</td>
<td>36317</td>
<td>190817</td>
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Figure 6.2.2.4-2 SB 90Klb Storable OTV Performance
6.3 MULTIPLE PAYLOAD CARRIER

Figure 6.3-1 shows an OTV multiple payload carrier mounted within the orbiter payload bay. The figure shows two PAM-D class and a IUS/INTELSAT class satellite payloads mounted on a centrally located ASE box frame assembly. The PAM-D satellites are each 7 feet in diameter, 9.25 feet long, and weigh 2030 pounds when intended for a cryogenic propellant OTV concept. For storable propellant OTV concepts, this weight increases to approximately 3904 pounds, accounting for the expendable apogee kick motors required. The corresponding weights of the IUS/INTELSAT class payload are 6600 pounds for the cryo OTV, 12,788 pounds (including apogee kick motor) for the storable OTV. The ASE box frame is approximately 8 feet long between satellites and is sized for the heavier loads at 3904 and 12,788 pounds. An additional ASE frame is added to support the opposite end of the heavy payload. Total overall length shown would be approximately 48 feet.

The payloads are each mounted on a conical payload attach fitting with an integral spin table and release mechanism. The spin tables are attached at the payload/OTV subframe which is shaded in the figure. Figure 6.3-2 and 6.3-3 show the configuration of the three payload multiple payload carrier with its payloads when installed on a ground-based cryogenic OTV. Figure 6.3-2 shows a side view, while Figure 6.3-3 is a section plan to showing the attachment of the payloads to the OTV. During the transfer from the orbiter payload bay to the OTV, the payload subframe is disconnected from the control ASE box frame and the forward ASE frame is disconnected from the heavy payload. The payloads are then transferred, while remaining attached to the payload subframe (shaded area), to the OTV attachment grid. The attachment grid has multiple attachment points which permits the payloads to be positioned with the required center of gravity.

Figure 6.3-4 shows a geomod generated view of the multiple payload carrier in the three spacecraft configurations. Figure 6.3-5 shows an exploded geomod generated view of a four PAM-D class configuration as required for installation in the orbiter payload bay. Two payloads are mounted on each of the payload subframes, which are in turn, attached to the central ASE box frame. The central ASE box frame remains in the orbiter payload bay after the payloads have been transferred to the OTV in LEO.

Table 6.3-1 gives the weight estimates for the multiple payload carrier in the four and three spacecraft configurations. These weight estimates are based on an aluminum central ASE frame (the structure that remains in the orbiter). A weight reduction can be realized by optimizing the design with the use composite materials. The subframe structure that flys with the OTV (where weight is more critical) uses composite materials in its design.
Figure 6.3-1 Multiple Payload Carrier and STS ASE

Figure 6.3-2 Multiple Payload Carrier and OTV (Side View)
Figure 6.3-3  Multiple Payload Carrier (End View)

Figure 6.3-4  Multiple Payload Carrier with 3 Spacecraft
Table 6.3-1  OTV - Multiple Payload Carrier (Aluminum)

<table>
<thead>
<tr>
<th>DESCRIPTION</th>
<th>PAYLOADS</th>
<th>(4) PAYLOADS</th>
<th>PAYLOADS</th>
<th>(3) PAYLOADS</th>
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<tbody>
<tr>
<td>AIRBORNE SUPPORT EQUIPMENT (ASE)</td>
<td></td>
<td>2305</td>
<td></td>
<td>633*</td>
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<td>FWD TRUSS</td>
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<td></td>
<td>1872*</td>
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<tr>
<td>AFT TRUSS</td>
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<td>66</td>
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<td>KEEL BEAM TRUNNION</td>
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<td>LONGERON TRUNNION</td>
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<td>20</td>
<td></td>
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<td>CONTINGENCY (15%)</td>
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<tr>
<td>PAYLOAD CARRIER - AIRBORNE TO GEO</td>
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<td>PAYLOAD/OTV SUBFRAME</td>
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<td>152</td>
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<td>HARD POINTS</td>
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<td>160</td>
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<tr>
<td>AVIONICS</td>
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</tr>
<tr>
<td>CONTINGENCY (15%)</td>
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<td>63</td>
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<tr>
<td>PAYLOAD DEPLOYMENT (SPIN TABLES)</td>
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<td>PAYLOADS</td>
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<td>(4) PAM-D/PAM-DII</td>
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<td>10710</td>
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<tr>
<td>(2) PAM-D PLUS (1) IUS/TOS/INTELSAT</td>
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<td></td>
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<tr>
<td>TOTAL CARRIER</td>
<td></td>
<td>12734</td>
<td></td>
<td>15728</td>
</tr>
</tbody>
</table>

* COMPOSITE MATERIALS WOULD SAVE APPROXIMATELY 468 LBS AND 634 LBS FOR THE (4) PAYLOAD AND (3) PAYLOAD CARRIER.
6.4 EVOLUTIONARY STRATEGY

This section presents a summary of the logic that went into our selection of the optimum OTV evolutionary strategy. The details backing up this logic are reported in Volumes III and VI. This summary encompasses the programmatic justification for OTV propellant selection, candidate evolutionary paths for the selected cryogenic approach, and program comparison data leading to our specific program recommendation.

The problem was approached in the broadest sense, showing both the benefit of undertaking the program as well as establishing the most desirable approach. A reusable space-based OTV functioning in conjunction with the Space Station was shown to be an important asset in the 1983 "Space Station Needs, Attributes, and Architectural Options Study". An important output of this study is a validation of this conclusion from the vehicle designers perspective. It is particularly important to prove that a significant advantage exists over the current expendable approach to high orbit access. The environment is competitive and the reason OTV is being considered for development is the attractiveness of reducing the cost of payload delivery, as well as providing a new roundtrip capability. If its advantage in the delivery mode is significant, it can justify earlier development of a ground-based capability and can make the STS more attractive to users, thus increasing self sufficiency. Of course any delivery cost advantage must be evaluated in the light of how rapidly it can pay back its development investment.

Any development recommendations are to be justified by the 'low' Revisions 8 OTV Missions Model, by MSFC direction. This mission model is only a projection of the OTV marketplace and should not be viewed as a fixed or absolute opportunity. In this light, the potential growth and flexibility of each option is important. An example of the desired flexibility is to be able to accommodate heavy payloads earlier than specified with little cost impact.

Risks attendant with OTV options and acquisition strategies are important because they reflect the possibility of increased cost. Key factors to be assessed are the risks that cannot be mitigated or controlled by the OTV design, such as STS delivery capacity.

The specific program evaluation factors that are important are as follows:

1) Return on investment
2) Cost per flight vs competition
3) Development cost
4) Payback
5) Risk
6) Growth/flexibility

We have evaluated each of these factors in our assessment of candidate evolutionary strategies.
6.4.1 Cryo/Storable Resolution

We carried the design activity for both cryogenic and storable OTV through the midterm review. A final decision between them was not made until a full operational and space-based accommodations assessment could be included in a full cost analysis. This trade reflects OTV programs that begin with ground based operation and transition to space based operation.

Table 6.4.1-1 summarizes the ground rules and assumptions that were used in conducting this cost analysis. The evaluation was originally conducted for the Rev. 7 'nominal' mission model, but was adjusted to reflect the 'low' Rev. 8 mission model. Figure 6.4.1-1 shows the cumulative discounted comparison of storable and cryogenic systems relative to an all expendable approach using the current/growth expendable upper stage stable (PAM, IUS and Centaur). The comparison shows the difference in program cost (in present value dollars) between the reference expendable program and the reusable program. This shows a payback for a storable investment slightly sooner, but the net advantage over the low model goes to the cryogenic approach.

Table 6.4.1-1 Storable vs Cryo OTV Ground Rules and Assumptions

1) All costs were calculated in 1985 dollars and exclude fees. Present Value (PV) comparisons reflected a 10% discount rate.
2) All cost estimates reflect midterm data (wt. mission model, etc.) generated for the cryogenic and storable stage families
3) DDT&E
   a) Maximum sharing of engineering & tooling efforts between stages was assumed where applicable.
   b) Ground test hardware includes STA, GVTA, MPTA and func. test article
   c) Dedicated flight tests required for the ground-based OTV: no space-based configuration flight test assumed
   d) Flight test articles refurbished to operations spares
   e) Space Station Equipment limited to tank farm impacts
4) Production
   a) Each unique stage assumes an initial production run of 2 units (1 operation, 1 spare (GVTA and FTA are refurbished for GB vehicles)
   b) 92% Wright learning curve assumed: Learning shared across stages
   c) Transportation charges for space-based production hardware included in production (68.5M/STS flt) (1.5 flts/full SB stage)
5) Operations
   a) Payload delivery costs assumed the same, transportation costs not included: No relights included
   b) Propellant usage based on 421 missions extracted from the midterm, nominal Rev. 7 mission model (32GB, 389SB), adjusted for Rev. 8 low model
   c) ETR launch only: STS CPF = $68.5M; ACC CPF = $2.3M
   d) Mission ops @ 35 man-yrs/yr
6) All cost estimates reflect midterm data (wt. mission model, etc.) generated for the cryogenic and storable stage families
Table 6.4.1-2 summarizes the comparison of investment and return on investment in addition to benefit. The ROI shown is calculated as (operational savings/DDT&E) -1. The storable approach requires less initial investment, but the cryo approach produces more benefit over the expendable fleet and a better return on investment for the low model. This advantage will increase for any more ambitious mission model. As a consequence, even though the cryo advantage is not large for the low model, we feel that cryogenics are clearly the correct selection. Our evolutionary analyses, therefore, were conducted for cryogenic families of Orbital Transfer Vehicles.

Table 6.4.1-2 Cryo/Storable Trade Results (Present Value)

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<th>Factors</th>
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<th>Cryo</th>
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<tr>
<td>Benefit ($M$)</td>
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<tr>
<td>Investment ($M$)</td>
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</table>

Scores

<p>| | | |</p>
<table>
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<td>ROI</td>
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<tr>
<td>Benefits</td>
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<td>10</td>
</tr>
<tr>
<td>Investment</td>
<td>10.0</td>
<td>9.1</td>
</tr>
</tbody>
</table>
6.4.2 Alternative Cryogenic Evolutionary Strategies

We evaluated five candidate evolutionary approaches to acquiring a space-based orbital transfer capability, as summarized in Figure 6.4.2-1. A totally ground-based program to achieve the same mission capability was also evaluated. These candidate programs provided the basis for investment/benefits comparisons, and led to our recommendation of the preferred OTV acquisition program. Three of the ultimately space-based programs shown in Figure 6.4.2-1 start with a ground-based OTV while two exist only in the space-based mode. Two of the initially ground-based programs start with an ACC configured OTV, while the third starts with a cargo bay configured OTV. The last parameter explored is whether the space-based OTV should also start unmanned and evolve to a man-rated system. We followed the philosophy that the program selected should be justifiable on the basis of the low Rev. 8 OTV mission model. The manned lunar sortie mission defined in the nominal mission model requires vehicle capability beyond the low model missions. The preferred means for achieving this capability is shown in the last column in Figure 6.4.2-1. These candidate programs were analyzed and are reported on in detail in Volume III of this report, and are summarized in section 6.4.3.

![Figure 6.4.2-1 Alternative OTV Growth Paths](image)

Figure 6.4.2-1 Alternative OTV Growth Paths
Figures 6.4.2-2 through 6.4.2-7 show a high level summary of the vehicles used in each of these program options. Figure 6.4.2-2 shows the two vehicles that comprise Option 1. The initial ground-based vehicle is the one described in detail in section 6.1.1, while the only other configuration is the initial space-based vehicle described in detail in section 6.1.2. Figure 6.4.2-3 shows the vehicles that comprise Option 2. This option adds an intermediate step between the vehicles used in Option 1. An initial non-man-rated vehicle using one main engine and with structure derived directly from the ground-based OTV is introduced. The potential advantage of this approach is lower development cost to become space-based. The full cost of achieving man-rating can be delayed. Option 4, as described in Figure 6.4.2-4, replaces the reusable ground-based ACC OTV with the expendable Centaur. Option 5, Figure 6.4.2-5, is the equivalent of option 2 with the expendable Centaur replacing the ground-based reusable ACC OTV. Option 6 is identical to Option 2 with the cargo bay OTV illustrated in Figure 6.4.2-6 replacing the ACC OTV. The final Option 7, Figure 6.4.2-7, is a totally ground-based approach. The initial step is the ground-based ACC OTV. The final Option 7, Figure 6.4.2-7, is a totally ground-based approach used in Options 1 and 2. It is followed with a non-man-rated version with propellant capacity increased to enable performance of the 20K delivery mission with the use of a second Shuttle flight. The final vehicle in this option is a man-rated version used only for the manned GEO missions.

![Option 1 - GB to Man-Rated SB Diagram](image-url)
AVIONICS: INTEGRAL
STRUCTURE: GRAPHITE EPOXY
AEROBRAKE: 40 FT
REDUNDANCY: NON-MAN RATED
PROP CAP: 45,000 Lb
LOADED WT: 50,363 Lb
ENGINE: 475 lsp/7500 Lb (1)

AVIONICS: RING
STRUCTURE: GRAPHITE EPOXY
AEROBRAKE: 40 FT
REDUNDANCY: NON-MAN RATED
PROP CAP: 52,500 Lb
LOADED WT: 58,282 Lb
ENGINE: 475 lsp/7500 Lb (1)

AVIONICS: RING
STRUCTURE: GRAPHITE EPOXY
AEROBRAKE: 44 FT
REDUNDANCY: MAN RATED
PROP CAP: 55,000 Lb
LOADED WT: 62,169 Lb
ENGINE: 475 lsp/7500 Lb (2)

GROUND BASED ACC DELIVERY
SPACE BASED ACC DELIVERY
SPACE BASED CB DELIVERY

Figure 6.4.2-3 Option 2 - GB to SB Followed by Man-Rating

AVIONICS: RING
STRUCTURE: GRAPHITE EPOXY
AEROBRAKE: 44 FT
REDUNDANCY: MAN RATED
PROP CAP: 55,000 Lb
LOADED WT: 62,169 Lb
ENGINE: 475 lsp/7500 Lb (2)

EXPENDABLE CB DELIVERY
SPACE BASED CB DELIVERY

Figure 6.4.2-4 Option 4 - Expendable to SB Man-Rated
EXPENDABLE CB DELIVERY

SPACE BASED ACC DELIVERY

SPACE BASED CB DELIVERY

Figure 6.4.2-5 Option 5 - Expendable to SB Followed by Man-Rating

GROUND BASED CARGO BAY OTV

VEHICLE DATA
CRYOGENIC PROPELLANT
TANK SIZE 48434 lbs
DRY WEIGHT 8642 lbs
LOADED WEIGHT 57076 lbs
A SE 5000 lbs
PAD WEIGHT 62076 lbs
SINGLE ENGINE THRUST 7500 lbs
ISP 475 sec
AVIONICS: SINGLE
FAULT TOLERANT

Figure 6.4.2-6 GB Cargo Bay OTV
The OTV program options described in the previous section were compared with a reference ‘competition’ program to develop benefit and return on investment parameters. That comparison is significant because it provides the reason for embarking on an OTV development in the near term. It is necessary to show an economic advantage for performing near term delivery missions since there are no near term missions that cannot be performed by existing vehicles. Near term capability requirements do not demand development of an OTV. All missions through 1998 in the low mission model can be delivered by existing expendable upper stages. Even after 1998 it is possible, but not likely cost effective, that heavier payloads as well as manned payloads could be captured by an existing or growth expendable upper stage. The competition for the reusable OTV options was constructed as follows.

The competition consisted of PAM D2, IUS, TOS/AMS, Centaur G’ and a growth version of Centaur. The growth Centaur had a 75 percent increase in propellant capacity and was presumed man-rated. Each mission in the ‘low’ Rev. 8 mission model was flown with the least expensive upper stage capable of supporting it. The total life cycle cost of this competitive program was estimated at $25,364M in 1985 dollars, $4,967M in discounted dollars. Using this array of expendable vehicles, 220 STS launches are required to perform the 145 missions in the low mission model. The resulting cost per STS flight is $120.8M in 1985 dollars, $23.7M in discounted dollars.
The first step in our program comparison was to compare Options 2 and 6. The purpose of this trade was to identify and select the preferred method of delivering the OTV to LEO for the 35 ground-based missions identified in the 'low' Rev. 8 OTV mission model. The results of this trade made the pivotal decision as to whether Option 2 or Option 6 would be traded against the remaining options delineated in Figure 6.4.2-1.

Ground-based delivery of OTV and scavenging of shuttle propellants both involve a selection between cargo bay and ACC. This correlation means it is necessary to evaluate both of these factors simultaneously. Table 6.4.3-1 summarizes the ground rules used for this trade study. They are consistent with the OTV ground rules provided by MSFC. The only addition to these standard requirements is the inclusion of an STS benefits factor. The ACC and cargo bay delivery of OTV for ground-basing have different benefits relative to providing additional payload volume and weight delivery capability. In the case of the ACC, it frees 30 feet of cargo bay for other payloads. The cargo bay concept, depending on length and weight, also makes it possible to manifest other payloads on an OTV mission. The inclusion of this benefit is justified since if a concept must pay for a particular development, it has the right to receive all direct and collateral benefits associated with that development.
Table 6.4.3-1 ACC vs Cargo Bay Trade Study Ground Rules and Assumptions

- General
  - Constant fiscal year 1985 dollars excluding fee and contingency
  - Discount rate of 10% per year
- R&T
  - Assumed $100M for AFE flight and $59M for advanced engine technology base for both candidates
- DDT&E
  - Ground test hardware includes STA, GBTA, MPTA and functional test article
  - Dedicated flight test required: Includes STS delivery, ACC and return charges as appropriate.
  - Flight test and GVTA articles refurbed to operational stages
  - GB ACC version includes ACC DDT&E ($163M); DB version includes $27M impact for orbiter bay modifications
  - Both options include DDT&E impacts for P/L clustering structure
- Production
  - Production for both options includes 2 P/L clustering structures 1) operations, 1 spare)
  - No stage production is required due to refurbishment of DDT&E hardware and low flight rates.
- Operations
  - All missions were manifested within the 72K lbs performance and 60' volume constraints of one STS flight
    - Included hardware dry weight, propellant, ASE
    - ACC Weight included for ACE version per study ground rule
  - STS user charge at $73M per flight (all missions exceeded 755 of orbiter performance): ACC CPF at $2.3M; KSC launch only
  - Low mission model (35 flights, 1994-1999)
  - Ground-based mission ops @ 34 M-yrs/yr
  - Minimum IVA charge due to P/L mating in ACC version (some missions exceed 24 hrs maximum, small IVA charge due to return flight assumed
  - IVA cost @$16K/hr
  - Aerobrake life = 1 flight
  - Engine life = 10 flights
  - Avionics, ECS str life = 40 flights
  - Ground refurbishment of stage based on a percentage of unit cost and analysis of current orbiter crew sizing
- Facilities
  - Facilities costs include
    - Provisions for manufacturing facility for initial stage and refurbishment hardware
    - Dedicated OTV launch processing facility (KSC)
    - Mission operations area at existing KSC facility
- Benefits
  - STS benefits were calculated based on 50% of weight and volume potential after OTV and P/W were manifested
Program cost for four scavenge and OTV delivery options were calculated. The detailed results of the calculations are presented in Volume III. Table 6.4.3-2 summarizes the results. The four options considered are:

1) Cargo Bay OTV Delivery/ACC Scavenging
2) Cargo Bay OTV Delivery/Cargo Bay Scavenging
3) ACC OTV Delivery/ACC Scavenging
4) ACC OTV Delivery/Cargo Bay Scavenging

It is clear that all combinations are viable solutions, but the ACC/ACC approach is far superior. Either ROI/Benefits or ROI/Investment as decision factors would result in choosing ACC/ACC. It is important to note that this conclusion is based on a relatively low STS flight rate. If a more optimistic rate is assumed, the scavenging benefits of the ACC scavenging concept would increase and thus make it even more attractive. These results and conclusions are sensitive to the assumptions concerning the ratio of scavenge flights to OTV flights. If scavenging was not a factor in the trade study, the cargo bay and ACC delivery ROI would be equal. As a consequence of these results, Option 6 was discarded.

The second step in our program comparison was to compare the remaining options delineated in Figure 6.4.2-1. Table 6.4.3-3 shows the ground rules and assumptions used in developing cost data for the economic evaluation of these options. Relatively high fidelity cost estimating was performed using the OTV WBS framework. Details of this estimation are presented in Volume III. The summary cost for each of the five evolutionary options, including interfacing systems, is shown in Tables 6.4.3-4 and 6.4.3-5 in constant dollars and in discounted dollars respectively. The interfacing systems costs -- Space Station, ACC, etc. -- are included as a ground rule requirement. The cost of payload delivery to LEO is also included by ground rule requirement, and adds $34.4M to the cost per flight of each option. The benefits shown include the STS collateral benefits that the ACC provides to each STS flight in terms of available volume and weight that can be used to deliver other cargos. This benefit reduces the cost of each OTV delivery by $8.6M per flight.

<table>
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<tr>
<th>ECONOMIC FACTOR</th>
<th>OPTIONS DEL/SCAV</th>
<th>CB/ACC</th>
<th>CB/CB</th>
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<td>7.7</td>
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<td>9.8</td>
<td>10</td>
<td>9.1</td>
<td>9.2</td>
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</table>
Table 6.4.3-3  OTV Evolutionary Program Trade:  
Ground Rules and Assumptions

- GENERAL

  - CONSTANT FISCAL YEAR 1985 DOLLARS EXCLUDING FEE AND CONTINGENCY
  - DISCOUNT RATE OF 10% PER YEAR ASSUMED: SPENDING CONSISTENT WITH IOC AND MISSION MODEL REQUIREMENTS

- R&T

  - ASSUMED $100M FOR AFE FLIGHT AND $59M FOR ADVANCED ENGINE TECHNOLOGY BASE FOR BOTH CANDIDATES

- DDT&E

  - GROUND TEST HARDWARE FOR INITIAL STAGE INCLUDES FULL STA, GVTA, MPTA AND FUNCTIONAL TEST ARTICLE: FOLLOW-ON STAGES INCLUDE GROUND TEST HARDWARE AS REQUIRED
  - DEDICATED FLIGHT TEST REQUIRED FOR INITIAL STAGE: INCLUDES DELIVERY AND PROPELLANTS
  - NO DEDICATED FLIGHT TEST FOR FOLLOW-ON STAGES
  - GVTA AND FLIGHT TEST ARTICLE OF INITIAL STAGE REFURBISHED TO MEET OPERATIONS REQUIREMENTS
  - MAXIMUM SHARING OF ENGINEERING AND TOOLING EFFORTS BETWEEN STAGES WAS ASSUMED WHERE APPLICABLE (EVOLUTIONARY APPROACH)
  - ALL OPTIONS INCLUDE DDT&E FOR P/L CLUSTERING STRUCTURE
  - SUPPORTING PROGRAM DDT&E (SPACE STATION ACCOMMODATIONS AND TANK FOR ACC AND PROPELLANT SCAVENGING WERE INCLUDED PER GROUND RULES AS APPLICABLE - COSTS WERE INCURRED CONSISTENT WITH BASELINE SCHEDULES AND IOC REQUIREMENTS

- PROVISIONS

  - EACH EVOLUTIONARY STAGE Requires TWO STAGES AT IOC (1 OPERATIONS UNIT, 1 SPARE)
    - REFURBISHED DDT&E HARDWARE CREDITED TO INITIAL OPTION STAGE
    - NO LEARNING ON STAGES ASSUMED DUE TO SMALL PRODUCTION RUN
    - EACH EVOLUTIONARY OPTION STAGE Requires 2 P/L CLUSTERING STRUCTURES (1 OPERATIONS UNIT, 1 SPARE)
    - TRANSPORTATION CHARGES OF PRODUCTION HARDWARE ALLOCATED TO OPERATIONS

- OPERATIONS

  - P/L TRANSPORTATION COSTS INCLUDED FOR ALL OPTIONS ACCORDING TO STS PROGRAM USER CHARGE GUIDELINES
    - 1994-1998 P/L's AND GB OTV STAGES WERE CONSIDERED AN INTEGRAL P/L UNIT AND CHARGED ACCORDINGLY
    - SPACE-BASED PAYLOADS (1999-2010) WERE CHARGED ACCORDING TO USER CHARGE GUIDELINES
    - OPTION 7 (GB EVOLUTIONARY OPTION) P/L's WERE CHARGED IN THE SAME MANNER AS 1999-2020 SB PAYLOADS (P/L GENERALLY CANNOT BE MANIFESTED ON THE SAME FLIGHT AS OTV HARDWARE)
OPERATIONS (CONTINUED)

- STS USER CHARGE OF 73M PER FLIGHT, ACC CHARGE OF 2.3M WHERE APPLICABLE
- LOW MISSION MODEL (145 FLIGHTS)
- GROUND-BASED MISSION OPS @ 35 M-YR/YR THROUGH OUT OPERATIONS PERIOD
- EXPENDABLE STAGES (OPTIONS 4 & 5, 1994-1998)
  - OPS COST INCLUDES STAGE CPF AND STS DELIVERY OF STAGE HARDWARE AND MISSION PAYLOAD
- GROUND-BASED OTV
  - OPERATIONS COST CONSISTENT WITH ACC - CB GB OTV TRADE STUDY
  - OPTION 7 (1999-2010) ASSUMES 1 SHUTTLE FLIGHT PER MISSION FOR OTV STAGE HARDWARE DELIVERY
- SPACE-BASED OTV
  - SPACE STATION IVA CALCULATED ON A PER MISSION BASIS @ $15K/HR
  - 2 OMV USES PER MISSION COSTS ACCORDING TO STUDY GROUND RULES @ 2 HOURS OUT, 1.5 HOURS BACK AND AVERAGE OF 500 LBS PROPELLANT PER MISSION
  - NO SPACE-BASED MISSION OPS OR EVA REQUIRED
  - STS COSTS INCLUDE DELIVERY OF INITIAL OPERATIONAL UNIT AND SPARES AS REQUIRED
  - ON-ORBIT PROPELLANT COSTS ARE THE COMPOSITE AVERAGE OF SCAVENGED AND STS TANKER DELIVERY COSTS, DETERMINED BY OPTION USAGE ($330 TO $360/LB)
- OPERATIONS SPARES
  - STS TRANSPORTATION APPLICABLE ONLY TO SB STAGES
  - AEROBRAKE LIFE = 5 FLIGHTS: 0.34 STS FLTS/BRAKE
  - ENGINE LIFE = 10 FLIGHTS: 0.1 STS FLT/ENGINE
  - AVIONICS, EPS, STR LIFE = 40 FLIGHTS: 1.0 STS FLT/REPLACEMENT

FACILITIES

- FACILITIES COSTS INCLUDE
  - PROVISION FOR MANUFACTURING FACILITY SPACE FOR INITIAL STAGE AND SPARES PRODUCTION
  - DEDICATED OTV LAUNCH PROCESSING FACILITY (KSC)
  - MISSION OPERATIONS SPACE AT EXISTING KSC FACILITY

- STS BENEFITS WERE CALCULATED BASED ON 50% OF WEIGHT AND VOLUME POTENTIAL AFTER GB OTV AND P/L WERE MANIFESTED
Table 6.4.3-4 Option Cost Summary - Constant Dollars

<table>
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<th>INTERFACING SYSTEM</th>
<th>OPTIONS</th>
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</thead>
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<td></td>
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<tr>
<td>GBU/SM/SM</td>
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<td>GBU/SBU/SM</td>
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<td>EXU/SBM/SM</td>
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Subtotal

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<td>R&amp;T</td>
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<tr>
<td>DDT&amp;E</td>
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<td>Prod.</td>
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<tr>
<td>OPS</td>
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<tr>
<td>Subtotal</td>
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</table>

TOTAL

14235.41  14094.11  16332.91  16176.21  19109.61

6-129
Table 6.4.3-5 Option Cost summary - Discounted Dollars

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<tr>
<td>GBU/MBM/MBM</td>
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<td>GBU/SBU/MBM</td>
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<td>EXU/MBM/MBM</td>
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<td>TOTAL</td>
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The final five evolutionary strategy options are shown with program costs vs cumulative missions in Figure 6.3.3-1. At OTV program start, Options 2, 5, and 7 immediately initiate major investments for OTV DDT&E while Options 4 and 5, which begin with expendables, have no initial investment required. At IOC as payback begins from initial flights, the all ground-based Option 7 shows the quickest return with payback at 25 flights. Options 1 and 2 show fast payback until the Space Station accommodations and propellant delivery costs reduce the payback and delay until 42 and 45 flights respectively. Option 2 cost benefits cross over the Option 7 ground-based curve at 66 flights and Option 1 at 71 flights. These curves show how ground-based vs space-based cost trades are impacted by the size and time phasing of the mission model. Options 4 and 5, both using expendables followed by space-based OTVs, delay the break even point to 72 flights for Option 5 and 74 for Option 4.

![Figure 6.3.3-1 OTV Evolutionary Strategy Comparison](image-url)
Table 6.4.3-6 shows the principle economic factors for the candidate options along with scoring. The scores are on a base 10. The best candidate is given a ten rating, and all other candidates are assigned proportional scores. No weighting of the factors was made (Investment, Benefits and ROI) however, the combination of ROI and Benefits are considered most important. ROI or Benefits to cost ratio measures the cost effectiveness of an option and Benefits measures the propensity for users to buy the OTV service over existing expendable upper stages. The ratio of any two options shows the relative leverage to attract business away from the expendable upper stages. The chart shows that the three viable candidates are Options 1, 2 and 7. These options have equal ROIs, but when we take ROI and Benefits, Options 1 and 2 are selected with essentially equal scores. Option 7 is attractive when ROI and Investment are considered. This option does carry with it considerable cost risk.

Option 7 is attractive only if the low investment cost, (DOT&E and Production), is real. It is tied directly to the STS user charges. Should the $73M cost to users not be achieved, the attractiveness of the option would be further eroded by a decrease in benefits. As an example, if the user charges were $100M instead of $73, Option 7's benefits would be reduced to $756M (discounted $) which would make the option economically unfeasible. That is, the investment would not be paid back within the 145 missions. Another aspect of Option 7 cost risk is its dependence on the lift capacity of the STS. With the ground ruled 72K lbs capacity, we found that 1.6 Shuttle flights per OTV mission was required. If only 65K lbs is achieved, the benefits over the competition would be reduced to $1625M (discounted $) and the ROI would reduce to 0.79. While still economically profitable, Option 7 would not be as attractive as option 1 and 2. Program considerations are also important reasons for eliminating the all ground--based option. The most important of these is freeing the STS to deliver revenue bearing cargos and to support the Space Station operations. Corollary to this is, if the number of missions for OTV increases, the burden on STS would be significantly increased. On the other hand, the availability of scavenged propellants is key to the cost effectiveness of space basing. If the cost of delivering propellants to the space station degraded to the $1123/pound associated with carrying them in the cargo bay, Option 7 would win over Option 1 by $44M (discounted $). At the present time, the propellant scavenging process is considered to be realistic. Option 1 or 2 is therefore selected over Option 7 because of projected benefits and lower cost risk.
Table 6.4.3-6 OTV Option Results

<table>
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<tr>
<th>DATA</th>
<th>1 GB/SBMR</th>
<th>2 GB/SB/SBMR</th>
<th>4 EXP/SBMR</th>
<th>5 EXP/SB/SBMR</th>
<th>7 GB/GB/GBMR</th>
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<td>INVESTMENT</td>
<td>1295</td>
<td>1301</td>
<td>920</td>
<td>921</td>
<td>907</td>
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</table>

The final issue is to select between Options 1 and 2. The major comparative characteristics of these options are:

1) Economics are equal
2) Option 1 maximizes early verification of man-rated reliability
3) Option 1 reduces Space Station operation complexity
   * Only two major program cycles
   * One robotic operation for assembly/disassembly
   * No loss of learning
4) Option 1 provides greater flexibility
   * Earlier heavy payloads capability
   * Earlier manned mission capability
   * Earlier Lunar Mission capability
5) Option 2 has higher cost risk
   * Three major program cycles
   * Avionics repackaging
   * One to two engine transition just prior to manned operation

Considering these differences, the compelling reasons for selection of Option 1 as the preferred OTV evolutionary concept are risk and flexibility. By starting the man-rated concept early, problems will be identified early and eliminated prior to the first manned missions. Starting with a more reliable vehicle will also reduce loss cost. Whenever a system goes through a series of block changes, there is always a cost risk, and DDT&E costs increase when design analyses must be redone. Load, thermal, FMEA, Vibration, Safety, and supporting tests are normally reported when significant changes in configuration are made. Flexibility to accommodate potential mission model changes is also important. By proceeding with a man-rated SBOTV, early heavy GEO missions or earlier manned missions to GEO or the moon could be readily accommodated. The latter would only require increased capacity propellant tanks. Our net conclusion is to recommend Option 1. This option is initially ground-based and transition to a man-rated configuration for space-based operations in 1999. The nature of this selected program is described further in the following sections.
6.5 DEVELOPMENT SCHEDULE

6.5.1 Space Transportation Architecture Summary - An OTV/Space Station/ACC Propellant delivery top level program schedule, shown in Figure 6.5-1, has been prepared to implement the Revision 8 low mission model. The ground-based OTV ATP is January 1998, with PDR in October 1988, and the CDR 9 months later in July 1989. The initial flight OTV is delivered during the third quarter of 1993 for initial flights in 1994. The space-based OTV is shown separately for clarity although an internal part of the same DDT&E program. The man-rated SBOTV begins the DDT&E phase in the first quarter of 1993 with PDR and CDR at 12 and 24 months respectively, leading to delivery in the late 3rd quarter, 1998 for Space Station based and unmanned payload flight in 1999. The main cryogenic rocket engine development would be initiated in 1989 to support the evolution from ground to space-based in 1999.

The dedicated aft cargo carrier ATP is in the first quarter of 1990 with an immediate PDR and CDR 12 months later. Delivery in the third quarter 1993 provides for mating with ET and GBOTV and STS flight in early 1994. Appropriate orbiter and KSC interfaces would be worked through the normal ET/STS/DSC integration organization and are not included in this schedule. The ACC would continue on a parallel schedule with the SBOTV while the propellant scavenging vehicle ATP would be delayed until 48 months prior to SB IOC on January 1995.

This schedule provides the GBOTV and ACC for initial flights in 1994 and SBOTV, ACC and propellant scavenging capability for 1999 SBOTV IOC.

The following four schedules (Figures 6.5.1-2 through 6.5.1-5) provide details for the acquisition of the ground and space-based OTV, the ACC and the scavenging system for the ACC.

The detail schedule for the recommended OTV concept and evolutionary strategy is included as Appendix B in Volume VI, Cost Estimates.

![Space Transportation Schedule Summary](image-url)

Figure 6.5.1-1 Space Transportation Schedule Summary
### Figure 6.5.1-2 Ground-Based OTV Development

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**Legend:**
- PDR: Preliminary Design Review
- CDR: Critical Design Review
- SUSTAINING
- IOC: Initial Operating Capability
| Year | 1 | 2 | 3 | 4 | 1 | 2 | 3 | 4 | 1 | 2 | 3 | 4 | 1 | 2 | 3 | 4 | 1 | 2 | 3 | 4 |
| 1993 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1994 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1995 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1996 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1997 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1998 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |
| 1999 |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |   |

- **ENGINEERING RELEASES**
  - MFG PLANNING/TDOLLING DESIGN
  - BID, FAB, IN STL & C/O
  - SPACE STATION HANGAR & FUEL ARM DESIGN
  - DEVELOPMENT & TEST: ORIENTATION CONTROL, ASE ASSY, AERO BRAKE, THERMO CONTROL, POWER, AVIONICS, SOFTWARE
  - FAB TEST ARTICLES
  - STA, GVTA, MFTA, FUNCTIONAL
  - CONDUCT TEST ON ARTICLES: STRUCTURAL, MODAL SURVEY, MAIN PROP., FUNCTIONAL
  - FAB & C/O DEVELOP. FLT. ART.
  - C/O & LAUNCH DEV. FLT. ART. (TO SPACE STATION)
  - FAB 1ST FLIGHT ARTICLE
  - C/O & LAUNCH 1ST FLT ARTICLE TO S/S
  - DESIGN & FAB SPACE HANGAR
  - DESIGN & FAB S/S FUEL FARM

Figure 6.5.1-3 Space-Based OTV Development
### MILESTONES

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- **ENGINEERING RELEASES**
- MFG PLANNING/TOOLING DESIGN & BID
- TOOLING FABRICATION INSTALLATION & CO
- FACILITIES FUNDS REQUIRED
- FACILITIES CRITERIA, A & E SELECTION
- FACILITIES DESIGN AND BID
- FACILITIES CONSTRUCTION ACTIVATION & C/O
- PROCUREMENT BID & LONG LEAD
- GROUND TEST ARTICLES FABRICATION
- GVTA & STA FAC DESIGN FAB & INSTALL
- STA SEPARATION, MODEL SURVEY, STATIC, & ACOUSTIC TEST
- MMC PRODUCTION BUILD FLIGHT ARTICLE
- ET PRODUCTION FLOW
- KSC PREPS C/O VERIFY & LAUNCH PROCESS
- KSC FACILITIES MODIFICATION

**Figure 6.5.1-4** Dedicated ACC Development

### PROPELLANT SCAVENGING VEHICLE

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</tbody>
</table>

- **DEVELOPMENT FLIGHT**
- **1ST OPERATIONAL FLIGHT**
- ENGINEERING DESIGN
- TOOLING/FACILITIES DESIGN, FAB, INSTL & C/O
- DEVELOPMENT TESTING
- REFINE CFM&E DESIGNS
- FAB GROUND TEST ARTICLES
- GROUND TESTING
- PROPULSION FIRING TEST
- FAB DEVELOPMENT FLT ARTICLE
- KSC C/O & LAUNCH DEV ARTICLE
- FAB 1ST OPERATIONAL ARTICLE
- UPGRADE GVTA TO FLT ARTICLE
- REFURB DEV ARTICLE TO FLT ARTICLE

**Figure 6.5.1-5** Propellant Scavenging System Development
6.5.2 Space-Based OTV Accommodations Time Phasing by Element - Determination of the space-based OTV accommodations element time phasing is actually not very complicated as accommodations must be available for use in quantum level jumps. The schedule planned is summarized in Figure 6.4.2-1. The propellant tank farm, the servicing and maintenance hangar with robotics, and the ground support elements must all be in place and operational by the time the space-based OTV is operational. A storage hangar or duplicate servicing and maintenance hangar, and enlargement of the original servicing and maintenance hangar (if necessary) must be in place and operational before the first scheduled 80K Lunar Delivery Mission.

Figure 6.5.2-1 Space-Based Accommodations Time Phasing
6.6 SUMMARY DDT&E AND PRODUCTION COST

6.6.1 Introduction - The evolutionary strategy evaluations resulted in the selection of an ACC ground-based OTV configuration transitioning to a man-rated space-based OTV on 1999. The principle characteristics of this selected option are shown in Figure 6.6.1-1. In this section provides a summary of the DDT&E and production costs.

![Diagram showing ground-based and space-based OTV configurations.]

**Figure 6.6.1-1 Selected Development Option**

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<tr>
<th>Ground Based ACC Delivery</th>
<th>Space Based CB Delivery</th>
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<tbody>
<tr>
<td><strong>AVIONICS:</strong> INTEGRAL</td>
<td><strong>AVIONICS:</strong> RING</td>
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<tr>
<td><strong>STRUCTURE:</strong> GRAPHITE EPOXY</td>
<td><strong>STRUCTURE:</strong> GRAPHITE EPOXY</td>
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<tr>
<td><strong>AEROBRAKE:</strong> 40 FT</td>
<td><strong>AEROBRAKE:</strong> 44 FT</td>
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<tr>
<td><strong>REDUNDANCY:</strong> NON-MAN RATED</td>
<td><strong>REDUNDANCY:</strong> MAN RATED</td>
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<tr>
<td><strong>PROP CAP:</strong> 45,000 Lb</td>
<td><strong>PROP CAP:</strong> 55,000 Lb</td>
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<tr>
<td><strong>LOADED WT:</strong> 50,363 Lb</td>
<td><strong>LOADED WT:</strong> 62,169 Lb</td>
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<tr>
<td><strong>ENGINE:</strong> 475 lsp/7500 Lb (1)</td>
<td><strong>ENGINES:</strong> 475 lsp/7500 Lb (2)</td>
</tr>
</tbody>
</table>
6.6.2 DDT&E - The premise for development of the DDT&E costs was that the design effort can be accomplished recognizing the planned block change and thereby minimize the cost of transition. Each subsystem's cost was approached on this basis. Table 6.6.2-1 shows a summary of the DDT&E cost broken down by main WBS element. It will be noted that the percentage change in OTV subsystem elements varies from about 1% to 38%. The structural change necessary to accommodate two engines and the avionics ring is the largest. The combination of avionics systems (GN&C, C&DH, electrical power and environmental control) only represents a 15% change. This is because the fundamental circuitry architecture is initially designed for the man-rated mission providing plug in redundancy capability. The major modification to this subsystem is only repackaging into a ring configuration from a structure integral design.

Table 6.6.2-1 shows the total contractor costs for the stage and the payload clustering structure to be $804.7 million to acquire the ground-based capability in 1994 with an additional $241.4 million to man-rate the OTV. Level II costs are those attributable to the NASA functions of management, integration and flight test. Facilities costs include acquisition of KSC and special factory facilities. The flight test costs are for a proto-flight of the ground-based OTV. No additional flight test of the man-rated configuration was deemed necessary since 35 flights of the ground-based OTV will provide sufficient confidence in the design.

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6-140
6.6.3  **Summary of Production Cost** - To minimize front end costs, the acquisition approach is based on refurbishing the ground vibration test article and the flight test article manufactured in the DDT&E phase. Table 6.6.3-1 shows a summary of the ground-based and space-based production costs. The total cost for the ground-based OTV and payload clustering structure is $29.9 million. If new hardware had been built instead of refurbishing test hardware, the cost of two units would have been $128.4 million.

Table 6.6.3-1 also provides the production cost for the space-based OTV at $115.4 million for two units. The low mission model traffic can be captured by one OTV at the Space Station with one back-up on the ground. All subsequent production to support operations has been charged as a part of operations costs.

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PROD TOTAL  | 29.9 | 115.4 | 145.3 |

6-141
7.0 REFERENCE DOCUMENTS & GLOSSARY

REFERENCE DOCUMENTS

1. ICD 80900000025; OTV/Dedicated ACC Interface; 8/1/83; Martin Marietta

2. JSC 07700 Vol XIV (Incl. Attachment 1); Space Shuttle System Payload Accommodations; Johnson Spaceflight Center; Latest Revision

3. Projected STS Lift Capability for Advanced Programs Planning Purposes; August, 1984; Johnson Spaceflight Center

4. TM 82478; Space and Planetary Environment Criteria Guidelines For Use in Space Vehicle Development; Volume 1; 1982 Revision; National Aeronautics and Space Administration

5. NASA TN D-5840; Deployment and Performance Characteristics of 5-foot Diameter Attached Inflatable Decelerators from Mach Number 2.2 to 4.4; H.L. Bohon and Z. Miserentino; August, 1970; NASA Langley Research Center

6. MMC TR-3709014; Viking Aerodynamics Data Book; B.F. Click; December 1970; Martin Marietta
GLOSSARY

The following acronyms and abbreviations are used in the text of this document and are listed here for convenience:

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<tr>
<th>Acronym</th>
<th>Definition</th>
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<td>Three Degree of Freedom</td>
</tr>
<tr>
<td>A</td>
<td>Area</td>
</tr>
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<td>ACC</td>
<td>Aft Cargo Carrier</td>
</tr>
<tr>
<td>ACS</td>
<td>Attitude Control System</td>
</tr>
<tr>
<td>ADAM</td>
<td>Aerobrake Deployment Assist Mechanism</td>
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<td>AFE</td>
<td>Aerasis Flight Experiment</td>
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<td>Air Force Rocket Propulsion Laboratory</td>
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<td>Apogee Kick Motor</td>
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<td>Al</td>
<td>Aluminum</td>
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<td>Authority to Proceed</td>
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<td>ASE</td>
<td>Airborne Support Equipment</td>
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<td>C3</td>
<td>Orbital Energy (km²/sec²)</td>
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<td>Drag Coefficient</td>
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<td>Center of Gravity</td>
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<td>CP</td>
<td>Center of Pressure</td>
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<tr>
<td>CPF</td>
<td>Cost per Flight</td>
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<tr>
<td>CMOS</td>
<td>Complementary Metal Oxide</td>
</tr>
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<td>Central Processor Unit</td>
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<td>CV</td>
<td>Cargo Vehicle</td>
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<tr>
<td>DACS</td>
<td>Data Acquisition and Control System</td>
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<td>Double Aluminized Kapton</td>
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<td>DD-250</td>
<td>Material Inspection and Receiving Report (Material Ownership Transfer from Contractor and Government)</td>
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<td>Design, Development, Test, and Stet</td>
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<td>DoD</td>
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<td>Dry-Tuned Inertial Reference Unit</td>
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<tr>
<td>Abbreviation</td>
<td>Full Form</td>
</tr>
<tr>
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</tr>
<tr>
<td>CN&amp;C</td>
<td>Guidance, Navigation and Control</td>
</tr>
<tr>
<td>CH₂</td>
<td>Gaseous Hydrogen</td>
</tr>
<tr>
<td>O₂</td>
<td>Gaseous Oxygen</td>
</tr>
<tr>
<td>GPS</td>
<td>Global Positioning System</td>
</tr>
<tr>
<td>GVTA</td>
<td>Ground Vibration Test Article</td>
</tr>
<tr>
<td>HEO</td>
<td>High Earth Orbit</td>
</tr>
<tr>
<td>I.D.</td>
<td>Inside Diameter</td>
</tr>
<tr>
<td>I/F</td>
<td>Interface</td>
</tr>
<tr>
<td>I/O</td>
<td>Input/Output</td>
</tr>
<tr>
<td>IMU</td>
<td>Inertial Measurement Unit</td>
</tr>
<tr>
<td>IOC</td>
<td>Initial Operational Capability</td>
</tr>
<tr>
<td>IR&amp;D</td>
<td>Independent Research &amp; Development</td>
</tr>
<tr>
<td>Isp</td>
<td>Specific Impulse</td>
</tr>
<tr>
<td>IRD</td>
<td>Interface Requirements Document</td>
</tr>
<tr>
<td>IUS</td>
<td>Inertial Upper Stage</td>
</tr>
<tr>
<td>IVA</td>
<td>Intra Vehicular Activity</td>
</tr>
<tr>
<td>JSC</td>
<td>Johnson Space Center</td>
</tr>
<tr>
<td>Klb</td>
<td>1000 Pounds</td>
</tr>
<tr>
<td>KSC</td>
<td>Kennedy Space Center</td>
</tr>
<tr>
<td>Kw</td>
<td>Kilowatt</td>
</tr>
<tr>
<td>L/D</td>
<td>Lift to Drag Ratio</td>
</tr>
<tr>
<td>L</td>
<td>Pound</td>
</tr>
<tr>
<td>LCC</td>
<td>Life Cycle Cost</td>
</tr>
<tr>
<td>LeRC</td>
<td>Lewis Research Center</td>
</tr>
<tr>
<td>LH₂</td>
<td>Liquid Hydrogen</td>
</tr>
<tr>
<td>Li</td>
<td>Lithium</td>
</tr>
<tr>
<td>LO₂</td>
<td>Liquid Oxygen</td>
</tr>
<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
</tr>
<tr>
<td>MAF</td>
<td>Michoud Assembly Facility</td>
</tr>
<tr>
<td>MLI</td>
<td>Multi-Layer Insulation</td>
</tr>
<tr>
<td>MMH</td>
<td>Mono Methyl Hydrazine</td>
</tr>
<tr>
<td>MMS</td>
<td>Multi-Mission Spacecraft</td>
</tr>
<tr>
<td>MPS</td>
<td>Main Propulsion System</td>
</tr>
<tr>
<td>MPTA</td>
<td>Main Propulsion Test Article</td>
</tr>
<tr>
<td>MR</td>
<td>Mixture Ratio (Ox to Fuel, by weight)</td>
</tr>
<tr>
<td>MSFC</td>
<td>Marshall Space Flight Center</td>
</tr>
<tr>
<td>MTBO</td>
<td>Mean Time</td>
</tr>
<tr>
<td>N₂O₄</td>
<td>Nitrogen Tetroxide</td>
</tr>
<tr>
<td>mm</td>
<td>Nautical Mile</td>
</tr>
<tr>
<td>NPSH</td>
<td>Net Positive System Head</td>
</tr>
<tr>
<td>OMS</td>
<td>Orbital Maneuvering System</td>
</tr>
<tr>
<td>OMV</td>
<td>Orbital Maneuvering Vehicle</td>
</tr>
<tr>
<td>ORU</td>
<td>Orbital Replacement Unit</td>
</tr>
<tr>
<td>OTV</td>
<td>Orbit Transfer Vehicle</td>
</tr>
<tr>
<td>P&amp;W</td>
<td>Pratt &amp; Whitney</td>
</tr>
<tr>
<td>P/L</td>
<td>Payload</td>
</tr>
<tr>
<td>PAM</td>
<td>Payload Assist Module</td>
</tr>
<tr>
<td>PCDA</td>
<td>Power Control &amp; Distribution Assembly</td>
</tr>
<tr>
<td>PDR</td>
<td>Preliminary Design Review</td>
</tr>
<tr>
<td>PHASE C/D</td>
<td>Development &amp; Operations Phases</td>
</tr>
<tr>
<td>PHI</td>
<td>Pump Head Idle</td>
</tr>
<tr>
<td>PIDA</td>
<td>Payload Installation Deployment Aid</td>
</tr>
<tr>
<td>PIP</td>
<td>Payload Integration Plan</td>
</tr>
<tr>
<td>PSF</td>
<td>Pounds/Ft²</td>
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</tbody>
</table>
psia  Pounds per Square Inch Absolute
PU  Propellant Utilization
PV  Present Value
q  Heating Rate (BTU/FT² Sec)
q  Dynamic Pressure (Lbs/FT²)
Q/D  Quick Disconnect
RAM  Random Access Memory
RCS  Reaction Control System
R/D  Research and Development
REM  Rocket Engine Module
RF  Radio Frequency
RFI  Radio Frequency Interference
RIG  Ring Laser Gyro
RMS  Remote Manipulator System
ROI  Return on Investment
ROM  Read Only Memory
RSI  Reusable Surface Insulation
RSS  Root Sum Square
RTV Sealer  Room Temperature Vulcanizing Sealer
S&M  Servicing and Maintenance
SB  Space Based
SBM  Space Based Manned
SBU  Space Based Unmanned
S/C  Spacecraft
SOFI  Spray on Foam Insulator
STA  Static Test Article
STS  Space Transportation System
T&C  Telemetry and Command
TCS  Thermal Control Subsystem
THI  Tank Head Idle
Ti  Titanium
TLM  Telemetry
TPA  Turbo-Pump Assembly
TPS  Thermal Protection System
TVS  Thermodynamic Vent System
VDC  Volts-Direct Current
W  Weight
WA  Watt of Aerobrake
WBS  Work Breakdown Structure
W Dry  Stage Dry Weight
WLS  Western Launch Site