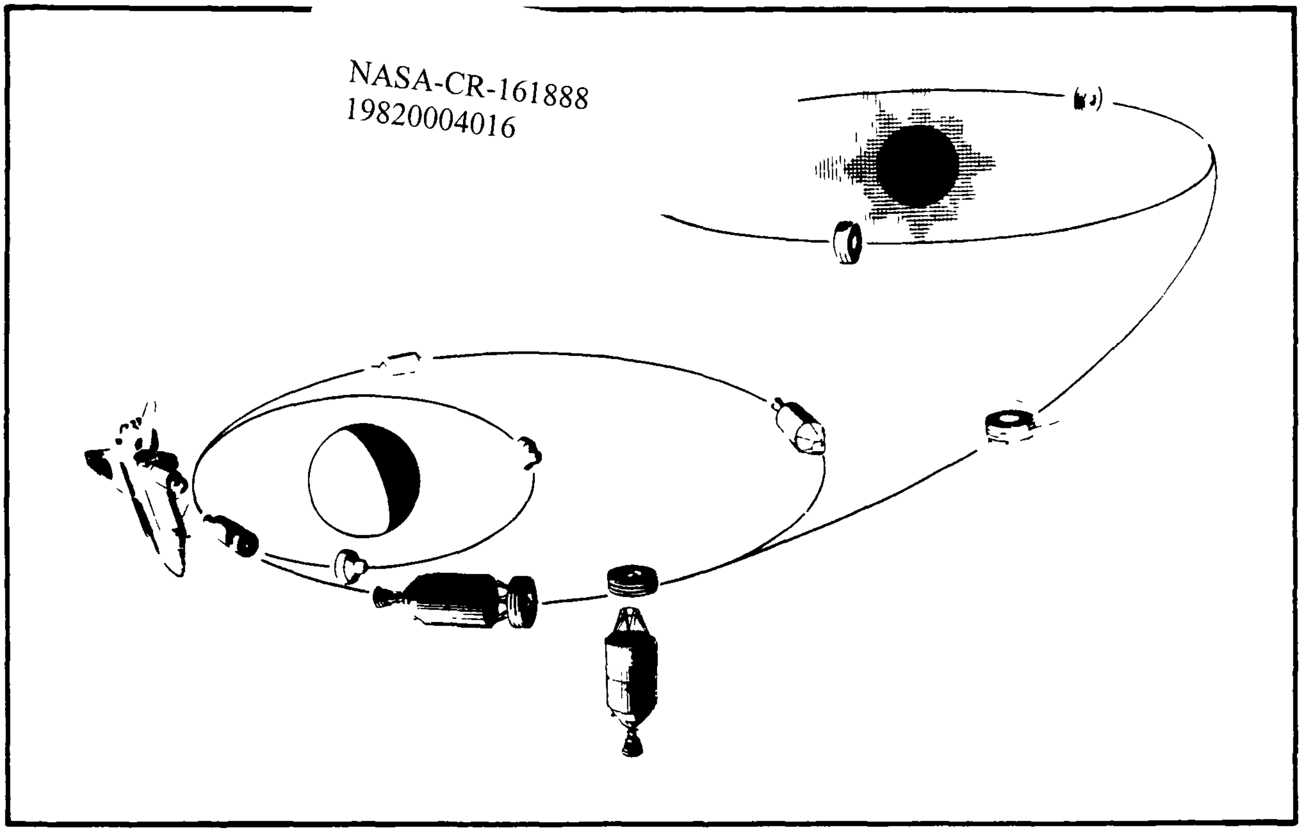
NASA CR-161,888

ANALYSIS OF SPACE SYSTEMS STUDY FOR THE SPACE DISPOSAL OF NUCLEAR WASTE



STUDY REPORT-VOLUME 2

TECHNICAL REPORT

1981

D180-26426-2

CONTRACT NAS8-33847
DPD 590
DR 4

THE
BOEING
COMPANY

ANALYSIS OF SPACE SYSTEMS STUDY FOR THE SPACE DISPOSAL OF NUCLEAR WASTE

STUDY REPORT
VOLUME 2

TECHNICAL REPORT

1981

D180-26426-2

Contract NAS8-33847

DPD 590

DR 4

Submitted to
The National Aeronautics and Space Administration
George C. Marshall Space Flight Center
By

Boeing Aerospace Company
Seattle, Washington 98124

N82-11889 #

FOREWORD

The study results described in this report are a part of an ongoing analysis to determine the feasibility and preferred approaches for disposal of selected high-level nuclear wastes in space. The Boeing Aerospace Company (BAC) study is an integral part of the ongoing Office of Nuclear Waste Isolation (ONWI) managed DOE/NASA program for study of nuclear waste disposal in space. The research effort reported here was performed by the Boeing Aerospace Company Upper Stages and Launch Vehicles Organization under NASA Contract NAS8-33847 from May of 1980 until March of 1981. The study objective was to identify, define and evaluate reasonable alternative concepts for the space disposal of nuclear waste, selecting alternative concepts of high merit for further evaluation, and documenting the evaluation and selection process.

The information developed during the study period is contained in this two-volume final report. The title of each volume is listed below:

Volume I Executive Summary
Volume II Technical Report

Inquiries regarding this study should be addressed to:

W. (Bill) Galloway
NASA/Marshall Space Flight Center
Attention: PS04
Huntsville, Alabama 35812
Telephone: (205) 453-2769

or

Richard P. Reinert, Study Manager
Boeing Aerospace Company
Mail Stop 8F-74
P. O. Box 3999
Seattle, WA 98124
Telephone (206) 773-4545

ACKNOWLEDGMENTS

The author wishes to acknowledge the generous assistance of the following individuals who provided significant contributions to the technical content of this report:

J. C. Jenkins:	o mission analysis o orbit transfer system concept development o cost and programmatics
D. Grim:	o electric propulsion
H. Whippo:	o propulsion system analysis
R. T. Conrad:	o structures and weights
C. L. Wilkinson:	o thermal analysis o aerothermodynamic analysis
H. Zeck:	o aerodynamics
R. J. Gewin:	o avionics o electrical power
A. R. Hathaway (BEC):*	o BEC study manager o radiation shield trades
J. L. Anderson (BEC):	o waste payload protection options and trade studies o waste payload configurations
M. P. Baze:	o radiation shield analysis
I. Arimura:	o radiation effects on avionics

Acknowledgment is also made to Mr. Claude C. (Pete) Priest, the NASA/MSFC contracting officer representative for the bulk of the study, for the outstanding guidance and cooperation provided; to his replacement, Mr. William Galloway, for valuable support during preparation of the final reports, and to various members of the MSFC staff, particularly Robert F. Nixon and Rowland E. Burns. Also, special thanks must go to Ralph E. Best, Manager of Alternative Concepts at the Department of Energy (DOE) Office of Nuclear Waste Isolation (ONWI), Columbus, Ohio, and Philip R. Compton, Energy Systems Division, Office of Aeronautics and Space Technology, NASA/Headquarters, Washington, D.C., for their guidance throughout the study.

* Boeing Engineering and Construction Company

TABLE OF CONTENTS

Page

Foreword ii
Acknowledgments iii

1.0 INTRODUCTION 1
1.1 Background 1
1.2 Objectives 3
1.3 Scope 3
1.4 Content 3

2.0 SPACE DISPOSAL DESTINATIONS 7
2.1 Introduction 7
2.2 Identification of Candidate Destinations 8
2.2.1 Identification of Destination Categories 8
2.2.2 Selection of Candidate Destinations 9
2.3 Define Candidate Destination Characteristics 13
2.3.1 Identify Key Characteristics 14
2.3.2 Characterize Candidate Destinations 17
2.4 Evaluation of Candidate Destinations 56
2.4.1 Evaluation of Long Term Risk of Earth Re-Encounter 56
2.4.2 Evaluation of Cost Related Factors 57
2.4.3 Evaluation of Other Factors 60
2.5 Candidate Destination Selection 60
2.6 Conclusions and Recommendations 62

3.0 MISSION ANALYSIS AND OPERATIONS 64
3.1 Introduction 64
3.2 Delivery Mission Analysis 64
3.2.1 Injection Mission Profile 64
3.2.2 OTV Recovery Option Trades 65
3.2.3 Placement Mission Profile 67
3.2.4 Solar Electric Stage Mission Profile 68

TABLE OF CONTENTS
Continued

	Page
3.3 Rescue Mission Analysis	69
3.3.1 Introduction	69
3.3.2 Definition of Abort Options	70
3.3.3 Rescue Mission Trajectories	71
3.3.4 Rescue Mission Performance Requirements	72
3.4 Operations Analysis	74
3.4.1 Introduction	74
3.4.2 System Level Composite Operations Flow	74
3.4.3 Detailed Definition of Operations	76
3.5 Mission Control	77
3.5.1 Introduction	77
3.5.2 SOIS Functional Requirements	77
3.5.3 SOIS Attitude External References	79
3.5.4 SOIS Attitude Control Concepts	79
3.5.5 Rescue Mission Control Requirements	80
4.0 ORBIT TRANSFER SYSTEMS	82
4.1 Introduction	82
4.2 Identify Orbit Transfer System Options	82
4.2.1 Assess Propulsion Systems for Application	82
4.2.2 Identification of Options	84
4.3 Define Candidate Orbit Transfer Systems	85
4.3.1 Characterize Candidate Vehicles	85
4.3.2 Candidate System Parametric Characterization	99
4.4. Select Candidate Orbit Transfer Systems	106
4.4.1 Parametric Performance Comparisons	106
4.2.2 Selected Option Definition	109
4.5 Rescue System Definition	110
4.5.1 Rescue System Requirements	110
4.5.2 Chemical Propellant Rescue System	112
4.5.3 Electric Propulsion Rescue System	117
4.5.4 Contingency Rescue Study Areas	117

TABLE OF CONTENTS

Continued

Page

5.0 LAUNCH VEHICLE SYSTEMS 118
5.1 Introduction 118
5.2 Candidate System Selection 118
5.2.1 Launch System Survey 118
5.2.2 Preliminary Launch System Screening 119
5.2.3 Candidate Launch System Description 120
5.3 Launch System Selection 121
5.3.1 Risk Assessment 121
5.3.2 Cost Assessment 122
5.4 Launch Vehicle Unique Requirements 125

6.0 WASTE PAYLOAD SYSTEMS 126
6.1 Introduction 126
6.2 Derivation of Protection System Requirements 127
6.2.1 Accident Effect Classification 127
6.2.2 Protection System Objectives 128
6.2.3 Accident End Conditions 129
6.3 Waste Payload Protection Trade Studies 130
6.3.1 Protection Options 130
6.3.2 Containment/Radiation Shielding Trades 130
6.3.3 Ejection From Launch Vehicle 140
6.3.4 Shield Removal Trades 141
6.3.5 Protection System Option Evaluation Conclusions. . 142
6.3.6 Thermal Analysis 143
6.4 Waste Payload Configuration Description 149
6.4.1 Integral Shield Configuration 149
6.4.2 Removable Shield Configuration 151
6.4.3 Dual Waste Payload 151
6.4.4 Waste Payload Weight Estimating Relationships . . 152
6.5 Waste Payload Protection System Definition 154
6.5.1 Aerodynamic Configuration Trades 155
6.5.2 Waste Payload Protective System Weight Estimating Relationships 158

TABLE OF CONTENTS
Continued

Page

7.0 FLIGHT SUPPORT SYSTEMS 160
7.1 Introduction 160
7.2 Flight Support System Functional Requirements 160
7.3 Launch Vehicle Flight Support Configurations 161
7.3.1 Flight Support System for Single Waste Payload . . 162
7.3.2 Flight Support System for Dual Waste Payload . . . 163

8.0 SYSTEMS INTEGRATION AND EVALUATION 166
8.1 Introduction 166
8.2 Systems Integration 167
8.2.1 Identification of Options 167
8.2.2 Selection of Options for Evaluation 168
8.3 System Performance Evaluation 169
8.3.1 Performance Evaluation Criteria 169
8.3.2 Option Throw Weight Evaluation 170
8.3.3 Option Cost Evaluation 170
8.3.4 Option Performance Evaluation 170
8.4 System Risk Evaluation 181
8.4.1 Risk Evaluation Criteria 182
8.4.2 Option Risk Evaluation 182
8.5 System Screening 183
8.5.1 Total System Evaluation Criteria 183
8.5.2 Space Systems Screening for Performance and Risk . 184
8.6 Definition of Selected Alternate Space Systems
Mission Scenarios 184
8.6.1 Chemical Propellant Systems 185
8.6.2 Electric Propulsion Systems 185

9.0 LAUNCH SITE SYSTEMS 190
9.1 Introduction 190
9.2 Launch Rates 190
9.3 Evaluation of Space Systems Support Facilities 191
9.4 Specialized Waste Payload Processing Facilities 192

TABLE OF CONTENTS

Continued

Page

9.5 Alternate Launch Site 192

9.6 Conclusions 192

10.0 CONCLUSIONS 193

11.0 RECOMMENDATIONS 195

APPENDICES:

A Acronyms and Abbreviations

B References and Bibliography

C Summary Characteristics of Surveyed Space Systems

D Reference System Description

E System Safety Design Requirements for the Reference System

F Orbit Transfer Vehicle Return Trajectory Trades

G Orbit Transfer System Detail Mass Statements

H Data for Radiation Shield Trades

I Flight Support System Detail Mass Statements

LIST OF FIGURES

		Page
2.2-1	Candidate Destination Categories	9
2.2-2	Candidate Destinations	10
2.3-1	Candidate Destination Characteristics Summary	18
2.4-1	Qualitative Cost Factor Determination	58
2.5-1	Candidate Destination Screening	61
3.2-1	Injection Mission Profile Schematic	65
3.2-2	Return Orbit Period Optimization	66
3.2-3	Placement Mission Profile Schematic	67
3.2-4	Solar Electric Stage Mission Profile Schematic	68
3.3-1	Rescue Mission Functional Categories	69
3.3-2	Analysis of Abort Options	71
3.3-3	HEO Rescue Trajectory Schematic	72
3.3-4	Deep Space Rescue Trajectory Schematic	73
3.4-1	Space System Composite Operations Flow	75
3.4-2	OTV/SOIS Separation Operations	76
3.5-1	SOIS Functional Requirements Summary	78
3.5-2	SOIS External Attitude References	79
3.5-3	Spin Stabilized SOIS Attitude Control Concept	80
3.5-4	3-Axis Stabilized SOIS Attitude Control Concept	81
4.2-1	Propulsion Systems Assessment	82
4.2-2	Identification of Candidate Orbit Transfer Systems for Space Disposal	83
4.3-1	Candidate Vehicles and Characteristics	87
4.3-2	Single Launch LOX/LH_2 Injection Stage Vehicle	89
4.3-3	Exploded View of LOX/LH_2 OTV	89

LIST OF FIGURES
(Continued)

Page

4.3-4 Avionics Block Diagram 91

4.3-5 LOX/LH_2 OTV Main Propulsion System Schematic 92

4.3-6 Dual Launch LOX/LH_2 Injection Stage Point Design 94

4.3-7 Long-Life OTV (LLOTV) LOX/LH_2 Injection Plus Placement Stage Configuration 94

4.3-8 LOX/LH_2 Placement Stage (Cryo SOIS) Configuration and Features 95

4.3-9 SES Technology Projections 97

4.3-10 SES Performance Summary 98

4.3-11 SES Point Design Configuration and Key Characteristics . . . 100

4.3-12 Injection Stage (OTV) Mass Trending Curves 101

4.3-13 Placement Stage (SOIS) Mass Trending Curves 102

4.3-14 Parametric Performance Characterization for Aerobraked Recoverable OTV with Storable SOIS 103

4.3-15 Parametric Performance Characterization for Aerobraked OTV with Cryogenic SOIS 104

4.3-16 Parametric Performance Characterization for LLOTV 105

4.4-1 Parametric Performance Comparison for 2-Stage Systems . . . 107

4.4-2 Parametric Performance Comparison of Staging Options 108

4.4-3 Selected Orbit Transfer Systems 109

4.5-1 Rescue System Requirements 111

4.5-2 Rescue Kit Configuration 112

4.5-3 Rescue Vehicle Modified from Standard SOIS 113

4.5-4 Rescue Vehicle Configuration and Summary Mass Statement . . 114

4.5-5 Rescue Orbit Transfer System Pursuit Configuration 115

4.5-6 Rescue Orbit Transfer System: Injection Configuration . . . 116

5.2-1 Candidate Launch Systems 119

LIST OF FIGURES
(Continued)

		Page
5.2-2	Candidate Launch Vehicle Characteristics	120
5.3-1	Launch System Risk Characteristics	122
5.3-2	Space Disposal of Nuclear Waste Launch Requirements	123
5.3-3	Life Cycle Cost Comparison for Candidate Launch Systems	123
5.3-4	Launch System Life Cycle Cost Sensitivity Studies	124
5.4-1	Unique Requirements for Launch Vehicles	125
6.2-1	Space Disposal Accident Classification	128
6.2-2	Accident End Conditions	129
6.2-3	Recommended Design Criteria for Impact Angle and Speed	130
6.3-1	Waste Paylod Protection Options	131
6.3-2	Waste Payload Packaging Arrangement Trade	132
6.3-3	Waste Payload Shape Trade	132
6.3-4	Candidate Shield Materials and Properties	133
6.3-5	Candidate Radiation Shield Comparison	138
6.3-6	Shield Removal Factors	141
6.3-7	Manifesting Mismatch Decreases the Cost Advantage of Shield Removal	142
6.3-8	Unshielded Waste Sphere Radius Limit at 0.85 AU	144
6.3-9	Removable and Integral Shield Temperature Profiles for LEO and 0.85 AU	145
6.3-10	Removable and Integral Shield Temperature Profiles for Free Convection Cooling in 100^{o}F Air at Sea Level	147
6.3-11	Integral Shield Temperature History After Post-Entry Burial	148
6.4-1	Candidate Waste Payload Configurations	150
6.4-2	Dual Waste Payload Configurations	151
6.4-3	Waste Payload Characteristics for Alternate Waste Form Sizes	153

LIST OF FIGURES
(Continued)

Page

6.4-4 Waste Payload Weight Estimating Relationships for Selected Shield Options 154

6.5-1 Drag Coefficient vs. Mach No. for Candidate Shapes. 156

6.5-2 Hypersonic Aft C. G. Limit/Body Length Trade 156

6.5-3 Hypersonic Pitching Moment/Aft Body Angle Trade 157

6.5-4 Waste Payload Protection System Configuration and Envelope Dimensions 158

6.5-5 Waste Payload Protection System Elements 159

6.5-6 Waste Payload Protection System Parametric Mass Characterization 159

7.2-1 Flight Support System Functional Requirements 161

7.3-1 Waste Payload System Options Requiring Flight Support 162

7.3-2 Flight Support System Configuration for Single Waste Payloads 163

7.3-3 Flight Support System Configuration for Dual Waste Payloads 164

7.3-4 Dual Waste Payload Flight Support System Operation Sequence 165

8.2-1 Identification of Options 168

8.2-2 Selection of Options for Evaluation 169

8.3-1 Space System Performance Assessment: Option DL-1 172

8.3-2 Space System Performance Assessment: Option DL-2 172

8.3-3 Space System Performance Assessment: Option SL-1 173

8.3-4 Space System Performance Assessment: Option SL-2 174

8.3-5 Space System Performance Assessment: Option SL-3 175

8.3-6 Space System Performance Assessment: Option SL-4 175

8.3-7 Space System Performance Assessment: Option SL-5 176

8.3-8 Space System Performance Assessment: Option SL-6 177

8.3-9 Space System Performance Assessment: Option SL-7 178

LIST OF FIGURES
(Continued)

Page

8.3-10 Space System Performance Assessment: Option SL-8 178

8.3-11 Space System Performance Assessment: Option SL-9 179

8.3-12 Space System Performance Assessment: Option SL-10 180

8.3-13 Comparison of Total System Performance 181

8.4-1 Qualitative Risk Ranking 182

8.5-1 Space System Screening 184

8.6-1 Space System Composite Operations Flow for Option DL-2 . . . 186

8.6-2 Orbital Operations for Option DL-2 187

8.6-3 Space System Composite Operations Flow for Option SL-10 . . . 188

8.6-4 Orbital Operations for Option SL-10 189

9.2-1 Launch Rates for Selected Options 191

9.3-1 Facilities Impact for Selected Options 192

1.0 INTRODUCTION

Since 1970 a number of concepts for space systems for nuclear waste disposal have been studied and evaluated. This study has integrated the results of these previous studies in a systematic fashion to identify and document viable alternative space system concepts having high merit. This effort is an integral part of the ongoing NASA/DOE program for evaluation of the space option for disposal of certain high level nuclear wastes in space as a complement of mined geologic respositories. This introduction provides a brief overview of the study background, scope, objective, and contents.

1.1 BACKGROUND

The need to isolate safely or dispose of nuclear waste materials is a problem for this nation and other nations of the world. This problem has been studied for many years, but comprehensive studies of space as a disposal site did not begin until early 1970's when the NASA Lewis Research Center (LeRC) began studies which included comprehensive design analyses and concept testing of a nuclear waste payload that could survive Earth atmospheric reentry and impact from space. The payload was to be launched into low Earth orbit by the Space Shuttle and then delivered to solar system escape by a space tug. The major problem with this concept was that it provided for a relatively small payload mass. This was due to three reasons: (1) the aerodynamic and radiation shield was carried all the way to the space destination, (2) waste loading was low, and (3) the space destination was solar system escape, a high energy option.

In 1975, the NASA's Marshall Space Flight Center (MSFC) initiated a further study of the disposal in space of nuclear waste. During this study, it was concluded that aerodynamic and radiation shielding could be carried only to low Earth orbit and that, beyond orbit, significant safety precautions for operational failures, including space rescue, were necessary to make this space option viable. In 1978, a concept was developed from these studies that was essentially an extension of the earlier LeRC and MSFC concepts. This concept involved: using the Space Shuttle to launch an aerodynamic and radiation shielded waste payload to a low Earth orbit; removal of the shielding; and use of an Orbit Transfer Vehicle (OTV) - formerly called a space tug - to inject the payload (without aerodynamic or radiation shielding) to a space destination transfer trajectory. A variety of destinations were treated. In 1976, the NASA Ames Research Center (ARC) initiated studies to investigate the long-term safety of space disposal of hazardous material. These analyses led to the conclusion that long-term (millions of years) stability of several lower energy space destinations was possible, leading to selection of two potential sites for further investigation: (1) a solar orbit at 0.85 AU and (2) the lunar surface. A considerable amount of data on other destinations was also developed.

During 1978, the MSFC study activities addressed the preliminary evaluation of the space disposal of defense nuclear waste from three defense waste storage sites. The analyses included the development of a reference concept employing the solar orbit at 0.85 AU as the space destination, the safety of the space disposal approach, and the environmental impact of selected credible accidents. The study concluded that space disposal of defense waste at the solar orbit destination was feasible.

The 1979-80 study program conducted by MSFC and the Battelle-Columbus Laboratories has continued the development of space option concept definitions; tradeoffs among transportation options, waste container, shielding, and reentry protection; transportation cost analysis; further characterization of the defense and commercial waste; and safety, risk, and environmental analyses.

These efforts provided the bases for the preparation of the comprehensive system safety design requirements summarized in Appendix E. These requirements, in combination with the data on space disposal destinations provided by the earlier AMES and MSFC contracted studies, have provided for the first time a systematic and comprehensive set of requirements for evaluation and design of space systems for the disposal of nuclear waste. These requirements, together with the extensive data base on space systems provided by the MSFC 79-80 study efforts, have provided the basis for this study.

1.2 OBJECTIVES

The overall objective of the NASA/DOE space system study activity is to select a baseline space system concept which shows positive risk benefits when used as a complement to the reference Mined Geologic Repositor (MGR).

Specific objectives of this study included (1) identification and definition of space systems concepts, (2) documentation of the definition of each concept, (3) aiding the ONWI/NASA program management effort in the development of space system concept selection criteria, (4) evaluation of these concepts as to their performance, risk, and technical viability, (5) selection of the most attractive space system concepts for further consideration, (6) documentation of the evaluation and selection process, and (7) providing of appropriate engineering, reliability, environmental impact and cost information to other program tasks as a part of the integrated NASA/DOE program.

1.3 SCOPE

This study covers the systematic identification, definition and evaluation of reasonable space system concepts leading to the integration and evaluation of total system concepts for space disposal of nuclear waste. Specific study areas included space destinations, space transportation options, launch site options, nuclear waste payload protection approaches, and payload rescue techniques. Maximum use was made of the previous studies and assessments of the space disposal of nuclear waste. Definition of the space system concepts was on a common basis, consistent with the study groundrules. Additional analyses and definition were performed only on space system concepts not thoroughly assessed in previous studies. Total system concepts resulting were evaluated for performance and risk, leading to selection of four concepts of high merit for further evaluation.

1.4 CONTENT

The following paragraphs briefly describe the contents of this volume (Volume 2). Volume 1 serves as the Executive Summary for Volume 2.

Section 2 summarizes the study effort on space disposal destinations. Primary topics include an initial screening of all reasonable destinations which resulted in the identification of 14 for definition; definition of the characteristics of the selected destinations; and the final screening and evaluation which selected a reference destination (0.85 AU heliocentric orbit) for the design and evaluation of alternative space systems.

Section 3 reports the work accomplished on mission analysis, operations and mission control. Delivery mission profiles are presented for both chemical and electric propulsion stages in Section 3.2, along with the analyses conducted in support of mission definition. In Section 3.3, Rescue Mission Analysis, rescue mission abort options are discussed, and rescue mission profiles for both Earth orbit and deep space transfer system failures are discussed. Section 3.4, Operations Analysis, describes a generic operations flow diagram which is shown along with the more detailed breakdown of operations for injection/placement stage separation derived to aid mission analysis. Section 3.5, Mission Control, describes general mission control issues and presents detailed definition of SOIS control requirements used for definition of SOIS avionics in Section 4.

Section 4 documents the study of orbit transfer systems which resulted in selecting four options for integration of total system options in Task 8. Section 4.2 describes the identification of orbit transfer options, which identified a total of 12 systems for further consideration. Section 4.3 describes the detailed characterization of the elements (stages) and the total orbit transfer systems. The selection of the final candidates is treated in Section 4.4, while Section 4.5 describes rescue system requirements and configurations for both chemical and electric orbit transfer systems. A final section identifies contingency rescue mission study areas.

Section 5 describes the evaluation and selection of launch vehicle systems for the space disposal mission. Section 5.2 describes the initial screening of candidate launch vehicle categories, which resulted in the selection of four candidate systems for further evaluation, and describes the key characteristics of the vehicles selected. Section 5.3 describes the selection criteria developed and their use in selection of launch system options for single launch and dual launch mission scenarios. Unique requirements imposed on launch systems by the space disposal mission are described in Section 5.4.

Section 6, Waste Payload Systems, reports the results of trade studies conducted to determine the best approach to protection of the waste payload for all mission phases. Section 6.2 discusses the use of past studies in definition of waste payload system requirements. Trade studies which evaluated the utility of radiation shield hardening, ejection from the launch vehicle, and shield removal as protection options are described in Section 6.3, along with the results of waste payload thermal and aerothermal analyses.

The resulting waste payload configurations and parametric weight estimating relationships are reported in Section 6.4. Section 6.5 describes the aereodynamic configuration studies used in definition of the ejectable waste payload protection system (WPPS). A description of the resulting configuration and derivation of WPPS weight estimating relationships complete the section.

Section 7, Flight Support Systems, describes the requirements and configurations of flight support systems used to support waste payloads in the launch vehicle. Section 7.2 describes the functional requirements identified for flight support systems. Configurations evolved to meet these requirements for both single and dual waste payloads are described in Section 7.3.

Section 8, Systems Integration and Evaluation, describes the integration, evaluation and selection of alternative total system concepts for space disposal judged to have sufficient merit for further study. Section 8.2 describes the systems integration effort resulting in selection of 12 candidate concepts for evaluation. Key characteristics and performance of the candidate systems are reported in Section 8.3. Section 8.4 describes the total system risk criteria and the evaluation of the relative risk of the 12 candidates. Final screening and selection of the four alternative systems of high merit are described in Section 8.5; Section 8.6 describes mission scenarios for the selected systems.

Section 9, Launch Site Systems, summarizes the assessment of impact of nuclear waste disposal mission on the KSC launch site. A brief assessment of an alternate launch site was also conducted. Section 9.2 summarizes the required launch rates for the selected total system options. Sections 9.3 and 9.4 describe the evaluation of space system support facilities and specialized waste payload processing facilities, respectively. Section 9.5 discusses the alternate launch site evaluation conducted.

Sections 10 and 11 present the primary conclusions and recommendations that have resulted from the study.

Appendix A provides definitions for the acronyms and abbreviations used in this volume; Appendix B contains a list of the references indicated in the text and a bibliography. Appendix C presents summary evaluations of the space disposal concepts from past studies evaluated in Task 8. Appendix D provides a description of the reference system for space disposal resulting from the MSFC 79-80 study; Appendix E describes the system safety design requirements for the reference system. Appendixes F through I support various aspects of the analyses performed in Sections 3, 4, 6, and 7.

2.0 SPACE DISPOSAL DESTINATIONS

2.1 INTRODUCTION

This chapter presents the results of an investigation of locations in space, or destinations, where nuclear waste payloads could be deposited for permanent isolation from the terrestrial biosphere. The investigation was aimed at defining a single distination to be used in definition of the space transportation systems required for the space disposal mission. Because of its pivotal role in setting space system performance requirements, this task was performed at the beginning of the study.

The primary issue was to determine if there was a safer or cheaper destination than the circular heliocentric orbit at 0.85 AU chosen as a reference destination in the 79-80 study effort.

An initial screening which encompassed every reasonable destination in the solar system as well as solar system escape orbits resulted in the selection of 14 candidate destinations for evaluation in depth. The candidate destinations included orbits about the Sun, Moon and Earth and the surfaces of planets, asteroids, and the Moon. The selected candidate destinations were evaluated for cost related factors primarily based on performance, risk related factors including potential for return of waste material to the terrestrial biosphere, and other factors including implications for future use and retrievability.

A final screening was conducted in which each candidate destination was ranked in terms of these three criteria. Candidates with costs and risks higher than those of the reference helicentric orbit destination were rejected for further consideration as were those which evidenced a problem with contamination of destinations which could compromise potential future usage.

Six candidate destinations satisfied all three criteria for selection: Both posigrade and retrograde high Earth orbits (HEO), orbits about the Earth-Moon Trojan points, lunar orbits, and heliocentric circular orbits at both 0.85 and 1.15 AU radius. From these six, the 0.85 AU radius heliocentric orbit was selected as a reference for the study. It offers a combination of the lowest long term risk of any destination studied and excellent characterization from previous studies not shared by the other selected destinations.

A high Earth orbit destination was recommended for evaluation as an alternate to the heliocentric orbit reference. A full evaluation of long term risk for this class of destination is not available, but potential risk could be comparable to the reference and some attractive cost benefits would result from factors such as lower performance requirements for orbit transfer systems. When evaluated, these cost benefits would indicate the potential payoff which could justify a later in-depth investigation of the concept's long term risk.

2.2 IDENTIFICATION OF CANDIDATE DESTINATIONS

The objective of the identification effort was to systematically survey the entire field of potential space disposal destinations, making maximum use of past studies but also identifying locations not previously treated. No locations were categorically ruled out for the first level of screening. Candidate destinations were identified in two steps: in the first, general destination categories were identified by reference to past studies; in the second step, individual "best" destinations were selected within each category.

2.2.1 Identification of Destination Categories. Categorization of destinations was accomplished to make sure no possibilities were overlooked. Broad distinctions were drawn between surface destinations (where the waste would be placed on a solid body such as a planet, asteroid or moon) and orbital destinations and between destinations in the geolunar (Earth-Moon) system and destinations in deep space.

Candidate destination categories identified are illustrated in Figure 2.2-1 and are tabulated as surface or orbit types with locations in the geolunar system or in deep space. The categories shown contain all reasonable destinations for space disposal of nuclear waste.

Surface Destination Categories. Surface destinations in the geolunar system are limited to Luna and the Earth. Ruling out Earth as a disposal site not directly relevant to studies of space disposal leaves Luna as the only surface

LOCATION \ TYPE	SURFACE	ORBITS
GEOLUNAR	• LUNA	HEO POSIGRADE HEO RETROGRADE LUNAR ORBITS LIBRATION POINTS
DEEP SPACE	PLANETS MOONS SUN ASTEROIDS	HELIOCENTRIC { INNER / OUTER PLANETARY ORBITS LIBRATION POINTS SOLAR SYSTEM ESCAPE

- SELECT CANDIDATE DESTINATIONS WITHIN CATEGORIES
- SELECT FOR UNIQUE ATTRIBUTE WITHIN CATEGORY

Figure 2.2-1: Candidate Destination Categories

disposal site in the geolunar system. Surface destination categories in deep space are more numerous. In principle, eight planets (excluding the Earth) are candidates along with about 40 moons (the exact count is expected to change with each Voyager planetary flyby) and some thousands of asteroids located in a belt between Mars and Jupiter, in the Jupiter-Sun libration or Trojan points, in the Apollo-Amor group with orbital radii of about 1 AU, and in a variety of irregular orbits. The only remaining deep space "surface" disposal site is the Sun.

Orbital Destination Categories. Orbital destination categories in the geolunar system include orbits about the Earth (both posigrade and retrograde at a range of orbital altitudes), orbits around the Moon, and orbits around the leading and trailing Earth-Moon libration points designated L4 and L5.

Orbital destination categories identified in deep space include circular heliocentric orbits both inside and outside the Earth's orbital radius of 1.0 AU; orbits about the planets; orbits about the Earth-Sun or Jupiter-Sun libration points; and solar system escape orbits which would carry the waste payload completely out of the solar system.

2.2.2 Selection of Candidate Destinations. Choices were based on which unique attribute of a given category was best represented by a given candidate destination within the category. In some cases the category possessed several favor-able attributes, leading to the selection of more than one candidate destination within the category. Figure 2.2-2 illustrates the candidate destinations selected, showing in summary form the rationale for their selection.

LOCATION	TYPE	CATEGORY	CANDIDATE DESTINATION	REF. NO	RATIONALE FOR SELECTION
GEOLUNAR SYSTEM	SURFACE	MOON	LUNAR SURFACE	1	ONLY AVAILABLE
	ORBITS	HEO	HEO POSIGRADE	2	LOWEST Δv
			HEO RETROGRADE	3	ENHANCED STABILITY
		LIBRATION POINTS	L4 TROJAN ORBITS	4	ENHANCED STABILITY
		LUNAR ORBITS	LUNAR ORBITS	5.	MINIMIZE RE-ENCOUNTER PROBABILITY
DEEP SPACE	SURFACE	PLANETS	VENUS IMPACT	6.	LOWEST Δv
			JUPITER ENTRY	7	NO LONG TERM CONTAMINATION
		MOONS	LUNA	1	LOWEST Δv (SEE GEOLUNAR)
		ASTEROID	APOLLO - AMOUR GROUP SOFT LANDING	8	LOWEST Δv, TRUE TIMES
		SUN	SOLAR IMPACT	9	ONLY AVAILABLE
	ORBITS	HELIOCENTRIC	EARTH ESCAPE ELLIPTICAL	10	LOWEST Δv FOR EHLIOCENTRIC
			0.85 AU CIRCULAR	11.	LOWEST Δv FOR VERIFIED STABILITY
			1 15 AU CIRCULAR	12	SEE GEOLUNAR
		LIBRATION POINTS	SUN-EARTH "TADPOLE" ORBITS	13	LOWEST Δv, STABILITY VERIFIED
		SOLAR SYSTEM ESCAPE	ESCAPE HYPERBOLA	14	LOW LONG TERM RISK

Figure 2.2-2: Candidate Destinations

Geolunar System. Within the geolunar system, surface destinations (excluding the Earth) are restricted to the Moon. Orbital destinations chosen include:

1) Posigrade High Earth Orbits - Offer the lowest delta-V requirements for destinations in the Earth-Moon system.

2) Retrograde Orbits - Retrograde high Earth orbits, which offer enhanced stability with respect to posigrade orbits (1)* at the cost of higher delta-V requirements.

3) Libration Points - Orbits around the L4 (or leading) Earth-Moon Trojan point. Analytical studies (1) indicate enhanced stability relative to posigrade high Earth orbits with comparable delta-V requirements.

4) Lunar Orbits - Lunar orbits which require delta-V's comparable to high Earth orbits, but offer a lower risk of re-encounter of the waste payload with the Earth by constraining re-encounter to the lunar surface.

Deep Space. Selection of destinations outside the Earth-Moon system provides a wide range of both surface and orbital destination choices. Surface destination choices include planets, moons, asteroids, and the Sun. Orbits were chosen about the Sun, about Sun-Earth libration points, and direct to solar system escape.

1) Planets - The surfaces of planets offer the possibility of eliminating the return of waste payloads by burying them deep in a planetary gravity well. Attributes of importance in selection of planets as waste disposal destinations include low delta-V for delivery, lack of solid surface to prevent permanent contamination, and frequency of launch windows.

Two planetary surface destinations were selected:

a) Venus - Requires the lowest delta-V of any planet for approach and landing. Its extremely hostile surface environment (carbon dioxide atmosphere at a pressure of about 90 atmospheres, with a surface temperature of 900°F) precludes any near term exploration or exploitation. The thick atmosphere would facilitate a soft landing; a number of probes have been landed by both the U.S. and the USSR.

b) Jupiter - Requires a high delta-V for approach and landing but, since it lacks a solid surface, the waste payload would continue to descend through the atmosphere until, in theory, it reached the dense core at a depth of over 60,000 km, rendering it completely inaccessible and eliminating any possibility of return.

The remaining planets were rejected as having higher delta-V requirements than Venus and possessing a surface immediately accessible for exploration or exploitation (Mercury, Mars) or sharing Jupiter's ability to dispose of the waste in deep atmosphere, but at the expense of increased delta-V.

*( ) denotes a reference listed in Appendix B.

2) Moons - The nearest planet having sizable moons is Jupiter. Mercury and Venus have no moons and the two moons of Mars are small (apparently captured asteroids). Jupiter's four Galilean satellites, while as big or bigger than Earth's Moon, offer no disposal advantages not offered by Jupiter itself. Accordingly, consideration of moons was restricted to Luna.

3) Asteroids - Asteroids as destinations for waste payloads would provide some of the advantages of planets without requiring a high delta-V penalty for soft landing. Advantages include partial meteroid shielding (at least 2pi steradians), a gravity well which would reduce risk of Earth re-encounter, and demonstrated orbital stability. Even small asteroids (radius of about 1 km) would provide sufficient surface area to accommodate large numbers of waste packages. Investigation of suitable asteroids has resulted in selection of the group of Apollo-Amor (Earth approaching) asteroids as the destinations of choice. These objects are in stable orbits and require a lower delta-V for rendezvous than any other objects outside the geolunar system (approx. 5.5 km/sec) (2).

4) Sun - The Sun offers the same advantages as Jupiter as a destination, plus the certainty that there will be no future exploitation activities to be compromised. The chief obstacle to use of the Sun is the high delta-V required for impact. Direct injection to solar impact was chosen for investigation; Jupiter flyby as an aid to solar impact was rejected as inferior to Jupiter impact.

Orbits. Orbits investigated as disposal sites included heliocentric orbits, orbits about Sun-planet libration points, and hyperbolic escape orbits which would allow the waste payload to escape the solar system altogether. Orbits about planets were rejected as having delta-V's higher than those required for selected planetary impact destinations, penalizing performance and offering no long term risk benefits.

1) Heliocentric Orbits - Consideration of heliocentric orbits was limited to circular orbits between Earth and Mars and Earth and Venus and elliptical orbits tangent to Earth's orbit at aphelion, with perihelions between 1.0 and 0.85 AU. Other closed heliocentric orbits were rejected as imposing a higher delta-V penalty with essentially no risk benefit.

a) Heliocentric Circular Orbits at 0.85 and 1.15 AU were selected as having the lowest delta-V for a heliocentric orbit compatible with analytically verified stability.

b) Earth Escape Elliptical orbits were proposed in an early study (3) and treated analytically in later NASA-sponsored efforts (4). This class of orbits has the lowest delta-V requirements of any ex-geolunar system destinations, lower in some cases than the delta-V to high Earth orbit.

2) Libration Point Orbits were investigated because of their inherent stability. Stable libration points at the Jovian Trojan points have been verified by telescopic observation of asteroids orbiting the points. Lonq term stability of these orbits is beyond question (5). Stability of objects in the so called "tadpole" orbits about the Earth-Sun Trojan points have been studied analytically, (6,5) with results implying positive stability for times on the order of 10^4 years. Tadpole orbits about the Sun-Earth Trojan points were selected for investigation because of the low potential delta-V and analytically verified stability.

3) Escape Hyperbolic Orbits allowing the waste to exit the solar system directly were chosen as the most direct and reliable implementation of the solar system escape option. Jupiter flybys as aids to escape were rejected as inferior to Jupiter impact.

2.3 DEFINE CANDIDATE DESTINATION CHARACTERISTICS

With 14 candidate destinations identified, the next step was to characterize them in areas suitable for comparison. Three broad areas for comparison were identified:

1) Long term risk of Earth re-encounter with the waste payload
2) Factors relating to mission cost
3) Implications for future use

The characterization effort was executed in two phases. In the first, past studies were reviewed to identify specific characteristics within the areas identified above which could be characterized in a fashion suitable for comparison. In the second phase, values for each characteristic identified were calculated or derived for each of the 14 candidate destinations.

2.3.1 Identify Key Characteristics. Identification of characteristics for definition was based on a series of studies for NASA conducted between 1977 and 1980. Key characteristics were identified for the areas described above.

Factors related to mission risk identified in a review of past studies included rescue options (4), long-term risk of Earth re-encounter (5,4), and long-term containment requirements.

Rescue Options allow continuation of a failed deployment mission by staging a dedicated rescue mission which would rendezvous with a failed deployment system, retrieve the waste payload, and deliver it to the original destination. With the rescue option it is not necessary to have deployment systems of extremely high reliability in order to achieve very low risk. Instead, the rescue mission provides this assurance through the powerful mechanism of redundancy. Such redundancy on a mission level is the best approach identified to date for achieving sufficient mission reliability to make space disposal a practical proposition (4). Accordingly, each candidate destination has been rated on its compatibility with incorporation of a rescue mission capability.

Long Term Risk of Earth Re-Encounter - Earth re-encounter refers to the possibility, however unlikely, that the waste payload could re-encounter the Earth, resulting in a high speed entry into the Earth's atmosphere with the subsequent possibility of waste form dispersion due to atmospheric burnup or high speed ground impact. Re-encounter can occur due to three primary causes: (1) a deployment system failure, (2) fallback of waste particles resulting from meteoroid fragmentation of the waste form, and (3) intersection of the waste payload destination orbit with the Earth's orbit. Intersecting orbits will eventually lead to re-encounter when both the Earth and the waste payload arrive at the intersection of the two orbits at the same time (4).

Long Term Containment Requirements - The third factor investigated was the requirement for long term containment for calcine or other powder waste forms. Liberation of powdered waste in Earth orbit would result in relatively rapid fallback to Earth (1).

Cost Related Factors - Factors related to mission cost were included to provide a differentiator for destinations having equivalent risk. Factors were evaluated for generic systems similar to the MSFC reference system, which uses chemical propellant injection stages and placement stages, except for the solar impact destination which requires a more advanced system because of its extremely high performance requirements. Evaluation of destinations against a common, generic transfer system allowed the relevant cost related factors to be evaluated on characteristics unique to the destinations.

Primary cost related factors identified included: delta-V or performance, injection and placement propulsion requirements, vehicle re-use options, and launch window requirements.

Delta-V - As the primary index of performance, the velocity increment from low Earth orbit is the primary factor in determining how much payload can be delivered to the specified destination for each shuttle launch. The net delivered payload per shuttle launch is the most important single factor in cost.

Injection and Placement Propulsion Requirements - Propulsion requirements were studied to determine the number of burns required of the injection and placement stage main propulsion and attitude control systems in a nominal mission. This provides an indicator of mission complexity. Injection and placement mission profiles were also investigated for critical guidance and navigation requirements (such as lunar soft landing or asteroid rendezvous) which require significant increases in capability when compared to the guidance requirements of the MSFC reference system. These critical requirements are reflected in higher costs for expended vehicles.

Vehicle Re-Use Options were identified as a function of destination to allow for the cost increases incurred by expending vehicles. Some options, such as lunar soft landing, mandate expending a relatively expensive stage; other options expend only a relatively low cost placement stage or allow recovery of all delivery vehicles. Re-use options were tabulated for each destination for injection, placement, and solar electric stages.

Launch Window Requirements - Some candidate destinations (such as those involving planetary surfaces) restrict launches of waste payloads to certain "windows" or periods of the year. This can have the effect of compressing a year's worth of launches into a period of 3 months, which leads to an increase in launch rate and the ground facilities needed to support launch. Launch windows were tabulated for each candidate destination in terms of percentage of the year available for launch.

Other Factors - Factors considered, in addition to risk and cost, include the ability to retrieve a waste form after it is placed at a destination, passive monitoring or the ability to detect and track a passive waste payload after deployment, and implications for future use.

Retrieval - Capability would allow use of space as a storage, rather than a permanent isolation site (7). Storage is not a requirement for the reference concept, but its adoption in the future is difficult to categorically rule out. Accordingly, the potential of each destination for use as a storage rather than permanent disposal site has been tabulated.

Passive Monitoring of waste payloads after delivery to the destination would allow determination of location and velocity to verify correct placement and would be necessary to any long term retrieval capability. Candidate destinations were examined to determine if they were close enough to the Earth for passive monitoring from the ground or from low Earth orbit (LEO).

Implications for Future Use - Removing nuclear waste from Earth is less desirable if an alternate useful location will be contaminated. Candidate destinations were examined in two areas to evaluate the likelihood of impacting future use:

1) Potential uses for candidate destinations were identified - examples include use of the L4 libration point as a site for space habitats or large scale exploitation of the lunar surface as a source of raw materials. No rigorous attempt at screening the feasibility of applications was made; in view of the uncertainty about what could happen in a thousand years, it is difficult to categorically exclude any future applications.

2) The degree of contamination of the destination was estimated.

Future use was considered to be compromised only if a potential use for the destination was identified, and the potential for widespread contamination was found.

2.3.2 Characterize Candidate Destinations. Identification of key characteristics allowed their evaluation for each candidate destination. Evaluations were based on results of past studies, supplemented by new analyses as required. Key characteristics determined are summarized in Figure 2.3-1. The following section provides a detailed description of each destination, including for each:

1. Mission scenario summary
2. Evaluation of long term risk of Earth re-encounter
3. Factors related to mission cost
4. Other factors, including implications for future use

REF No.	CANDIDATE DESTINATIONS / EVALUATION CRITERIA	RESCUE OPTION AT ANY POINT IN DEPLOYMENT SEQUENCE?	LONG TERM (10^6 YRS) RISK OF EARTH RE-EN COUNTER FOR 880 DEPLOYED PAYLOADS OVER 10^6 YRS (Kg OF RETURNED WASTE): DUE TO DEPLOYMENT FAILURE (WITH RESCUE IF AVAILABLE)	DUE TO FALLBACK OF PARTICLES RESULTING FROM WASTE FORM FRAGMENTATION BY METEOROID IMPACT	DUE TO EARTH INTERSECTION WITH NOMINAL DESTINATION ORBIT	TOTAL EXPECTED RETURN OF WASTE FORM IN 10^6 YR	ESTIMATED MAXIMUM DEPOSITION RATE KG/YR.	FACTORS RELATED TO MISSION COST: LONG TERM CONTAINMENT REQUIRED FOR CALCINE OR POWDERED WASTE FORMS	APPROXIMATE ΔV REQUIRED FOR DEPLOYMENT FROM 160 N.M. LOW EARTH ORBIT (KM SEC)	PLACEMENT REQ'MTS OR MISSION COMPLEXITY — INJECTION STAGE: MAIN PROP BURNS	RCS BURN (MIDCOURSE)	CRITICAL PLACEMENT? **	PLACEMENT STAGE: MAIN PROP BURNS	RCS BURNS	CRITICAL PLACEMENT? ***	DEPLOY VEHICLE RECOVERY OPTIONS: CHEMICAL PROPELLANT INJECTION STAGE	CHEMICAL PROPELLANT PLACEMENT STAGE	SOLAR ELECTRIC STAGE	LAUNCH WINDOW REQUIREMENTS	OTHER FACTORS: WASTE FORM RETRIEVABILITY STORAGE POTENTIAL	PASSIVE WASTE FORM MONITORING FROM EARTH SURFACE/LEO	IMPLICATIONS FOR FUTURE USE	COMMENTS
1.	LUNAR SURFACE SOFT LANDING	YES WITH ORBIT BIASING DURING IMPACT COMMIT BURN	-0-	-0-	0	0	0	NO	6.06	5	2	Y	--	--	--	EXP ONLY	--	--	ANY DAY WITH TARGET ORBIT BIAS	DIFFICULT REQUIRES EXTENSIVE LUNAR SURFACE SUPPORT OPS	DIFFICULT	MINOR TO NEGLIGIBLE	MAY REQUIRE COMPLEX LUNAR OPERATIONS TO ISOLATE WASTE PAYLOAD FROM LANDING SITE
2.	HIGH EARTH ORBIT POSIGRADE <25 E.R.	-YES-	-0-	ORDER OF 10	0	ORDER OF 10	ORDER OF 1×10^{-6} (OR 10 MG/YR)	YES	4.11 MAX	3	1	N	--	--	--	RCVR	--	RCVR	ANY DAY	STRAIGHT FORWARD	YES LASER OR SIRTF	NO IMPACT FOR A < 25 E.R.	STABILITY OF HIGH EARTH ORBITS NOT VERIFIED FOR LONG TERM "SKYLAB SYNDROME" MAY BE A FACTOR
3.	HIGH EARTH ORBIT RETROGRADE <25 E.R.	-YES-	-0-	ORDER OF 10	0	ORDER OF 10	ORDER OF 1×10^{-6} (OR 10 MG/YR.)	YES	4.40	3	2	N	--	--	--	RCVR	--	RCVR	ANY DAY	STRAIGHT FORWARD	YES LASER OR SIRTF *	NO IMPACT FOR A < 25 E.R.	STABILITY OF HIGH EARTH ORBIT NOT VERIFIED FOR LONG TERM "SKYLAB SYNDROME" MAY BE A FACTOR
4.	EARTH/MOON TROJAN ORBITS (L4)	-YES-	-0-	ORDER OF 10	0	ORDER OF 10	ORDER OF 1×10^{-6} (OR 10 MG/YR)	YES	3.90	4	2	N	--	--	--	RCVR	--	RCVR	ANY DAY	STRAIGHT FORWARD	YES LASER OR SIRTF *	MINOR TO NEGLIGIBLE	THEORETICALLY STABLE FOR LONG TERM REQUIRES VERIFICATION
5.	LUNAR ORBIT	-YES-	-0-	-0-	0	0	0	YES	4.25	4	2	N	--	--	--	RCVR	--	RCVR	ANY DAY WITH TARGET ORBIT BIAS	STRAIGHT FORWARD	YES LASER OR SIRTF *	NO IMPACT	STABILITY OF LUNAR ORBITS NOT VERIFIED
6.	VENUS IMPACT	-YES-	-0-	-0-	0	0	0	NO	3.90	1	1	--	--	2	N	RCVR	EXP	RCVR	APPROX 3 MO LAUNCH WINDOW ON 19 MONTH CENTERS (16%)	NO	NO	SUBSTANTIAL POTENTIAL FOR SURFACE CONTAMIN	LOWEST ΔV FOR PLANETARY IMPACT WASTE FORM INTEGRITY FOR LONG TERM DOUBTFUL
7	JUPITER IMPACT	-NO-	ORDER OF 1×10^4	-0-	0	ORDER OF 1×10^4	INTACT REENTRY OF WASTE FORM YIELDS ORDER OF 3×10^3 KG IN 30 SEC.	NO	7.01	2	1	--	--	2	N	EXP ONLY	EXP	EXP ONLY	APPROX 3 MO LAUNCH WINDOW ON 13 MO CENTERS (23%)	NO	NO	NO IMPACT	LACK OF SOLID SURFACE ALLOWS ELIMINATION OF WASTE FORM FROM ACCESSIBLE AREAS
8.	ASTEROID 'SOFT' LANDING	-YES-	-0-	-0-	0	0	0	NO	5.60	2	1	N	1	3	Y	RCVR	EXP	RCVR	APPROX 3 MO LAUNCH WINDOW ON 19 MONTH CENTERS (16%)	DIFFICULT REQUIRES EXTENSIVE SURFACE OPS	NO	SUBSTANTIAL POTENTIAL FOR SURFACE CONTAMIN	COMPLEX RENDEZVOUS AND "LANDING" OPS HALVES METEOROID FLUX GRAVITY WELL TOO SHALLOW FOR FRAG CONTAIN ON IMPACT LANDING
9.	SOLAR IMPACT	-NO-	ORDER OF 5×10^4	-0-	0	ORDER OF 5×10^4	INTACT ENTRY OF 1 WASTE FORM YIELDS ORDER OF 3×10^3 KG IN 30 SEC.	NO	24.06	CONTIN-UOUS	--	--	--	--	--	--	EXP NEPS	--	ANY DAY	NO	NO	NONE	HIGHEST COST HIGH RISK
10.	EARTH ESCAPE ELLIPSE .85 AU P/H	-YES-	-0-	-40-	ORDER OF 1×10^5	ORDER OF 1×10^5	INTACT ENTRY OF 1 WASTE FORM YIELDS ORDER OF 5×10^3 KG IN 30 SEC.	NO	3.65	2	1	N	--	--	--	RCVR	--	RCVR	ANY DAY	DIFFICULT	NO	NONE	UNACCEPTABLE RISK OF RE ENCOUNTER
11	HELIOCENTRIC ORBIT INNER (0.85 AU)	-YES-	-0-	-0-	0	0	0	NO	4.64	2	1	N	1	1	N	RCVR	EXP	RCVR	ANY DAY	DIFFICULT	NO	NONE	KICKSTAGE RCS BURN IS FOR REORIENTATION JUST PRIOR TO MPS BURN SPIN STABILIZED KICKSTAGE
12.	HELOCENTRIC ORBIT OUTER (1.15 AU)	-YES-	-0-	-0-	0	0	0	NO	4.72	2	1	N	1	1	N	RCVR	EXP	RCVR	ANY DAY	DIFFICULT	NO	NONE	SUBSTANTIALLY IDENTICAL TO HELIOCENTRIC ORBIT (INNER) SLIGHTLY HIGHER ΔV
13.	SUN/EARTH "TADPOLE" ORBITS	-YES-	-0-	-0-	0	0	0	NO	5.50 (FOR 300 DAY PHASING)	2	1	N	1	2	Y	RCVR	EXP.	RCVR	ANY DAY	DIFFICULT	NO	NONE	VERY PROMISING IF PHASING DURATION/ΔV CAN BE DECREASED. VERY PRECISE NAV REQ'MTS FOR PLACEMENT
14.	SOLAR SYSTEM ESCAPE	-NO-	ORDER OF 1×10^4	-0-	0	ORDER OF 1×10^4	INTACT ENTRY OF ONE WASTE FORM YIELDS ORDER OF 5×10^3 KG IN 30 SEC.	NO	8.75	1	--	--	--	--	--	EXP ONLY	--	EXP ONLY	ANY DAY	NO	NO	NONE	LOWEST RISK OPTION IF NO RESCUE

NOTE: EXP = EXPEND
RCVR = RECOVER
E.R. = EARTH RADIUS

* SHUTTLE INFRARED TELESCOPE FACILITY

** "CLOSED LOOP" REAL TIME MANEUVER CAPABILITY

**** HIGH NAVIGATIONAL ACCURACY/RENDEZOUS CAPABILITY

Figure 2.3-1: Candidate Destination Characteristics Summary

CANDIDATE DESTINATION 1. LUNAR SURFACE SOFT LANDING

1) Mission Scenario Summary

The mission includes the following primary events: (1) delivery of the waste payload to low Earth orbit by the uprated shuttle and subsequent transfer to the OTV, (2) a velocity impulse maneuver to place the payload into lunar capture orbit, (3) a lunar velocity impulse maneuver to place the payload into lunar capture orbit, and (4) final burns to soft land the package and OTV in the selected landing area on the Moon's surface in the crater Billy (selenographic coordinates of -50° longitude and -14° latitude) (7).

2) Long Term Risk of Earth Re-Encounter

Long term risk is negligible after deployment. Proper biasing of the delivery trajectory will allow rescue after failure at any point in the delivery trajectory (1); rescue is the chosen method for elimination of Earth encountered due to deployment failure (8).

3) Factors Related to Mission Cost

The relatively high delta-V of this mission reduces the delivered payload per shuttle launch relative to the MSFC defined reference mission payload. The mission also expends a capable and relatively expensive orbit transfer vehicle and, by restricting launch to about 50% of the available time to avoid possible high risk abort modes, would have a substantial impact on launch facilities. Extensive lunar surface operations required to protect waste payloads from subsequent landings would also add to mission cost (9).

4) Other Factors

Waste payload retrieval (if desired) would be expensive in terms of performance and would also require extensive lunar surface operations (possibly manned). Monitoring and tracking would be simplified as the payloads location would be fixed once delivered.

An examination of implications for future use indicates that a considerable body of work has been generated concerning use of lunar resources for construction of space habitats (10) and solar power satellites (11,12). Use of the Moon as a location for science and astronomical bases is also likely (9). The possibility of large scale contamination, however, is small. Reasonable precautions should reduce the probability of a payload impact outside the target crater to extremely low values (8). Even if a payload did impact outside the selected crater, it is highly probable that the impact effects would be confined to a total area of considerably less than 1 km due to burial of the waste payload (13) and would not be a serious obstacle to exploitation of the remaining 3.8×10^7 km^2 of lunar surface.

CANDIDATE DESTINATION 2. POSIGRADE HIGH EARTH ORBIT

1) Mission Scenario Summary

Primary events include: (1) delivery of the waste payload to low Earth orbit by the uprated space shuttle with subsequent transfer to the OTV, (2) a velocity impulse maneuver to an elliptical transfer orbit with apoapsis at the target orbit altitude of 55,000 km (7), (3) a circularizing velocity impulse maneuver to place the payload into a circular orbit at the destination altitude, and (4) recovery of the OTV or re-use after payload release by reversing the maneuvers described above.

2) Destination Stability

Stability of prograde high Earth orbits has been extensively studied by Science Applications Incorporated (SAI) for NASA/ARC (1). Analysis has indicated that stable, low inclination prograde orbits must be inside about 25 Earth radii (ER) (159,000 km) for long term stability. Numerical integrations used for stability calculations were run over time periods about three orders of magnitude shorter than the 1 x 10^6 time periods of interest; it is quite possible that further integration would have shown instability for the 25 ER case. Further analysis is required to define the upper altitude bounds for long term stability. The 55,000 km orbit radius chosen as the reference HEO destination is sufficiently inside the 25 ER limit to represent conditions required for long term stability.

3) Long Term Risk of Earth Re-Encounter

The low eccentricity elliptical transfer orbits and circular destination orbits chosen allow ample orbital lifetime for rescue mission staging for OTV failure at any point in the mission. Rescue from the possible failure orbits is straightforward and will ensure against waste payload re-entry due to failure during deployment.

The total mass liberated by meteoroid strikes for fallback to Earth was calculated by combining statistical odds of meteor strikes and characteristics of waste payload fragment liberation from Reference 14 with estimates of Earth orbit lifetime for a range of fragment sizes from Reference 1.

Figures indicate that in a $1x10^6$ year period, about 2 out of 580 deployed waste payloads would experience meteor strikes which would liberate a total of 0.2% of the material in the payloads. Of approximately 25.5 kg of liberated material, about 28% would be fragments in the size range which would fall back to Earth within $1x10^6$ years, resulting in a total expected fallback in $1x10^6$ years of about 10 kg. The maximum deposition rate resulting, based on particle size distribution and orbital lifetime, will be approximately 10 mg/yr. Calcine or other powdered waste forms in these orbits would require containment for periods of about $1x10^6$ years. If containment were ruptured, approximately half of the liberated waste form would fall back in $1x10^6$ years.

4) Factors Related to Mission Cost

The velocity increment (delta-V) from low Earth orbit required for disposal in high Earth orbit is 4.0 km/sec (7), the third lowest for the 14 candidate destinations considered. The lower delta-V provides several advantages:

- The payload deployed is increased
- No separate placement stage is required
- The injection stage is recovered for reuse
- Launch windows are available any day, at any time (7)

These factors offer the potential for significant cost savings when compared to the reference heliocentric orbit destination.

5) Other Factors

Waste payload retrieval from the HEO destination is straightforward operationally and relatively undemanding in terms of performance, making HEO a valid candidate for storage as well as long term isolation (7). Position and velocity of waste payloads in HEO could be monitored from the ground using passive optical sensors such as the GEODSS (ground-based electro-optical deep space surveillance) system being developed for the USAF by TRW (15) or by active laser radars similar to those used on the ALSEP and LAGEOS programs. The SIRTF (Shuttle Infrared Telescope Facility) planned by NASA would be capable of tracking waste payloads by their own infrared emission from LEO (14). Passive tracking could be used to verify orbit elements independent of active beacon transponders and could aid in rescue or recovery operations.

Implications for future use are minimized by choosing an orbit higher than the geosynchronous orbit altitude of 42,241 km with an inclination of 28 deg (7). Traffic outside the geosynchronous arc is expected to be minimal.

6) Comments

The high number of payloads to be deployed (perhaps 60 to 100 per year) has led to consideration of grouping them at depots (16). A depot capable of handling 100 payloads would be deployed about once a year; 100 years of operation would lead to a depot population in the disposal orbit about the same as the 1980 population of satellites at GEO (approximately 90 satellites) with a minimum average intra-depot spacing of 3840 km . Depots could simplify keeping track of waste payloads and could minimize the already low probability of collisions between waste payloads or between previously deployed waste payloads and delivery vehicles. Against this must be weighed the increased mission complexity associated with depot rendezvous and the additional transfer operations associated with attaching the waste payload to the depot.

CANDIDATE DESTINATION 3. RETROGRADE HIGH EARTH ORBIT

1) Mission Scenario Summary

The demanding performance requirements imposed in achievement of a retrograde Earth orbit (equivalent to a 180 deg plane change) led to a short trade study aimed at defining the minimum energy delivery trajectory. The results led to a selection of a mission scenario somewhat more complex than that for the prograde HEO destination. Primary events after delivery of the OTV and waste payload to LEO include: (1) a velocity impulse maneuver to an elliptical outbound transfer orbit with an apoapsis at 4×10^5 km (about lunar distance); (2) a second velocity impulse maneuver at apoapsis of the transfer orbit which simultaneously accomplishes a 180 deg plane change and inserts vehicle and payload into an elliptical inbound transfer orbit with periapsis at the destination orbit radius of 160,000 km (just at the limit determined for long term orbit stability at 25 Earth radii (1) to minimize delta-V); (3) a third velocity impulse which circularizes vehicle and payload in the destination orbit. Recovery of the OTV for reuse after payload release is accomplished by reversing the maneuvers described above.

2) Destination Stability

Stability of retrograde high Earth orbits was addressed by the same study which looked at prograde orbits (1). Analysis has indicated enhanced stability compared to prograde orbits at the same radius and inclination.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. Failure of the delivery vehicle at most points in the delivery mission would leave the vehicle and payload in stable orbits with ample time for staging of rescue missions, providing sufficient insurance against Earth re-encounter. Failure during approximately the last 20 sec of the first velocity impulse would, however, leave the vehicle in a highly eccentric orbit ($e > 0.95$) subject to solar perturbations of periapsis altitude for certain choices of launch date. Absolute insurance against encounter within 100 days of launch (adequate time for rescue) could require restricting launches to about 120 days of the year when the Sun is at solar longitudes of -20° to 45° or 160° to 225° (1). Two alternate strategies were evaluated which would allow launch at any time: (1) provide an independent propulsion and guidance system capable of inserting the vehicle into an abort orbit with eccentricity low enough and periapsis high enough to ensure against re-encounter until rescue can be effected (the delta-V required for this maneuver is about 50 m/sec); (2) implement an intermediate velocity impulse between the first and second, in effect inserting the vehicle into a low eccentricity transfer ellipse to an apoapsis of about 6,000 km, and then initiating the insertion to the transfer to 4×10^5 km when 6,000 km radius is reached (this trajectory will leave the vehicle in a stable orbit after a failure at any arbitrary point at a delta-V penalty of 1.8 km/sec).

The abort orbit option has been selected as baseline for the study because of its substantially lower delta-V penalty.

Deployment System Failure. Abort orbit provisions plus rescue capability are the primary insurance against Earth encounter due to deployment system failure.

Due to Particle Fallback After Meteroid Strike. Total expected return, maximum deposition rate, and long term containment requirements are identical to destination 2, HEO posigrade.

3) Factors Related to Mission Cost

The delta-V required for the retrograde destination is 0.4 km/sec higher than for the posigrade orbit. This and the requirement to add redundant altitude control and auxiliary propulsion penalize the payload slightly compared to the prograde case, but the other important cost saving features are identical:

- o No separate placement stage
- o Injection stage recovered and reused
- o Launch window any day at any time (with abort orbit redundancy provisions)

The major cost difference remains the higher velocity increment.

4) Other Factors

Retrievability, compatibility with passive tracking, and implications for future use are identical to destination 2.

CANDIDATE DESTINATION 4. EARTH-MOON LIBRATION POINT ORBITS (L_4)

1) Mission Scenario Summary

Placement of the waste payload in a libration point orbit is similar to injection into a high Earth orbit at a radius of about 314,000 km. The target orbit is approximately elliptical in shape and close to but not exactly centered on the L_4 or leading Earth-Moon Trojan point. The orbit has a semi-minor axis of 71,000 km aligned with a line connecting the orbit center with the center of the Earth; the semi-major axis is about 144,000 km. Waste payloads would make one rotation about the orbit in about 28.8 days; the motion is phase-locked with the relative angle of the Sun to the orbit center (17).

The mission profile has been adapted from reference 7 to eliminate near term risk of Earth encounter in the event of a system failure, allowing rescue. Primary events after injection of payload and OTV into LEO by the uprated STS include:

(1) A velocity impulse maneuver which injects the vehicle into an intermediate transfer ellipse. The major axis and eccentricity of the intermediate orbit is limited so that, in the event of failure during or after the injection burn, lunar and solar perturbations will not cause a re-encounter with Earth prior to execution of a rescue mission.

(2) At apoapsis of the intermediate orbit, a second velocity impulse maneuver accomplishes a plane change aligning the inclination of the OTV orbit to a value compatible with destination orbit insertion targeting requirements, and raising the periapsis to an altitude of 5000 km, ensuring stability of the final transfer orbit against solar perturbation induced periapsis lowering into the Earth's atmosphere.

(3) A third velocity impulse at the new periapsis injects the OTV and payload into the final transfer orbit with apoapsis at the destination orbit radius of 318,000 km.

(4) The fourth velocity increase inserts the waste payload into the destination Trojan point orbit.

Return of the OTV is accomplished by a single de-orbit and plane change burn followed by a direct descent along an inbound transfer ellipse to an aerobraked deceleration into LEO and recovery for reuse by the shuttle. Total mission duration is about 9.5 days.

2) Destination Stability

The L_4 Trojan point orbit stability has been demonstrated analytically using linear stability analysis (17). Further analyses using numerical methods similar to those of Katz (18) are suggested if further verification is required.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The reference trajectory described leaves the OTV and payload in a stable orbit after an OTV failure at any point in the delivery mission. Sufficient time is available for multiple rescue attempts.

Deployment System Failure. Ensured rescue capability after a failure at any point in the deployment orbit allows mission continuation after deployment system failure.

Due to Particle Fallback After Meteoriod Strike. Total expected return, maximum deposition rate, and long term containment requirements are identical to destination 2, HEO prograde.

Due to Earth Intersection with Nominal Destination Orbit. Earth intersection with the Trojan point orbit is impossible.

4) Factors Related to Mission Cost

The delta-V for the Trojan point orbit deployment is 3.9 km/sec from LEO (10). This is about 0.1 km/sec lower than the delta-V required for HEO disposal; other factors are similar:

- o The payload deployed is increased when compared to the reference heliocentric orbit destination
- o No separate placement stage is required
- o The injection stage is recovered for reuse
- o Launch windows can be chosen for any day at any time

Except for the minor decrease in velocity increment, cost should be essentially similar to the cost for HEO disposal.

5) Other Factors

Retrievability and compatibility are essentially identical to destination 2. Retrieval is facilitated by the relatively low delta-V; passive tracking from ground or LEO is possible using laser radar or the shuttle based infrared telescope facility.

Implications for future use are not indicated. No alternate uses for Trojan point orbits have been proposed; even the proposed space colonies which would occupy the L_4 point itself have been "moved" to a perturbed high Earth orbit at an altitude of about 2×10^5 km (10). Other uses for the L_4 and L_5 points have been proposed; large radio telescopes are one possibility. Use of the libration points for these purposes should not be compromised as the closest approach of the waste payloads in their orbit will be about 71,000 km.

6) Comments

Use of orbiting depots (similar to those discussed for destination 2, high Earth orbit) deserves consideration to avoid crowding of individual payloads.

CANDIDATE DESTINATION 5. LUNAR ORBIT

1) Mission Scenario Summary

Placement of the waste payload into a lunar orbit with radius of 21,700 km is broadly similar to the HEO destination mission, but slightly more complex, requiring four OTV main propulsion system (MPS) burns plus midcourse corrections using the reaction control system (RCS). The mission profile has been adapted from Reference 7 to eliminate near term risk of Earth re-encounter in event of a system failure. Primary events after injection of payload and OTV into LEO by the updated space shuttle include:

(1) A velocity impulse maneuver which injects the vehicle into an intermediate transfer ellipse. The major axis and eccentricity of the intermediate orbit is limited so that, in the event of failure during or after the periapsis burn, lunar and solar perturbations will not cause a re-encounter with Earth prior to execution of a rescue mission.

(2) At apoapsis of the intermediate orbit, a second velocity impulse maneuver accomplishes a plane change aligning the inclination of the OTV orbit to a value compatible with destination orbit insertion targeting requirements and raising the periapsis to an altitude of 5000 km, ensuring stability of the final transfer orbit against solar perturbation induced periapsis lowering into the Earth's atmosphere.

(3) A third velocity impulse at the new periapsis injects the OTV and payload into the final transfer orbit which places the OTV within the lunar sphere of influence and near the desired close approach to the Moon.

(4) Burn four injects the OTV into the desired lunar orbit.

Return of the OTV is accomplished by a single de-orbit and plane change burn followed by a direct descent along an inbound transfer ellipse to an aerobraked deceleration into LEO and recovery for re-use by the shuttle. Total mission duration is about 10 days.

2) Destination Stability

Lunar orbits in the Earth-Moon travel plane are expected to be stable for extremely long periods of time (7). Further analysis using numerical methods similar to those described in Reference 1 are suggested if further verification is required.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The reference trajectory described leaves the OTV and payload in a stable orbit after an OTV failure at any point in the delivery mission. Sufficient time is available for multiple rescue attempts.

Deployment System Failure. Ensured rescue capability after failure at any point in the deployment orbit allows mission continuation after deployment system failure.

Due to Particle Fallback After Meteoroid Strike. The lunar orbit option is potentially less susceptible to Earth re-encounter due to particle fallback as liberated particles from a payload in lunar orbit would have a tendency to fall into the lunar surface. The exact benefits in terms of reduction in fallback to Earth require further investigation (7).

Due to Earth Intersection with Nominal Destination Orbit. Earth intersection with the lunar orbit is impossible.

4) Factors Related to Mission Cost

The total delta-V to lunar orbit deployment is 4.25 km/sec from LEO (7). This is about 0.15 km/sec higher than the delta-V required for HEO disposal resulting in a slight decrease in payload delivered per mission; other factors are similar:

- No separate placement stage is required.
- The injection stage is recovered for re-use.
- Launch windows can be chosen for any day at any time.

Except for the payload decrease due to the slightly higher velocity increment, costs should be essentially similar to the cost for HEO disposal.

5) Other Factors

Retrievability and compatibility with passive remote tracking are similar to destination 2; retrieval delta-V is approximately equivalent, and passive tracking from ground or LEO could be accomplished using laser radar or the shuttle based infrared telescope facility. Implications for future use are minor to negligible if orbits are chosen which will not interfere with future Earth-Moon transport. No alternative uses have been identified for the lunar orbit destination and lunar fallback should have negligible impact on proposed habitats which are universally viewed as being radiation hardened against solar flares and cosmic radiation.

6) Comments

Use of orbiting depots similar to those considered for destination 2, high Earth orbit should be considered to avoid crowding of individual payloads.

DESTINATION 6. VENUS IMPACT

1) Mission Scenario Summary

Injection to the transfer orbit to Venus impact is one of the simplest mission profiles studied. Primary events after injection of payload and OTV into low Earth orbit by the uprated space shuttle include: (1) a primary velocity impulse maneuver which injects OTV and payload into the minimum energy Venus impact transfer orbit; (2) separation of the waste payload attached to an orbit correction stage; (3) an initial midcourse correction performed by the orbit correction stage at about injection plus 5 days to correct for injection errors; (4) if required a second maneuver at entry minus 30 days to correct execution errors from the preceding maneuvers; and (5) Venus entry at injection plus 160 days. The orbit correction stage enters with the payload and is expended.

Injection stage (OTV) recovery begins with a retroburn immediately following separation of the payload. This injects the stage into an elliptical return orbit. An aerobraking maneuver at periapsis inserts the OTV into a phasing orbit. Burn at apoapsis of the phasing orbit raises periapsis to 260 km and is followed in half a revolution by LEO circularization. OTV recovery operations and shuttle landing complexes the mission. Total OTV mission duration away from LEO is about 62 hours.

2) Destination Stability

Stability of the Venus surface destination against Earth re-encounter is absolute.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The selected injection trajectory allows the waste payload and associated orbit transfer systems to be rescued after failure at any point in the deployment sequence with sufficient time before re-encounter to allow multiple rescue attempts.

Due to Deployment System Failure. Capability for execution of multiple rescue attempts after deployment system failure at any point ensures mission continuation after deployment system failure.

Due to Particle Fallback After Meteoroid Strike. Fallback from the Venus surface destination is impossible.

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with Venus orbit is impossible.

4) Factors Related to Mission Cost

Total delta-V from LEO to the Venus impact transfer orbit is 3.99 km/sec, lower than all but one of the geolunar system destination options. Total delta-V required for the orbit correction stage is about 50 m/sec, and the orbit correction stage mass is negligible compared to the payload mass (about 5% of the waste payload mass).

Primary additional influences on cost include:

- Recovery of the injection stage for re-use.
- An expendable placement stage is used up with each mission.
- Launch windows are available for only 3 months out of each 19. This amounts to about 16% availability, and could require an excessive launch rate of very close to one launch per day for 3 months.

Except for the excessive launch rate, costs should not differ greatly from costs of disposal at destinations in the geolunar system.

5) Other Factors

Retrievability or passive tracking are impossible for payloads once delivered to the surface of Venus.

Implications for future use are serious. The potential for widespread contamination exists and could seriously compromise the terraforming of Venus as speculated on by Sagan et al. The ambient temperature (482°C) and corrosive environment at the surface could lead to early loss of containment and waste form integrity. Venus surface probes have discovered surface winds of several km/hr, which at the very high pressure and density (CO_2 at 100 atmospheres pressure) present on the surface, could spread the degraded waste forms across the surface.

6) Comments

Launch window constraints would probably limit consideration of Venus surface disposal to waste mixes requiring much lower launch rates than the reference PW-4B waste mix.

DESTINATION 7. JUPITER IMPACT

1) Mission Scenario Summary

The mission scenario for Jupiter impact is one of the simplest studied, as no stage recovery operations are involved. A long life placement stage is used for trajectory corrections after separation from the expended injection stage. Primary events after injection of payload and orbit transfer system into low Earth orbit by the uprated space shuttle include: (1) a primary velocity impulse maneuver which injects OTV and payload into the minimum energy Jupiter impact transfer orbit (the injection stage is expended in this maneuver); (2) separation of the waste payload attached to an orbit correction stage; (3) an initial course correction maneuver at about injection plus 5 days to correct for injection errors; (4) if required, a second course correction and targeting maneuver at about entry minus 60 days to correct execution errors from preceding maneuvers; and (5) Jupiter entry at approximately injection plus 1000 days (2.74 yrs).

2) Destination Stability

Stability of the Jupiter surface destination against Earth re-encounter is absolute.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. Because of the relatively large hyperbolic excess velocity, extended duration, and extreme ranges (up to 6 AU) involved in transfer to Jupiter, rescue is beyond the capabilities of the orbit transfer systems used for deployment. As development of high capability, advanced space systems strictly for rescue is not considered feasible, the Jupiter impact option is considered to preclude rescue as an option for coping with deployment system failures.

Due to Deployment System Failure. Probability of Earth re-encounter due to deployment system failure was evaluated using data from Reference 5. The assumption was made that, due to similarity in departure hyperbolic excess velocity, the probability of re-encounter for Jupiter impact was identical to the calculated value in Reference 5 for re-encounter due to deployment failure during solar system escape.

The probability as calculated of Earth re-encounter within $1x10^6$ years due to deployment system failure is 0.006 for a single mission (5). The probable number of Earth re-encounters for the 580 missions in the reference mission set becomes (0.006)(580) or about 3.5 payloads. For the reference payload of 3270 kg, this is equivalent to a rounded total deposition of about 10,000 kg. Even more important is the deposition rate. Instead of being distributed over a $1x10^6$ year span, the waste payload would be deposited in the atmosphere in 3270 kg increments over a period of 30 sec. Three or more such events are probable in the million years following deployment. Atmospheric dispersion of the waste payload is probable due to an entry velocity of between 11 and 41 km/sec. Most entries would occur at velocities between 11.2 and 12 km/sec, with about 10% ranging upwards between 12 and 41 km/sec (5). The higher velocities in this range would pose a severe risk to waste payload containment.

Due to Particle Fallback After Meteoroid Strike. Fallback from the Jupiter atmosphere destination is impossible.

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with the Jupiter orbit is impossible.

4) Factors Related to Mission Cost

Total delta-V from LEO to Jupiter impact transfer orbit insertion is about 7 km/sec (3). An orbit correction stage with 50 m/sec delta-V capability is used. The relatively high velocity increment reduces payload by about 1/3 compared to the reference destination, and requires expending a capable and relatively costly cryogenic propellant OTV. Primary additional influences on cost include:

- An expendable placement stage is used up with each mission.

- Launch windows are available for only 3 months out of each 13 (3).

This amounts to about 23% availability and would require an excessive launch rate of two launches every 3 days.

This destination shows the potential for significant cost increase compared to the reference heliocentric orbit destination.

5) Other Factors

Retrieval or passive tracking are impossible for payloads delivered into the atmosphere of Jupiter.

Implications for future use are negligible. Waste payloads or dispersed particles of waste payloads are compelled, because of their high molecular weight, to sink rapidly to depths in the Jovian atmosphere where pressure and temperature are high enough to preclude exploration or exploitation. Given enough time, waste particles will diffuse all the way to Jupiter's dense metallic core; about 60,000 km below the cloud tops.

6) Comments

Launch window constraints, high long-term risk of re-encounter, and low performance make this option relatively unattractive despite very efficient isolation of delivered payloads.

DESTINATION 8. ASTEROID SOFT LANDING

1) Mission Scenario Summary

The asteroid soft landing is, along with lunar soft landing, the most complex mission scenario studied. In addition to the re-usable injection stage, an expendable placement stage with a primary propulsion capability of 1.0 km/sec and autonomous rendezvous and landing capability is required for terminal rendezvous with the asteroid destination.

Primary events after injection of the payload and orbit transfer system into low Earth orbit by the uprated space shuttle include: (1) a primary velocity impulse maneuver which injects the OTV and payload into the minimum energy asteroid transfer orbit; (2) separation of the waste payload attached to the placement stage; (3) an initial midcourse correction performed by the placement stage at about injection plus 5 days to correct for injection errors; (4) if required, a second maneuver at rendezvous minus 30 days to correct execution errors from the preceding maneuvers; (5) a second primary propulsion maneuver which leaves placement stage and payload in the same orbit with the asteroid at a distance of about 750 km; (6) terminal rendezvous operations; and (7) landing/docking operations followed by explosive anchoring to the asteroid surface.

Injection stage (OTV) recovery begins with a retroburn immediately following separation of the payload. This injects the stage into an elliptical return orbit. An aerobraking maneuver at periapsis injects the OTV into a phasing orbit. Burn at apoapsis of the phasing orbit raises periapsis to 1260 km and is followed in half a revolution by LEO circularization. OTV recovery operations and shuttle landing complete the mission. Total OTV mission duration away from LEO is about 62 hours.

2) Destination Stability

Stability of the asteroid surface destination against Earth re-encounter is absolute.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The selected injection trajectory allows the waste payload and associated orbit transfer systems to be rescued after failure at any point in the deployment stages with sufficient time before re-encounter to allow multiple rescue attempts.

Due to Deployment System Failure. Capability for execution of multiple rescue attempts after deployment system failure at any point ensures mission completion after deployment system failure.

Due to Particle Fallback after Meteoroid Strike. Fallback from the asteroid surface destination is impossible.

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with the asteroid orbit is impossible.

4) Factors Related to Mission Cost

Typical delta-V from LEO to the asteroid surface destination (1943 Anteros) is 5.50 km/sec, divided approximately into a 4.6 km/sec injection maneuver and a 0.9 km/sec asteroid orbit insertion maneuver. Due to the requirement for 0.9 km/sec placement delta-V, the placement stage mass is equal to about 60% of the payload mass; delivered payload mass is almost identical to the mass of payload delivered by the reference heliocentric orbit mission. Primary additional influences on cost include:

- o Recovery of the injection stage for re-use.
- o Complexity/cost of the automated rendezvous and "landing" capability required by the placement stage.
- o Expending of the placement stage with each mission.

- Launch windows are available for only 3 months out of each 19. This amounts to about 16% availability and would require an excessive launch rate of very close to one launch per day for 3 months.

The excessive launch rate and the complex guidance required for the placement stage combine to increase the cost of this option relative to disposal at the reference heliocentric orbit destination.

5) Other Factors

Retrievability of payloads is not considered feasible due to energy (delta-V) requirements and the operational difficulties involved in waste payload location and pickup on the asteroid surface.

Passive tracking of waste payloads from Earth surface or LEO locations is impossible for existing technology due to the extreme range required.

Implications for future use are minimal. Any asteroid selected could be at least partly barred from other uses but, with a total of 47 asteroids identified so far in the Apollo-Amor group (2), designation of one for space disposal is a reasonable approach.

6) Comments

The asteroid, in effect, serves as a waste "depot" with a stable orbit. The relatively high velocity requirements, the cost of expending a complex placement stage, and the limited launch window weigh against the assured stability of the destination in assessment of this option.

DESTINATION 9. SOLAR IMPACT

1) Mission Scenario Summary

The extreme energy requirements of the solar impact destination (C_3=725 km^2/sec^2) are equivalent to an impulsive hyperbolic excess velocity of about 24 km/sec (5) and are beyond the practical capabilities of chemical propulsion systems. Reference 5 examined advanced propulsion options for solar impact disposal of nuclear waste and selected two nuclear electric propulsion (NEP) options as feasible. For this study, the most economical option in terms of shuttle launches was used. The mission scenario is simple. Primary events after injection of the NEP stage and waste payload into low Earth orbit by the approved space shuttle include: (1) engine startup and spiral to Earth escape injection (about 800 days); (2) a continuous low acceleration propulsion maneuver which results in placement in a solar impact trajectory (about 560 days); and (3) passive inbound fall to solar impact (355 days) (total mission duration from start burn in LEO to photosphere entry is 1120 days or about 3 years).

2) Destination Stability

Stability of the solar impact destination against Earth re-encounter is absolute.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. Because of the high hyperbolic excess velocity required (in excess of 24 km/sec) and extreme ranges (in excess of 3.5 AU) involved in the transfer to the solar impact orbit, rescue in times shorter than reasonable system lifetimes for the deployment orbit transfer systems is beyond their capabilities. Failure of a beacon transponder at the ranges required would render the vehicle untrackable, also preventing rescue. Accordingly, the solar impact option is considered to preclude rescue as an option for coping with deployment system failures.

Due to Deployment System Failure. Probability of Earth re-encounter due to deployment system failure within 1×10^6 years was calculated as 6×10^{-3} for a single mission (5). This is identical to the probability calculated for destination 6, Jupiter impact, and following the analysis conducted for destination 7, it is probable that this will result in three or more waste payload reentries in 1 x 10^6 years. Entry velocities would range between 12 and 45 km/sec, but in this case the chance of entry velocity being over 15 km/sec is 65% with a significant (greater than 20%) probability that the velocity will be over 40 km/sec. The extreme entry velocities increase the likelihood of loss of containment during entry, allowing some portion of the waste payload to disperse in the atmosphere.

Due to Particle Fallback from Meteroid Strike. Fallback from the solar impact destination is impossible.

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with the solar impact destination is impossible.

4) Factors Related to Mission Cost

Total cost for the solar impact mission studied is dominated by the cost of expending the nuclear electric propulsion stage. Costs for the stage were estimated from Reference 15 at approximately $115M per unit (1980 $). Exercising this over a range of assumptions on unit production costs led to selection of an average cost per stage of $100M per unit. This figure so dominates mission cost that other factors were neglected.

5) Other Factors

Retrieval and passive tracking are impossible with existing technology; implications for future use or other impact are negligible.

6) Comments

The combination of high cost, due to expending an extremely expensive upper stage, and high risk, due to lack of rescue capability, must be weighed against the security of the solar surface as a destination.

DESTINATION 10. EARTH ESCAPE ELLIPSE

1) Mission Scenario Summary

The injection to a minimum energy Earth escape ellipse requires less energy than any other destination studied and shares the simplest injection mission profile with solar system escape. Primary events after injection of payload and orbit transfer system into low Earth orbit by the upgraded space shuttle are limited to a single velocity impulse maneuver which injects the waste payload into an elliptical heliocentric orbit with a perihelion at about 0.85 AU.

Injection stage recovery from the escape orbit begins with a retroburn immediately following separation of the payload. This injects the stage into an elliptical return orbit. An aerobraking maneuver at periapsis inserts the OTV into a phasing orbit; a burn at apoapsis of the phasing orbit raises periapsis to 260 km and is followed in half a revolution by LEO circularization. OTV recovery operations and shuttle landing complete the mission. Total OTV mission duration away from LEO is about 62 hours.

2) Destination Stability

Probability of re-encounter of a single waste payload deployed in a heliocentric, elliptical orbit tangent to the Earth's orbit has been calculated in reference 4 as about 0.2 for a period of 1 x 10^6 years. This result, calculated using a numerical Monte Carlo simulation of planetary encounters, is lower than the probabilties as predicted by analytical methods by a factor of 4, but still implies (within a factor of 2 to 5) that 20% of all payloads deployed will re-encounter the Earth within 1 x 10^6 years. For the nominal mission model of 530 waste payloads, this is equivalent to re-entry of 116 waste payloads within the first 1 x 10^6 years. Using the most optimistic value for the range of uncertainty, this becomes 23 waste payloads or on the order of 1 x 10^5 kg of waste form returned. Each unprotected entering waste payload would liberate up to 1500 kg of waste form in a period of about 30 sec (14). The remainder would hit the ground and be buried. This is equivalent to a better than 95% probabilty that the first re-encounter will take place in the first 1000 years.

Accordingly, this destination cannot be considered stable; risk of Earth re-encounter is unacceptable and this destination will not be considered further.

DESTINATION 11. HELIOCENTRIC ORBIT (INNER) AT 0.85 AU

1) Mission Scenario Summary

This mission was chosen as the reference concept for space disposal of nuclear waste by the 1979-80 Battelle/MSFC study. Its characteristics have been used as the standard for evaluation of other concepts. The reference mission scenario uses a two-stage orbit transfer system: a cryogenic, re-usable OTV is used as the injection stage with a simpler, storable propellant second stage used for placement. Primary events after injection of payload and orbit transfer system into low Earth orbit include: (1) a primary velocity impulse maneuver which injects OTV and payload into the minimum energy transfer orbit; (2) separation of the waste payload attached to the placement stage; (3) if required, a midcourse correction performed by the placement stage at about injection plus 5 days to correct for injection errors; and (4) a second primary propulsion maneuver which leaves placement stage and payload in the destination heliocentric orbit.

Injection stage (OTV) recovery begins with a retroburn immediately following separation of the payload. This inserts the stage into an elliptical return orbit. An aerobraking maneuver at periapsis inserts the OTV into a phasing orbit; a burn at apoapsis of the phasing orbit raises periapsis to 260 km and is followed in half a revolution by LEO circularization. OTV recovery operations and shuttle landing complete the mission. Total OTV mission duration away from LEO is about 62 hours.

2) Destination Stability

Stability of the heliocentric orbit destination against Earth re-encounter has been analytically verified to 1×10^6 years (27).

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The selected injection trajectory allows the waste payload and associated orbit transfer systems to be rescued after failure at any point in the deployment sequence with sufficient time before re-encounter to allow multiple rescue attempts.

Due to Deployment System Failure. Capability for execution of multiple rescue attempts after deployment system failure at any point ensures mission continuation after deployment system failure.

Due to Particle Fallback After Meteoroid Strike. Fallback from the heliocentric orbit destination is negligible (14).

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with the heliocentric orbit destination is impossible.

4) Factors Related to Mission Cost

The delta-V from LEO to the 0.85 AU heliocentric orbit destination is 4.54 km/sec divided approximately into a 3.4 km/sec injection maneuver and a 1.3 km/sec heliocentric orbit placement maneuver. The 1.3 km/sec delta-V capability required for the placement stage results in a stage mass equal to about 70% of the waste payload mass; the payload mass delivered by this option (5075 kg) has been used in this study as a standard of comparison for the other destinations studied. Additional cost characteristics of the reference mission include:

- o Recovery of the injection stage for re-use
- o The placement stage is expended
- o Launch windows can be chosen any day at any time

This option has the lowest analytically verified risk level and except for the rejected Earth escape ellipse destination is the lowest cost option outside the geolunar system.

5) Other Factors

Retrieval from the heliocentric orbit destination is prohibitively expensive due to the delta-V required and duration of the retrieval mission; passive tracking is prohibited by the distance from the disposal site to Earth (a minimum of about 22 million km).

Implications for future use are not indicated. No alternate uses have been identified for the destination orbit.

6) Comments

Risk and cost factors make this the logical choice as a reference destination. The risk is as low as practically achievable, and the cost is lowest of viable deep space options considered.

DESTINATION 12. HELIOCENTRIC ORBIT (OUTER) AT 1.15 AU

The characteristics of destination 12 are substantially identical to those of destination 11.

DESTINATION 13. SUN/EARTH LIBRATION POINT "TADPOLE" ORBITS

Tadpole orbits about the L4 Sun/Earth libration points were investigated in Reference 5 which concluded that the orbits could be stable for long periods of time and might be accessible with delta-V's comparable to or lower than those for the reference heliocentric orbit. A further investigation was undertaken using Reference 4.

The chosen orbit passes within 75,000 km of the Sun/Earth L4 point and is representative of the most stable tadpole orbit configurations. Other configurations require lower delta-V's but exhibit reduced stability (17).

The mission scenario is similar to that for the heliocentric orbit destination but involves a circularization at about 1.0 AU rather than 0.85 AU. The two-stage orbit transfer system used is also similar. A cryogenic, reusable OTV is used as the injection stage with a simple, storable propellant upper stage used for placement. Primary events after insertion of payload and orbit transfer system into low Earth orbit by the uprated space shuttle include: (1) a primary velocity impulse maneuver which injects OTV and payload into an elliptical phasing orbit; (2) separation of the waste payload and placement stage; (3) if required, a midcourse correction at about injection plus 5 days to correct for injection errors; (4) an aphelion trim burn at perihelion of the transfer orbit at about injection plus 150 days (this burn fixes the aphelion radius to within about 0.001 AU in preparation for placement); and (5) a final placement burn at about injection plus 300 days which inserts the payload and expended placement stage into the destination "tadpole" orbit about the Sun/Earth L4 point.

Injection stage (OTV) recovery begins with a retroburn immediately following separation of the payload which injects the stage into an elliptical return orbit. An aerobraking maneuver at periapsis injects the OTV into a phasing orbit; a burn at apoapsis of the phasing orbit raises periapsis to 1260 km and is followed in half a revolution by LEO circularization. OTV recovery operations and shuttle landing complete the mission. Total OTV mission duration away from LEO is about 62 hours.

2) Destination Stability

The extensive investigations documented in Reference 17 indicate that "...the tadpole orbits are stable against perturbations... for at least 10^4 years. The lack of any change in the libration amplitude suggests that the orbits are stable for considerably longer periods." Further numerical integration of the type conducted in Reference 17 is recommended if further investigations show promise for this destination but there appears to be good reason to believe that these orbits will have stability at least equivalent to the reference heliocentric orbits.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. The selected injection trajectory allows the waste payload and associated orbit transfer systems to be rescued after failure at any point in the deployment sequence with sufficient time before re-encounter to allow multiple rescue attempts.

Due to Deployment System Failure. Capability for execution of multiple rescue attempts after deployment system failure at any point ensures mission completion after deployment system failure.

Due to Particle Fallback after Meteoroid Strike. Fallback from the tadpole orbit is impossible.

Due to Earth Intersection with the Nominal Destination Orbit. Earth intersection with the tadpole orbit is impossible.

4) Factors Related to Mission Cost

Typical velocity increments required for tadpole orbit injection vary from about 4.2 km/sec to 5.5 km/sec, depending on the exact tadpole orbit chosen, and constraining the time spent in transfer orbit to less than 1 year. A typical value would provide a velocity split identical to the reference heliocentric orbit destination, with a total delta-V of about 4.5 km/sec divided into a 3.4 km/sec injection maneuver and a 1.3 km/sec tadpole orbit placement maneuver. The 1.3 km/sec delta-V capability required for the placement stage results in a stage mass equal to about 70% of the waste payload mass; the payload mass delivered by this option (5075 kg) would be identical to that of the reference system. Additional cost characteristics of the reference mission include:

- o Recovery of the injection stage for re-use
- o The placement stage is expended
- o Launch windows can be chosen any day at any time

This option offers both cost and risk comparable to the reference system; some savings in delta-V are possible with further verification of tadpole orbit stability parameters.

5) Other Factors

Retrieval from the tadpole orbit destination would be very expensive due to the delta-V required and duration of the retrieval mission; passive tracking is prohibited by the distance from the disposal orbit to Earth (a minimum of about 80 million km).

Implications for future use are not indicated. No alternate uses have been identified for the destination orbit.

6) Comments

This option offers the best alternative to the reference heliocentric orbit destination among the deep space destinations investigated. Any savings in delta-V must be weighed against the increased navigational accuracy required for placement and the longer (300 to 330 day vs. 160 day) mission duration.

DESTINATION 14. SOLAR SYSTEM ESCAPE

1) Mission Scenario Summary

The mission profile for solar system escape is the simplest of the 14 studied. The orbit transfer system is a single stage expendable cryogenic OTV. The single primary event after insertion of payload and orbit transfer system into low Earth orbit by the uprated space shuttle is a 15-min injection burn which provides payload and OTV with the 8.75 km/sec delta-V required for insertion into the hyperbolic solar system escape orbit. The OTV is expended. No placement stage is required as the OTV is capable of achieving the required accuracy at initial injection.

2) Destination Stability

Stability of the payload against Earth re-encounter once injected into a hyperbolic escape orbit is absolute.

3) Long Term Risk of Earth Re-Encounter

Rescue Options. Because of the unique characteristics of the solar system escape mission, rescue for all possible failure modes of the orbit transfer system is impossible. Failure of the OTV propulsion system late in the injection burn could leave the payload on a pseudo-cometary Earth crossing orbit with a period ranging from one to hundreds or thousands of years. Rescue missions using orbit transfer systems similar to the deployment system would take similar periods to accomplish rescue. These durations, however, are far beyond the capability of existing or planned systems, making rescue impossible. In effect, the failed vehicle would simply outrun any pursuing rescue vehicle.

An equally dangerous failure mode would be failure of the onboard transponder, which would eliminate the ability to carry out the long term tracking required to determine the vehicle trajectory with any degree of accuracy. A subsequent (or simultaneous) propulsion system failure would leave the waste payload in an unpredictable Earth crossing orbit; lack of detailed trajectory data would make rendezvous and rescue impossible, regardless of the performance of the rescue system.

Due to Deployment System Failure. Probability of Earth re-encounter due to deployment system failure without rescue was evaluated for the solar system escape mission using data from Reference 5. The probability of Earth re-encounter within 1×10^6 years due to deployment system failure is calculated as 0.006 for a single mission (5). The probable number of Earth re-encounters for the 580 missions in the reference mission set becomes (0.006)(580) or about 3.5 payloads. For the reference payload of 3270 kg, this is equivalent to a rounded total deposition of about 10,000 kg. Even more important is the deposition rate. Instead of being distributed over a 1×10^6 year span, the waste payload would be deposited in the atmosphere in 3270 kg increments over a period of 30 sec. Three or more such events are probable in the million years following deployment. Efficient deposition is assured by an entry velocity of between 17 and 41 km/sec. Most entries would occur at velocities between 11.2 and 12 km/sec, with about 10% ranging upwards between 12 and 41 km/sec (5).

Due to Particle Fallback After Meteoroid Strike. Fallback from the hyperbolic escape orbit destination is impossible.

Due to Earth Intersection With the Nominal Destination Orbit. Earth intersection with the hyperbolic escape orbit is impossible.

4) Factors Related to Mission Cost

Total delta-V from LEO to solar system escape orbit is about 8.75 km/sec (7). The relatively high velocity increment reduces payload by about 1/3 compared to the reference destination. The primary additional influence on cost is the expending of a capable and expensive OTV with each mission.

Use of solar system escape as a space disposal destination would cost significantly more than disposal at the reference heliocentric orbit destination.

5) Other Factors

Retrieval and passive tracking are impossible; implications for future use are not indicated; significant contamination of interstellar space is impossible.

6) Comments

Both cost and risk of Earth re-encounter are significantly higher for solar system escape than for the heliocentric orbit reference. The increased risk illustrates the distinction between risk and mission reliability.

Mission Reliability, in the sense of the ratio of successful deployments to failed missions, is higher for solar system escape than for heliocentric orbit due to the extreme simplicity of the mission profile.

Mission Risk, however, is also higher due to the lack of rescue capability, so that even with the higher reliability per mission, more payloads re-encounter the Earth due to failures during the solar system escape mission. Failures during deployment to heliocentric orbit are prevented from resulting in re-encounter with high efficiency by use of the rescue option (4).

2.4 EVALUATION OF CANDIDATE DESTINATIONS

The destination characteristics shown in Section 2.3 were compared to provide relative evaluations of the worth of the destinations in the three primary areas of risk, costs, and implications for future use.

2.4.1 Evaluation of Long-Term Risk of Earth Re-Encounter. Long term risk is the primary evaluation criteria for destinations. Relative risk of the 14 destinations considered are summarized in Figure 2.3-1. Risk is expressed in terms of total expected return of waste form in the first million years following deployment and in terms of the estimated maximum deposition rate in kilograms per year for returning payloads. Numbers are based on inital deployment of 580 waste payloads.

Four destinations show a significantly greater risk than the heliocentric orbit reference destination. All offer the possibility of intact reentry of a waste form, an event capable of distributing on the order of 1000 kg of waste form into the atmosphere within a period of 30 sec. Three of the destinations (destination 7; Jupiter impact; destination 9; solar impact; and destination 14; solar system escape) are characterized by lack of a rescue capability to cope with failure of the deployment system.

Analyses of the consequences of deployment system failures are discussed in Section 2.3. Destination 10, the Earth escape ellipse, poses an excessive risk of Earth encounter due to the intersection of the nominal deployment orbit with the Earth's orbit. Studies discussed in Section 2.3 indicate that return of more than 10 waste payloads could be expected from the Earth escape ellipse destination in the first million years following deployment.

The remaining destinations are essentially equivalent in risk. Destination 1, lunar surface soft landing; destination 5, lunar orbit; destination 6, Venus impact; destination 8, asteroid soft landing; destinations 11 and 12, heliocentric orbits; and destination 13, Sun/Earth tadpole orbits are characterized by essential zero return of waste payload in the first million years following deployment. For three destinations in the Earth-Moon system (destination 2, high Earth orbit; destination 3, retrograde high Earth orbit; and destination

4, orbits about the Earth-Moon Trojan points), the expected return is on the order of 10 kg over a million years due to particle fallback resulting from meteroid fragmentation of one or more of the orbiting payloads. The estimated maximum deposition rate per year due to this fallback, however, is limited to approximately 10 mg/yr; a level far below the threshold of observable affects (14).

2.4.2 Evaluation of Cost Related Factors. Cost factors were evaluated primarily as a differentiator for risk equivalent destinations. Primary inputs to the cost evaluation included staging options, reuse options, required launch rates and payload deployed per mission, a factor strongly dependent on the delta-V required to reach a particular destination.

The objective was to use these inputs to provide a qualitative comparison of the relative cost for disposal at the 14 destinations. The approach used was first to determine a factor proportional to the number of missions required to deploy the reference mass of payloads to a given destination and then to determine a factor proportional to the cost of each mission. These factors were combined to yield an overall cost factor for each destination which was then normalized to the cost of delivery to the reference 0.85 AU heliocentric orbit.

The first step was to determine a factor proportional to the number of missions required for delivery of waste payload to a given destination. This factor was termed the mission quantity factor, or MQF, and was derived by dividing the reference payload by the payload deployed per mission for each of the 14 destinations. Each destination showed a slightly different payload deployed per launch of a single uprated shuttle; the differences were primarily due to the delta-V differences between missions. Values for the payload deployed per mission were obtained from past studies or calculated as required. Values shown in Figure 2.4-1 range from a low of 0.86 for the relatively low delta-V Earth/Moon Trojan point destination to a high of 3.01 for the high delta-V solar system escape mission with its proportionally smaller payload deployed per mission.

The next step was to determine a factor proportional to the cost of each individual mission. Costs determined were then normalized to a reference mission

DESTINATION	FACTORS RELATED TO COST PER MISSION				MCF (ΣFN)	MQF	QCF (MCFXMQF)
	F1	F2	F3	F4			
1. LUNAR SURFACE	0.42	0	0	0.71	1.13	1.83	2.07
2. HEO POSIGRADE	0.13	0	0	0.71	0.84	0.90	0.76
3. HEO RETROGRADE	0.13	0	0	0.71	0.84	1.00	0.84
4. E/M TROJAN	0.13	0	0	0.71	0.84	0.86	0.72
5. LUNAR ORBIT	0.13	0	0	0.71	0.84	0.94	0.79
6. VENUS ENTRY	0.13	0.08	0.18	0.71	1.11	0.88	0.98
7. JUPITER ENTRY	0.39	0.08	0.11	0.71	1.29	1.54	1.99
8. ASTEROID	0.13	0.16	0.13	0.71	1.13	1.20	1.36
9. SUN (NEP)	2.63	0	0	0.71	3.34	1.00	3.34
10. EARTH ESCAPE ELLIPSE	0.13	0	0	0.71	0.84	0.84	0.71
11. H/C INNER	0.13	0.16	0	0.71	1.00	1.00	1.00
12. H/C OVTER	0.13	0.16	0	0.71	1.00	1.04	1.04
13. S/E TROJAN	0.13	0.18	0	0.71	1.03	1.20	1.24
14. SSE	0.39	0	0	0.71	1.10	3.01	3.31

Figure 2.4-1. Qualitative Cost Factor Determination

cost of $38 million. Four primary mission cost factor areas were identified. The first factor, F1, was determined as a function of reuse options and stage complexity. Values normalized to the reference system total mission costs are shown in Figure 2.4-1 and range from 0.13 for full reuse of the injection stage to a high of 2.63 reflecting the high individual cost of the nuclear electric propulsion system used for the solar impact mission.

F2, the second factor, reflected normalized cost of the placement stage used on some missions. Placement stages were differentiated on the basis of navigation system complexity and whether or not they incorporated a primary propulsion system. Normalized values for F2 shown in Figure 2.4-1 range from 0.08 to 0.16, illustrating the relatively low cost of placement stages relative to total mission cost.

F3, the third cost factor, reflected the facilities impact cost required by the higher launch rates needed by some destinations, distributed over the total number of missions and normalized to the total mission cost. Distributed facilities impact for the most restrictive launch windows considered amounted to approximately 20% of the cost per mission.

F4, the final factor evaluated, reflected launch costs and was common to all options, reflecting a $27 million launch cost. When normalized this represents 71% of the total reference mission cost, and when compared to the other cost factors illustrates the dominance of launch cost in the nuclear waste disposal mission.

Cost factors F1 through F4 were then added to obtain a single composite mission cost factor (MCF). Mission cost factors are shown in Figure 2.4-1. Most values are fairly close to the reference value, illustrating again the dominance of launch cost. The only real standout in cost per mission is the solar impact option reflecting the influence of a $100 million average cost of the nuclear electric propulsion stage.

Mission cost factors were then multiplied by the mission quantity factor to obtain the qualitative cost factor (QCF) shown in Figure 2.4-1. Qualitative cost factors range from a low of about 70% of the reference system cost factor to a high of about three times the reference system cost for the solar system escape and solar impact destinations.

All together, when compared to the reference destination 11, six destinations imply significant increases in deployment costs. Destination 1, lunar surface, is penalized by a relatively low payload per mission and the requirement to expend the orbit transfer vehicle. Destination 7 deploys a relatively low payload per mission due to the high delta-V involved. The same high delta-V requirement requires expending the injection stage which also adds to the cost. Destination 8 combines the disadvantages of a high launch rate, an expendable placement stage, and a payload smaller than the reference mission payload. Destination 9, solar impact, is heavily penalized by the high cost of the nuclear electric injection stage. Destination 13, orbits around the Sun/Earth Trojan points, has costs comparable to but slightly higher than the reference due to the slightly lower payload per mission. Destination 14, solar system escape is severly penalized by the relatively low payload deployed per mission which requires a total of almost three times the number of flights needed for deployment to the reference heliocentric orbit destination.

Three destinations in the geolunar system show the potential for significant savings in deployment cost. Destination 2, posigrade high Earth orbit, shows a cost factor only three-quarters of that for the reference mission due primarily to increased payload deployed per mission, combined with the ability to recover and reuse the injection stage and the savings involved in not expending a placement stage. Essentially the same reasons account for the similar cost effectiveness of destination 4, orbits about the Earth/Moon Trojan point, and destination 5, lunar orbit.

2.4.3 Evaluation of Other Factors. Factors besides cost and risk investigated in evaluation of destinations included implications for future use, which was ranked with cost and risk as a primary differentiation criteria; and the capability for passive monitoring from the Earth's surface or low orbit of deployed payload position and velocity. A third factor was the ability to retrieve a deployed payload rapidly and economically, which was considered as being potentially useful but of secondary importance in destination differentiation. Only one of the destinations studied, destination 6, Venus impact, posed serious contamination problems, combining identified potential uses for the destination with the potential for wide spread contamination.

Four options in the geolunar system offer straightforward recovery options. Posigrade and retrograde high Earth orbits, orbits about the Earth/Moon Trojan points, and lunar orbit share the potential for relatively low cost rapid recovery of waste forms.

The same locations are close enough to the Earth to make passive position monitor using Earth based laser radar or low Earth orbit based infrared telescopes possible. This passive tracking is a primary factor enabling waste form recovery.

2.5 CANDIDATE DESTINATION SELECTION

Final screening to select destinations for the study used the characteristics defined in Section 2.4 and the primary criteria of risk, cost, and implications for future use.

The criteria were weighted as follows:

1. Risk. Destinations showing significantly increased risk compared to the reference 0.85 AU heliocentric orbit destination were rejected for further consideration.

2. Cost. Destinations showing significantly increased cost compared to the reference destination were rejected.

3. Implications for future use. Destinations evaluated as having negative implications for future use were rejected.

Results of the screening are illustrated in Figure 2.5-1.

CANDIDATE DESTINATION	RISK SCORE	COST SCORE	IMPL FOR FUTURE USE	INITIAL SCREENING RESULTS	FINAL SCREENING
1) LUNAR SURFACE	1	X	1		
2) HEO POSI	1	1	1	HEO POSIGRADE	
3) HEO RETRO	1	1	1	HEO. RETROGRADE	
4) E/M TROJAN	1	1	1	EARTH/MOON TROJAN ORBITS	EARTH MOON TROJAN ORBITS • ANALYTIC STABILITY • REPRESENTATIVE OF GENERAL GEOLUNAR DESTINATION
5) LUNAR ORBIT	1	1	1	LUNAR ORBIT	
6) VENUS	1	1	X		
7) JUPITER	1	X	1		
8) ASTEROID	1	X	1		
9) SUN	X	X	1		
10) EARTH ESCAPE ELLIPSE	X	1	1		
11) H/C INNER	1	1	1	HELIOCENTRIC ORBIT 0.85AU	HELIOCENTRIC ORBIT 0 85AU • BEST CHARACTERIZED • EXISTING REF FOR COMPARISON
12) H/C OUTER	1	1	1	HELIOCENTRIC ORBIT 1 15AU	
13) S/E/ORBITS	1	X	1		
14) SSE	X	X	1		

Figure 2.5-1· Candidate Destination Screening

In the columns marked risk score, cost score and implications for future use, a 1 indicates acceptability, and an X indicates that the destination was unacceptable in the category indicated. The screening identified four candidates in the geolunar system satisfying all criteria: posigrade and retrograde high Earth orbits, orbits about the Earth/Moon Trojan points, and orbits about the Moon itself. Two destinations are identified in deep space which satisfied all criteria: heliocentric orbits at 0.85 AU and heliocentric orbits at 1.15 AU. Review of the characteristics tabulated in Section 2.4 indicates the characteristics of destinations in both geolunar system and deep space were similar enough to allow for selection of a single geolunar destination and a single deep space destination as representative of all those considered acceptable.

In the geolunar system the Earth/Moon Trojan orbits were selected as a representative destination. Among the geolunar destinations considered, Earth/Moon Trojan orbits demonstrate the best stability and have delta-V's and mission durations comparable to the other three.

The heliocentric orbit at 0.85 AU was selected as the reference deep space destination. This destination is the best characterized as a result of the work performed by the Marshall Space Flight Center in previous studies and is identical with the existing reference destination which facilitates comparison of alternate space system concepts in other areas.

2.6 CONCLUSIONS AND RECOMMENDATIONS

1. A top down survey of all reasonable destinations showed none with less risk than the reference destination.

2. Several destinations in the geolunar system offer the possibility of significant costs savings with risk equivalent to the risk of the reference system.

Recommendations include:

1. Continuation of the 0.85 AU radius heliocentric circular orbit as reference destination for the study. This choice provides verified stability and the best long term risk level identified for any destination considered.

2. Evaluation of the geolunar system destination as an alternate. This choice offers equivalent long term risk level for the cermet payload carried as a baseline in the study, and provides potential benefits including better rescue options which could act to reduce the overall system risk of Earth re-encounter. Life cycle costs are lower due to use of a better performance vehicle, lower total delta-V, the ability to use a fully reusable orbit transfer system, and the elimination of separate development efforts for the placement stage.

3.0 MISSION ANALYSIS AND OPERATIONS

3.1 INTRODUCTION

This chapter is divided into three sections. The first, analysis of delivery mission profiles, describes the characterization of the missions required to deliver payloads to the 0.85 AU heliocentric orbit destinations selected in the destination task. The second part describes our analysis of mission operations for the selected missions, and the third, the studies of mission control for the space disposal mission selected.

3.2 DELIVERY MISSION ANALYSIS

Objectives of the mission analysis effort were to provide mission profiles in terms of events sequences, time lines and performance requirements for the delivery mission. Key mission profile events which are shared by all chemical propulsion orbit transfer system options are: injection stage (OTV) recovery and coast and placement. A separate mission analysis was provided for the solar electric orbit transfer system options.

3.2.1 Injection Mission Profile. The injection mission profile is illustrated in Figure 3.2-1. The profile shown is used with a recoverable OTV, but the injection hyperbolic escape orbit is equally applicable to expendable vehicle options. Key events in the injection mission profile are numbered from 1 through 6 in the illustration. First event is the injection burn into the Earth escape hyperbolic orbit followed by event 2, separation of the solar orbit insertion or placement stage, followed by a retroburn of the orbit transfer vehicle injection stage which places it into the elliptical return orbit. A third maneuver at apoapsis of the return orbit adjusts the periapsis altitude to the correct value for the aerobraking pass. Coast is followed by an aerobraking maneuver which provides the bulk of the velocity increment required for insertion to low orbit. The final maneuver, shown as 5, raises the apoapsis to the 160 nmi value required for rendezvous with and recovery by the space shuttle orbiter. A final small burn, event 6, circularizes the orbit transfer vehicle in the recovery low Earth orbit. A detailed discussion of the injection mission profile and its derivation is contained in Appendix D.

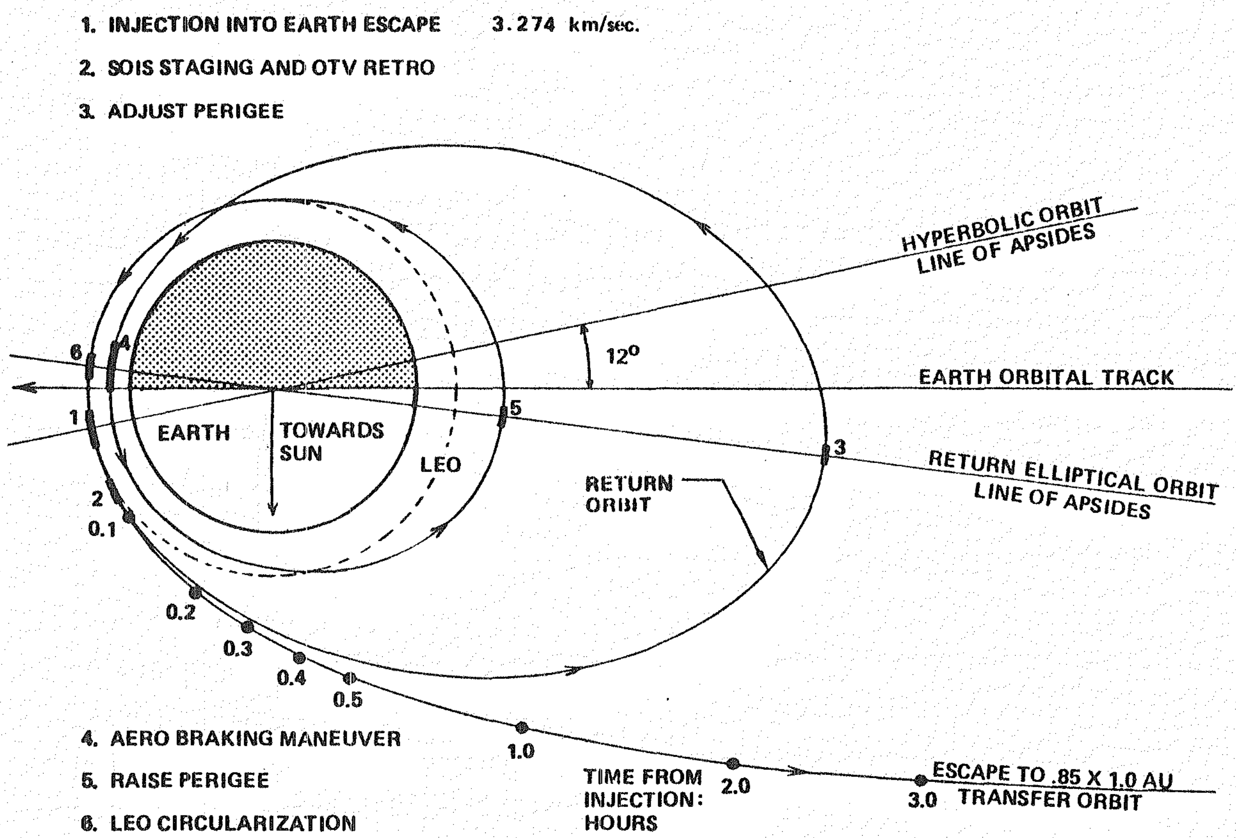


Figure 3.2-1: Injection Mission Profile Schematic

3.2.2 <u>OTV Recovery Option Trades</u>. Recovery of the injection stage following solar orbit insertion stage injection into heliocentric transfer orbit can be an important contributor to reducing the cost of the disposal mission. A review of past references was unable to uncover definitive data on optimization of OTV recovery trajectories. The trajectories for recovery of a vehicle using aerobraking were also undefined for return from injection to hyperbolic escape orbits.

Accordingly a study was conducted to define optimum recovery modes for both all propulsive and aerobraked OTV recovery options. Several issues are involved in characterizing the return orbit. There is a discrete amount of time required to separate the OTV from the solar orbit insertion stage and reorient it for the retro maneuver. This time has an impact on the magnitude and the direction of the velocity change (delta-V). In addition the total delta-V required for the return to LEO is affected by the magnitude of retro delta-V.
Two different methods of performing the braking maneuver on a hyperbolic orbit

were analyzed:

1. Tangential delta-V in which the braking velocity vector is applied tangent to the orbital path.

2. Co-apsidal delta-V in which the braking velocity vector is oriented such that there is no shift in orbit line of apsides.

A comparison of total return delta-V as a function of return orbit period for the two options resulted in selection of the tangential delta-V option as the most economical of delta-V. The comparison is treated in detail in Appendix F, Orbit Transfer Vehicle Return Trajectory Trades.

With the optimum return trajectory option defined, studies were conducted to optimize the return orbit period for minimum mission cost. The key trade is between payload mass injected, which increases with longer return orbit periods due to low return delta-V, and launch costs which increase with longer return orbit periods due to orbiter on orbit stay time charges of $0.5 million per day.

Results of the study are summarized in Figure 3.2-2. An estimate of total mis-

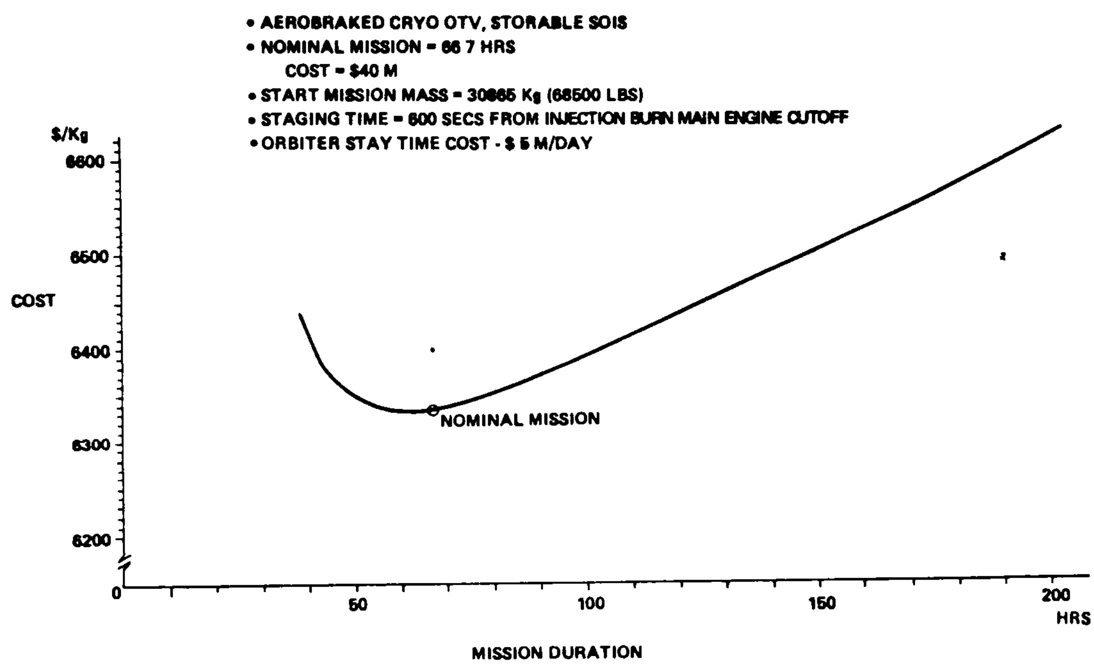


Figure 3.2-2: Return Orbit Period Optimization

sion cost per payload delivered to heliocentric orbit destination in dollars per kilogram is plotted on the vertical scale. The horizontal scale shows mission duration in hours. Minimum cost is shown at about 60 hours mission duration. An inspection of the scale shows that this is not a strong maximum and return mission period may be chosen any time between 40 and 100 hours with little overall penalty.

The duration chosen was 62 hours which is very close to optimum and which matches the capability of the baseline orbit transfer vehicle studied in the OTV Phase A Concept Definition Study, allowing use of a standard unmodified OTV for the injection mission.

3.2.3 Placement Mission Profile. Figure 3.2-3 schematically illustrates the placement mission profile. The injection mission places the solar orbit insertion stage and payload into a Hohmann transfer orbit to the 0.85 AU heliocentric orbit placement location. Primary events following injection include an optional trajectory trim maneuver at about injection plus 10 days to correct for injection inaccuracies. An approximately 165 day coast to periapsis of the transfer orbit is followed by orientation and the placement burn of 1.283 km/sec at the 165 day point. This mission profile is common to both storable propellant and cryogenic propellant solar orbit insertion (placement) stages.

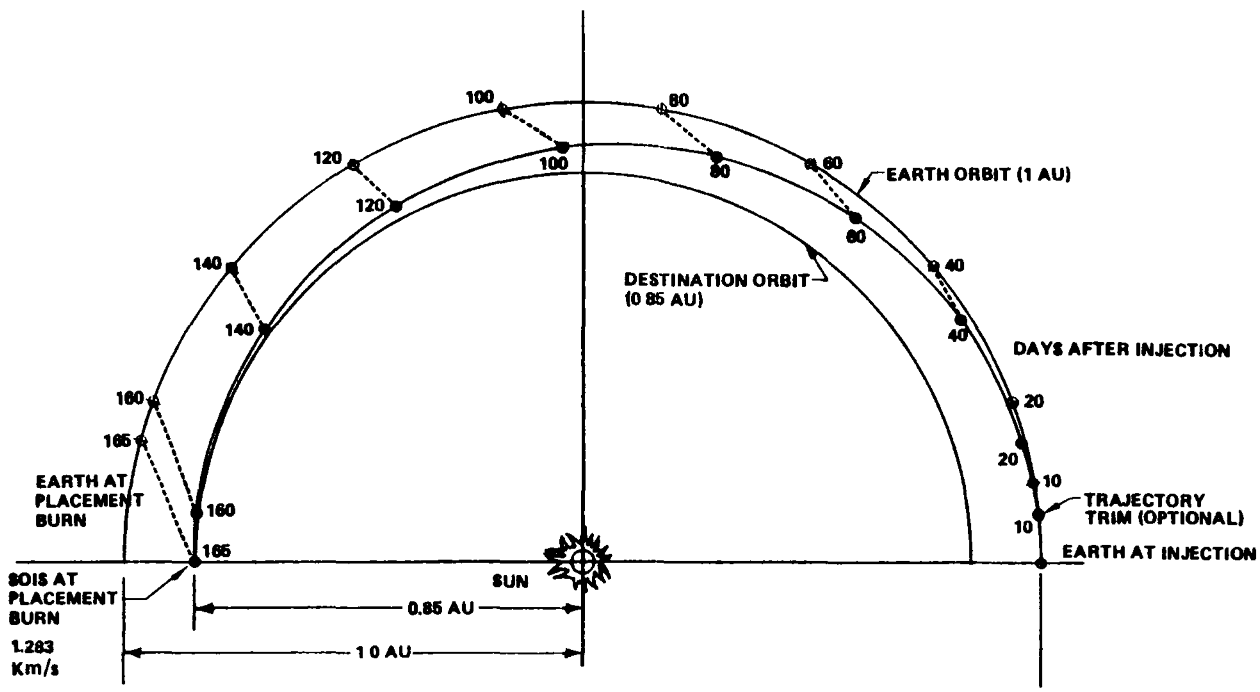


Figure 3.2-3: Placement Mission Profile Schematic

3.2.4 Solar Electric Stage Mission Profile. The mission profile for the solar electric stages as illustrated in Figure 3.2-4 differs significantly from the profile for chemical stages described in the previous section in terms of both duration, performance requirements and operation sequence.

LEO ORBIT: 160NM @ 28.5°	
TIME TO 0.85AU	545 DAYS
ΔV TO ESCAPE	7.728 Km/SEC
ΔV ESCAPE → 0.85AU	2.521 Km/SEC
	10.249 Km/SEC

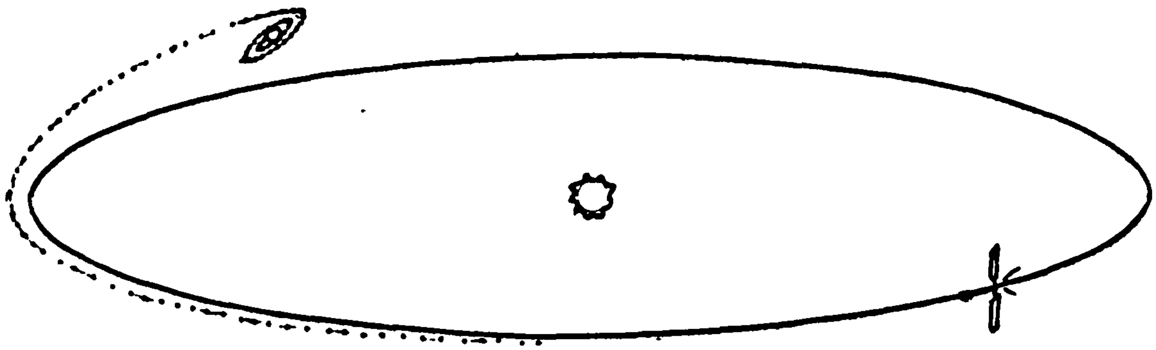


Figure 3.2-4· Solar Electric Stage Mission Profile Schematic

The baseline mission studied for the electric stage involves the following key events:

1. A multiturn spiral from lower Earth orbit toward the escape injection involving a delta-V of approximately 7.7 km/sec over a period of about 1 year.

2. Approximately two-thirds turn spiral in to the heliocentric orbit destination involving a delta-V of about 2.2 km/sec applied over a duration of 180 days. Total mission duration is about 545 days with a total delta-V of about 10.25 km/sec. Increased delta-V for this option is largely due to G losses suffered during the 1-year spiral to Earth escape injection.

3.3 RESCUE MISSION ANALYSIS

3.3.1 Introduction. Rescue mission analysis was accomplished for high Earth orbit and deep space rescue missions to provide definition of: (1) abort modes, (2) trajectories for rescue, and (3) the performance requirements in terms of delta-V and duration required to implement the rescue missions. An initial screening aimed at focusing the effort spent on rescue mission analysis divided rescue missions into two categories shown in Figure 3.3-1.

NOMINAL: 99%

- VEHICLES ARE FAIL SAFE
- OTV OR SOIS FAIL BUT NOT BOTH
- TARGET VEHICLE IS ACTIVE FOR RENDEZVOUS AND PAYLOAD TRANSFER
  - ACTIVE BEACON FOR RENDEZVOUS
  - ATTITUDE STABILIZED

CONTINGENCY 1%

- VEHICLES FAIL CATASTROPHICALLY
- BOTH FAIL
- TARGET VEHICLE IS DEAD
  - NO BEACON
  - TUMBLING
  - DAMAGED

- VEHICLE DESIGN REQUIREMENTS DICTATED BY RESCUE
  - VEHICLES ARE FAIL SAFE
  - SOIS ACS, XPONDER GOOD FOR 2 X NOMINAL RESCUE MISSION DURATION

Figure 3.3-1: Rescue Mission Functional Categories

For reliabilities typical of existing space systems, approximately 99% of all failures will fall into the nominal failure categories shown. This requires small design penalties in that the stages must be fail safe and transponder life must be equal to at least two times the duration of the maximum rescue mission.

The remaining 1% of rescue missions pose a more formidable challenge, but it is safe to assume that basic rescue systems will be the same in terms of performance and most operations. Key operations which would differentiate contingency rescue operations from nominal rescue operations would include rendezvous with noncooperative targets and recovering of tumbling, massive vehicles. Further study of these subjects is required but is beyond the scope of the current effort and not directly related to differentiation of alternate space system concepts. Accordingly, in this study efforts were devoted to consideration of the nominal rescue mission only. Nominal rescue conditions cover 99% of the failures and the system evolved for dealing with the nominal rescue mission will serve as the foundation for coping with the remaining 1%.

Resulting assumptions which govern the rescue study include: (1) treat nominal failures only, injections or placement stage failure but not both; (2) placement stage is available for abort function if the injection stage failures; (3) the NASA deep space network or equivalent is available to support the rescue mission task; and (4) use of delivery systems in the rescue mission is to be maximized.

3.3.2 Definition of Abort Options. Abort mode definition was required for two reasons: first, to find the best way to assure waste payload placement in a safe orbit following injection stage failure; second, to find the characteristics of the resulting orbits to provide initial conditions for rescue missions.

Results of the abort option study are illustrated in Figure 3.3-2, which shows the abort options available for failures of the injection stage as a function of injection stage propellant consumption during the injection burn. Primary quantities plotted as a function of percent of propellant consumed include: (1) delta-V required to continue to transfer orbit insertion; (2) delta-V required for circular orbit insertion. Also plotted are failure orbit apoapsis altitude and the delta-V from the initial low Earth orbit and the placement stage delta-V capability.

A logical abort strategy is suggested: (1) Up to about 55% propellant depletion injection stage failure would result in the placement stage being used to place payload and placement stage in a circular abort orbit. Maximum altitude of this abort orbit would be about 40,000 km. (2) Beyond 55% propellant deple-

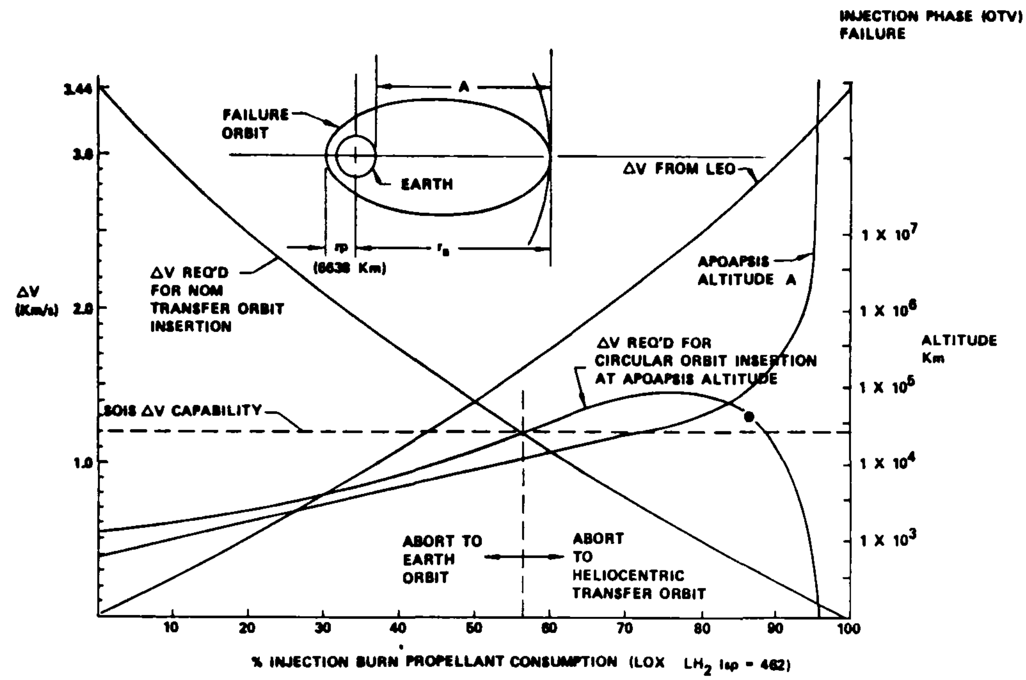


Figure 3.3-2: Analysis of Abort Options

tion failure of the injection stage would result in the placement stage being used to complete injection into the heliocentric transfer orbit, establishing initial conditions for subsequent deep-space rescue.

Accordingly, for the nominal rescue cases studied, only two basic rescue mission profiles need be considered: (1) deep space rescue with the target vehicle in a 0.85 x 1.0 AU transfer orbit ellipse, resulting either from contingency injection using the placement stage following injection stage failure or from a failure of the placement stage propulsion system during the circularization burn at the end of the initial transfer; and (2) high Earth orbit rescue, with the target vehicle in a circular or slightly elliptical holding orbit and an orbital altitude ranging between 500 and 40,000 km.

3.3.3 Rescue Mission Trajectories. The study of abort mode options provided definitions of rescue mission initial conditions; either a circular Earth orbit at an altitude of 40,000 km or below, or an elliptical heliocentric transfer orbit with apoapsis between 1.0 and 0.85 AU and periapsis at 0.85 AU. Rescue mission trajectories were defined for both Earth orbit and deep-space rescue locations (4).

Figure 3.3-3 illustrates the reference trajectory for rescue of vehicles stranded in earth orbit. Launch opportunities exist once a day. Figure 3.3-4 illustrates the 3-burn deep space rescue mission trajectory. This profile provides for rendezvous with the malfunctioning vehicle at its second perhihelion and offers reduced delta-V when compared to 2-impulse transfers. This trajectory is applicable to a wide range of solar orbit insertion stage (SOIS) failures and provides for maximum mission times of under 2 years.

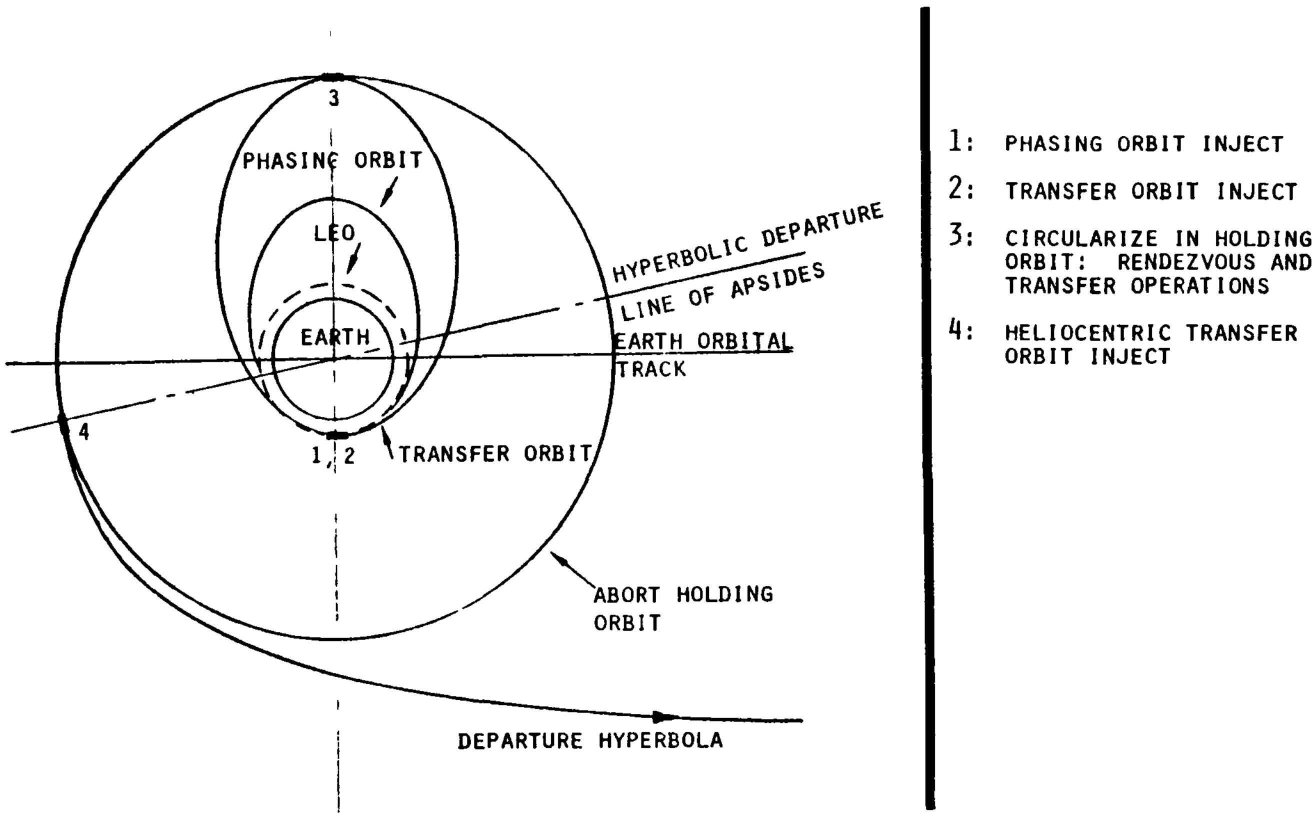


Figure 3.3-3: High Earth Orbit Rescue Trajectory Schematic

3.3.4 Rescue Mission Performance Requirements. Performance requirements are specified in terms of delta-V and for both Earth orbit and heliocentric rescue mission trajectories. The payload mass to be thrown is described in Task 6.

Performance required for rendezvous in Earth orbit was established by an analysis of abort options for failures during the injection burn. Two primary options were identified; abort to Earth orbit and abort to heliocentric transfer orbit. The first option encompasses using the SOIS to place the waste payload in a circular storage orbit around the Earth. Maximum radius for circular

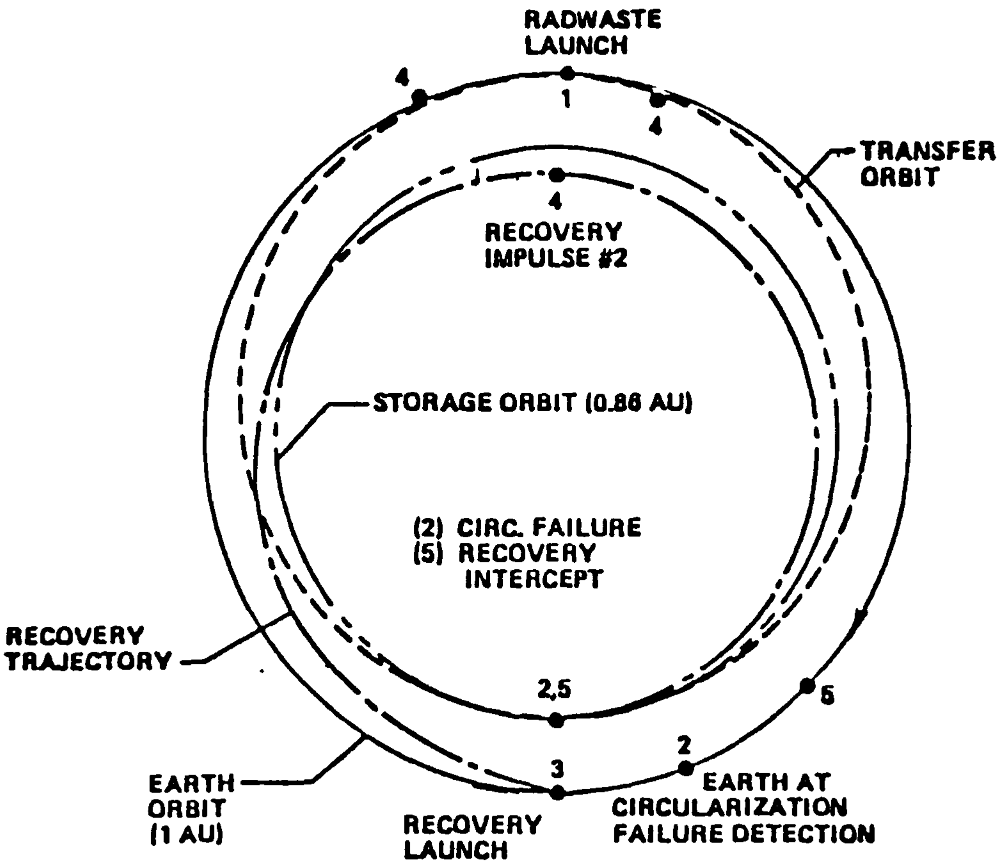


Figure 3.3-4: Deep Space Rescue Trajectory Schematic

storage orbits is about 4 X 10^4 km; failures which would result in larger radii are more easily handled by abort to transfer orbit. Delta-V's required for rendezvous with the failed vehicle in the higher circular orbits are approximately 3.2 to 3.5 km/sec; subsequent to waste payload transfer to the rescue vehicle, injection takes approximately 2.0 km/sec, followed by a standard placement. Performance requirements for deep-space rescue missions are bounded by the case of total SOIS failure. For a typical 3-impulse transfer (Figure 3.3-4), injection delta-V to a 0.85 AU perihelion is in the range of 3.5 km/sec, with two intercept delta-V's of approximately 1.2 km/sec each used for rendezvous with the target at the target's second perihelion, followed by a final 1.18 km/sec placement burn. Total rescue mission duration from launch to placement is 0.8453 years or about 308 days. This compares favorably with the 6 km/sec injection, 1.46 km/sec rendezvous, 1.18 km/sec placement delta-V's, and 1.292 year mission duration specified in Reference 4 (Case 3A).

3.4 OPERATIONS ANALYSIS

3.4.1 Introduction. The objective of the operations analysis task was to define system operations to the level required for system design support and to provide standards for qualitative differentiation between alternative concepts in the areas of mission complexity and risk evaluation.

Our approach was to define a comprehensive top level operations flow encompassing (1) launch vehicle options using one or two launch vehicles per mission, (2) orbit transfer system staging and reuse options, and (3) shield removal options. This top level flow was used as a basis for definition of key lower level operations flows as required. An example would be definition of OTV and SOIS separation operations to provide an input for the OTV return mission optimization. The flow as defined is generic; detail operational flows specific to total system options selected in Task 8 are presented in Task 8. This section presents the details of the generic top level flow and documents the detail definition developed for injection and placement stage separation operations.

3.4.2 System Level Composite Operations Flow. Figure 3.4-1 illustrates the top level operational flow for space disposal of nuclear waste which was evolved as a tool for comparison of configurations and as a framework for location of areas needing further definition. Operations are coded as follows:

- 1.0 = Launch vehicle ascent operations
- 2.0 = Low Earth orbit operations
- 4.0 = Space transportation system landing operations
- 5.0 = Solar electric stage operations
- 6.0 = Orbit transfer vehicle operations (injection stage)
- 7.0 = Long-life OTV (LLOTV) operations
- 8.0 = SOIS (placement stage) operations

By correct choice of direction at the "or" branch points on the diagram any specific mission scenario can be defined in terms of the operations coded. Specific examples of complete operation flows for selected total space systems are shown in Section 8, System Integration and Evaluation.

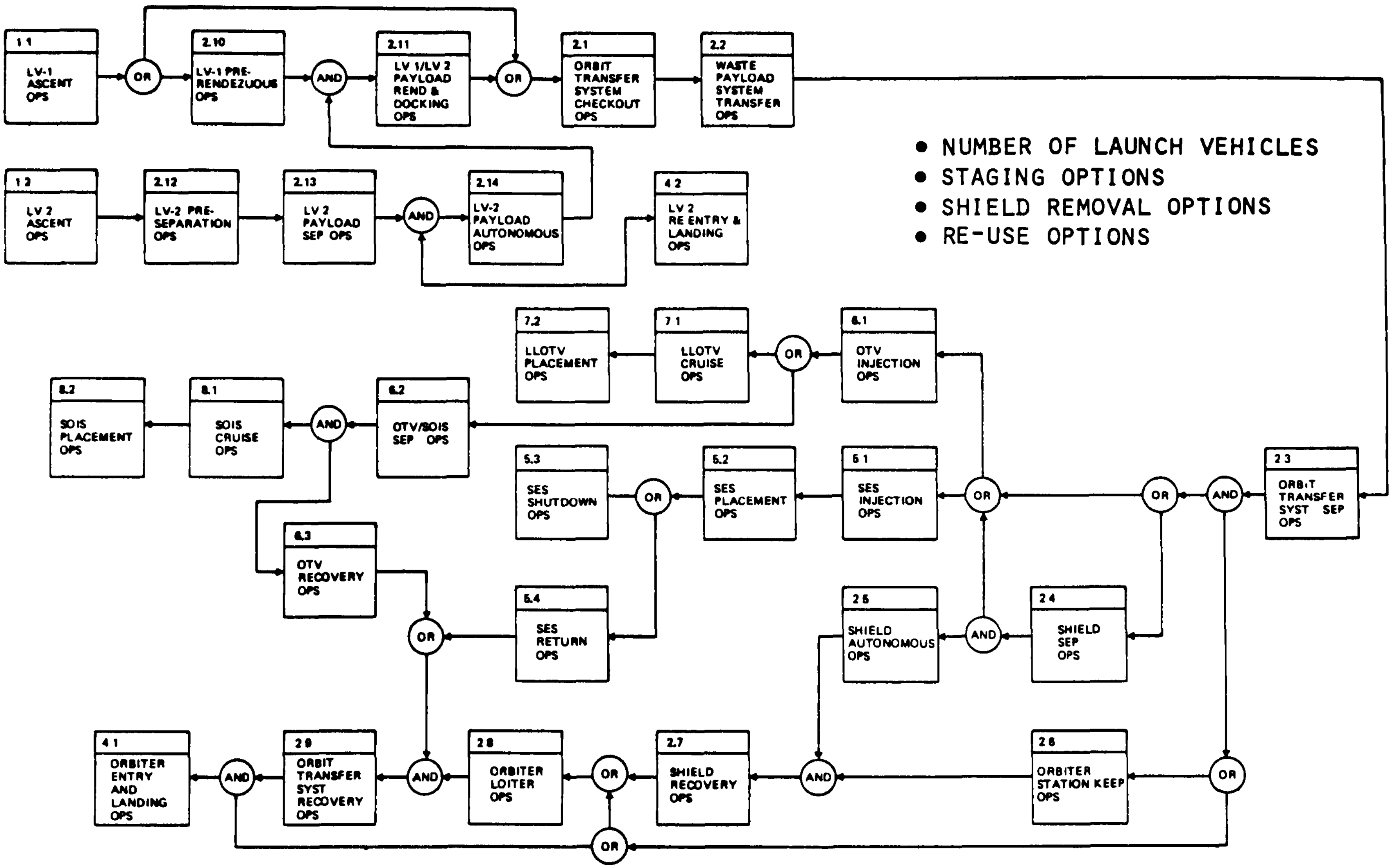


Figure 3.4-1: Space System Composite Operations Flow

3.4.3 Detailed Definition of Operations. Detailed definition was provided for OTV/SOIS separation to allow evaluation of separation delays on orbit transfer vehicle return trajectory delta-V requirements (see Section 3.2.2). Data was derived from similar timelines defined for the Boeing/USAF IUS.

The resulting operations flow is illustrated in Figure 3.4-2. The 6.2 in the title of the figure refers to the appropriate block in the composite operations flow (Figure 3.4-1). Separation events for the OTV begin at main engine cutoff

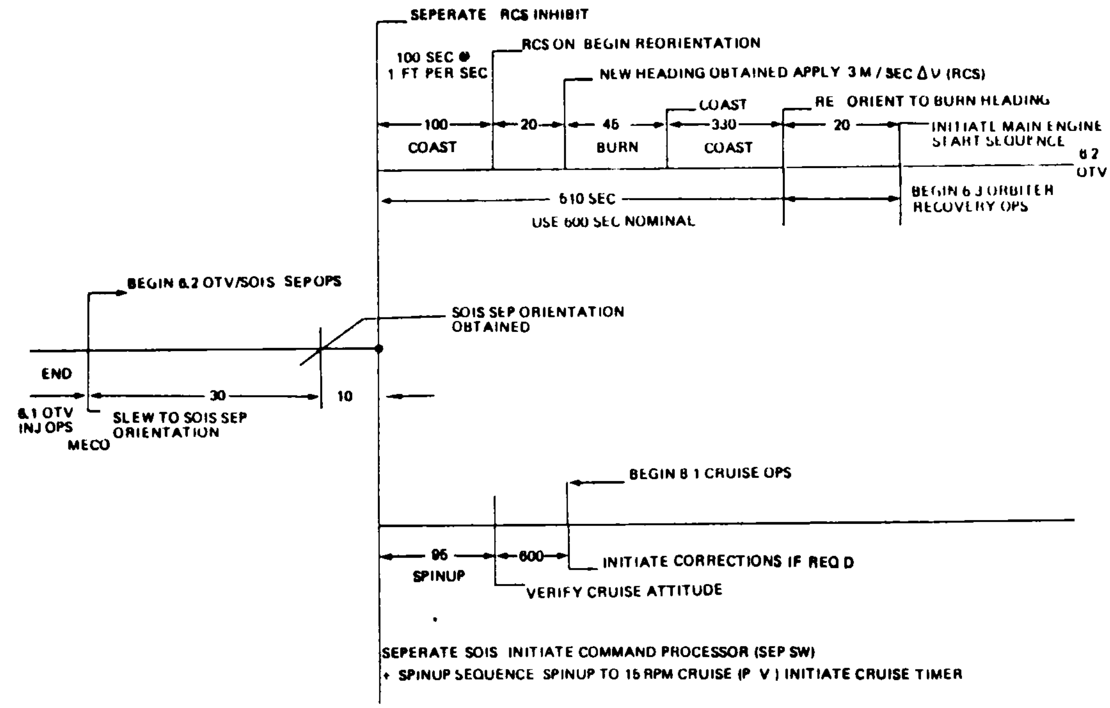


Figure 3.4-2. OTV/SOIS Separation Operations

(MECO). The vehicle is then slewed to the separation heading and separation nuts are fired, allowing the SOIS to be separated under impetus of separation springs. The same command which initiates separation acts to inhibit the OTV reaction control system. This is followed by a 100-sec coast by the OTV to a position clear of the separated solar orbit insertion stage. At the end of the coast period, the OTV reaction control system is enabled and a 20-sec slew to coast separation heading is accomplished. This maneuver is followed by application of a 3-m/sec delta-V using the reaction control system followed by a 330-sec coast to obtain clearance for main engine start. A 20-sec reorientation to the final burn heading is followed by initiation of the main engine start sequence. Total time elapsed during the separation operations is 660 sec or 11 min.

3.5 MISSION CONTROL

3.5.1 Introduction. Mission control issues for the launch vehicles, orbit transfer vehicles, and solar electric stage have been well defined in ongoing studies of the space transportation system, interim upper stage, and solar electric propulsion systems. Control requirements to meet SOIS, however, remain largely undefined. Earlier studies (19) present the SOIS avionics as essentially identical to the OTV avionics in including provisions such as closed circuit television and high data rate communications links useful only in the contingency situation of a rescue mission.

The available data base on vehicles such as Burner 2, IUS, and the Pioneer Venus Orbiter indicated the potential for a significant simplification of SOIS avionics when compared to the avionics carried by the OTV. The primary differences are in control requirements and reliability.

SOIS control requirements differ significantly from OTV and IUS requirements. SOIS navigation is ground based. It has no requirement for autonomous state vector updates or real time processing. Its primary requirement is for orientation control only.

SOIS reliability is not a primary risk issue. The stage operates only after solar system escape velocity is reached. Any conceivable failure mode after escape is reached leaves the failed stage and payload in an orbit with a mean time to Earth reencounter measured in thousands of years. Accordingly, the rescue mission is the primary backup for malfunction of the SOIS. The primary effect of SOIS reliability is to dictate the number of rescue missions. Accordingly, SOIS reliability and complexity is primarily a mission cost driver. Avionics cost impact due to increased reliability must be balanced against the decrease in cost resulting from a lower number of rescue missions required.

The thrust of this task became one of defining requirements for the SOIS control functions for use in Task 6 vehicle definition efforts.

3.5.2 SOIS Functional Requirements. SOIS functional requirements are illustrated by mission phase in Figure 3.5-1 for both spin stabilized and three axis

MISSION PHASE	SPIN STABILIZED	3-AXIS STABILIZED
6.2 OTV/SOIS SEPARATION	• SPIN-UP • VERIFY CRUISE ORIENTATION	• VERIFY ATTITUDE REFERENCE ACQUISITION • INITIATE ATTITUDE HOLD IN CRUISE ORIENTATION
8.1 SOIS CRUISE OPERATIONS	• MAINTAIN CRUISE ATTITUDE • PROVIDE CONTINGENCY TRAJECTORY TRIM	• MAINTAIN CRUISE ATTITUDE • PROVIDE CONTINGENCY TRAJECTORY TRIM
8.2 SOIS PLACEMENT OPERATIONS	• PERFORM PITCHOVER TO BURN ORIENTATON • VERIFY ORIENTATION • PERFORM BURN • ALLOW ORBIT ELEMENT VERIFICATION	• PERFORM YAW TO BURN ORIENTATION • VERIFY ORIENTATION • PERFORM BURN • ALLOW ORBIT ELEMENT VERIFICATION

Figure 3.5-1: SOIS Functional Requirements Summary

stabilized SOIS vehicles. Separation requirements for the spin stabilized vehicle include spin up to the cruise r/min of about 15 r/min and verification of cruise orientation by downlink to the ground. Requirements differ for three axis stabilized vehicle during separation. Primary requirements for the three axis stage include preseparation verification of attitude reference acquisition after initial orientation by the injection stage by downlink to ground control, and initiation of attitude hold using reaction control system in cruise orientation following separation from the injection stage.

Cruise and placement operation requirements are identical for spin stabilized and three axis stabilized stages. Cruise operations include maintenance of cruise attitude, and the ability to provide contingency trajectory trim to compensate for injection errors.

Placement operations include maneuvering to the correct burn orientation, verification of the correct orientation by ground control using the telemetry link, performance of the placement burn, and providing a continuous signal to allow post burn tracking verification of the elements of the destination orbit.

3.5.3 SOIS Attitude External References. Attitude control during all mission phases is a prime requirement identified for the SOIS. Characteristics of the heliocentric transfer orbit followed by the SOIS (Figure 3.2-3) were examined to determine appropriate external references for SOIS attitude control. Figure 3.5-2 illustrates the references chosen. The primary SOIS reference is the Sun. The Sun-SOIS line is established by Sun sensors on the vehicle. Orientation about the Sun-SOIS line is provided by star sensors locked on to a suitable star such as Canopus.

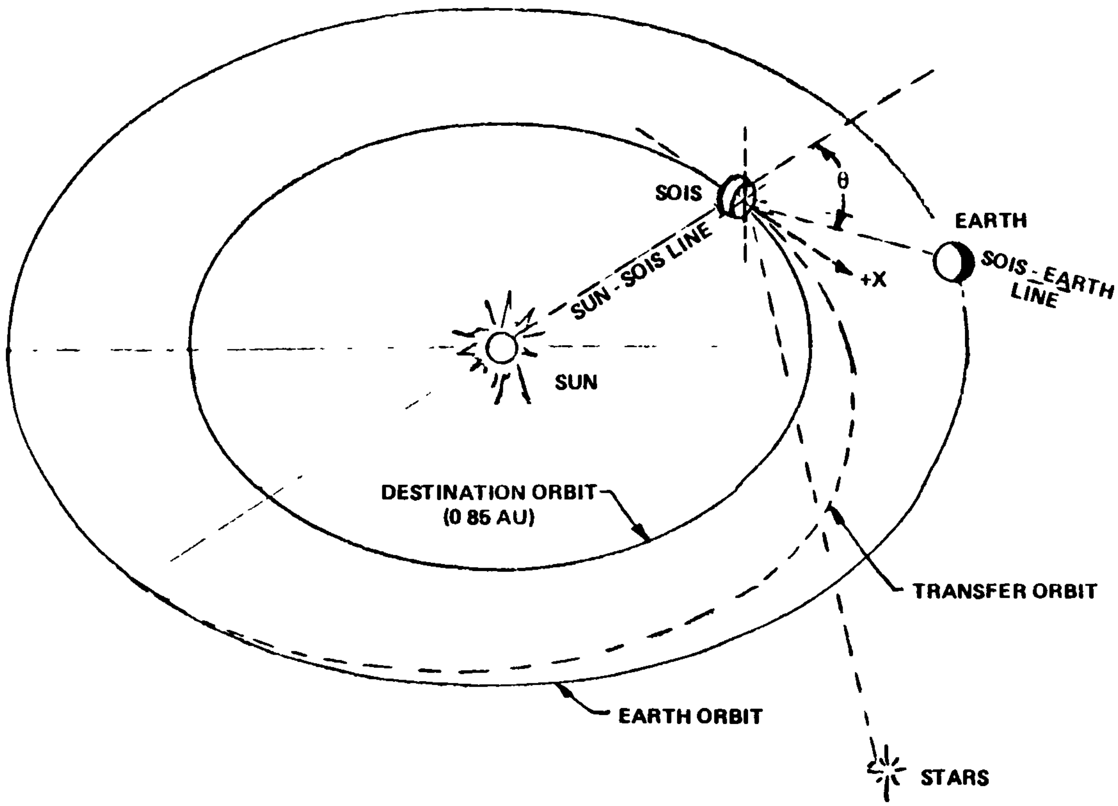


Figure 3.5-2: SOIS External Attitude References

3.5.4 SOIS Attitude Control Concepts. Attitude control concepts were defined for both spin stabilized and three axis stabilized SOIS stages to meet the requirements defined in Section 3.5.2, using the external attitude references shown in Figure 3.5-2. The attitude control concepts defined were used as the basis for Task 6 SOIS vehicle mass estimates.

Key features of the spin stabilized SOIS attitude control system are shown in Figure 3.5-3. Attitude control about roll pitch and yaw axes is provided by a combination of eight axial thrusters and four radial thrusters. Vehicle atti-

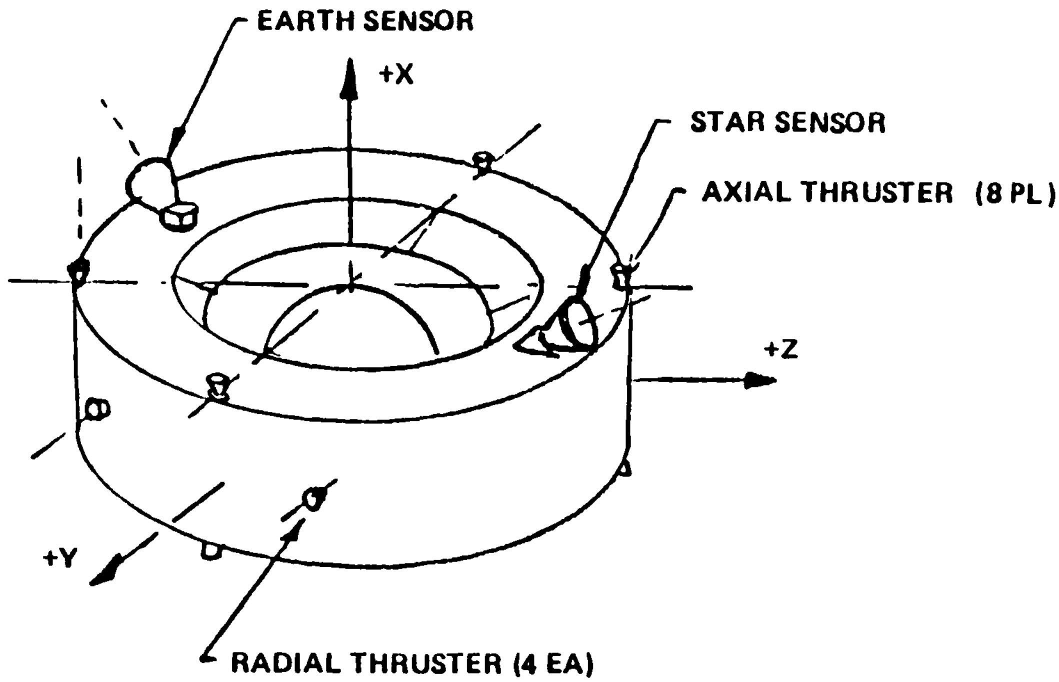


Figure 3.5-3: Spin Stabilized SOIS Attitude Control Concept

tude is determined by star and Sun sensors. Pulsing of axial thrusters at an angular reference determined by star and Sun sensors precesses the vehicle spin axis to accomplish pitch and yaw control. Roll control as well as spin up and spin down for stabilization are accomplished by four radial thrusters which can be pulsed to provide a torque about the vehicle's plus X axis. The Earth sensor serves for placement burn attitude verification.

Figure 3.5-4 illustrates features of the three axis stabilized SOIS attitude control concept. Vehicle attitude control is provided by four thruster clusters which provide redundant thrusters for control of vehicle pitch, roll, and yaw. Vehicle attitude is determined by redundant Sun sensors and star sensors. One of the star sensors is used as an Earth sensor for placement burn orientation verification. Closed loop attitude control during the placement burn is provided by the rate gyro attitude reference package which is used to hold a constant heading during the placement burn only.

3.5.5 Rescue Mission Control Requirements. Additional control requirements necessary for the rescue mission include establishing the target vehicle location after ground based navigation to within approximately 1000 km of the target, automated rendezvous and docking with the target vehicle, and waste payload transfer from the failed vehicle to the rescue vehicle. Real time

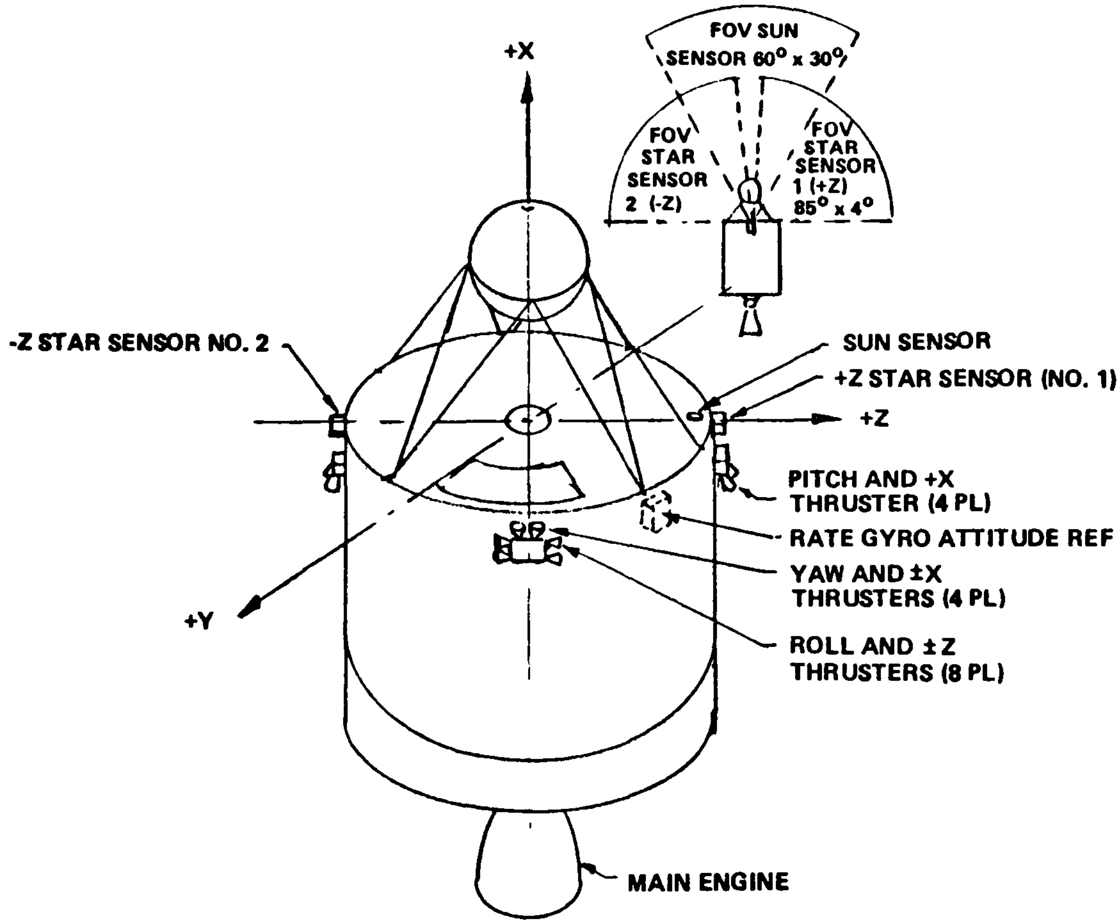


Figure 3.5-4: 3-Axis Stabilized SOIS Attitude Control Concept

monitoring including video and contingency real time control have also been assumed as rescue mission requirements. These requirements were used as the basis for estimates of mass impact to vehicles used in Task 6 performance assessments.

4.0 ORBIT TRANSFER SYSTEMS

4.1 INTRODUCTION

The objective of Task 4, Orbit Transfer Systems, was to trade a full range of orbit transfer system options (propulsion, staging, reuse) in a systematic manner to define the optimum orbit transfer system for the space disposal mission.

The approach used was to use the extensive existing data base on IUS, SEPS, and OTV to provide the basis for estimates of mass cost and performance. Trades were conducted in four primary areas: (1) assessment of propulsion systems for application, (2) identification of candidate systems and options, (3) definition of candidate vehicles, and (4) comparison of total system performance.

Key findings resulting from the study include: identification of eight primary candidate orbit transfer systems, including single stage systems using cryogenic chemical propellants and electric propulsion; two stage systems using all propulsive and aerobraked approaches to reuse of the injection stage; and using storable liquid, cryogenic liquid, and electric second stages. Characterization of these candidate systems and trade studies performed to define the best performing systems resulted in selection of four for consideration in Task 8.

4.2 IDENTIFY ORBIT TRANSFER SYSTEM OPTIONS

Options were identified in two steps. First a preliminary screening was used to identify the optimum applications for each propulsion system considered (Figure 4.2-1). Second, the remaining propulsion options were arrayed against staging and reuse options to define a matrix of all remaining orbit transfer system options (Figure 4.2-2).

4.2.1 Assess Propulsion Systems for Application. An initial screening of propulsion system candidates resulted in selection of cryogenic liquid, storable liquid, solid and solar electric propulsion for preliminary evaluation. Other concepts such as nuclear electric propulsion or laser propulsion were rejected

PROPULSION SYSTEM	ISP	MISSION PHASE: INJECTION	MISSION PHASE: PLACEMENT	COMMENTS
ROCKET ENGINE CRYOGENIC LIQUID PROPELLANTS ($LOX-LH_2$)	RANGE 450 TO 470	REFERENCE IN REUSABLE MODE EVALUATE EXPENDABLE	EVALUATE BOIL-OFF, AVIONICS IMPLICATIONS OF 165 DAY MISSION	• HIGHEST Isp FOR CHEMICAL SYSTEMS • LARGE DATA BASE • EVALUATE FOR INJECT & PLACEMENT
ROCKET ENGINE STORABLE LIQUID PROPELLANTS (N_2H_4-MMH)	RANGE 270 TO 309	REJECT DUE TO PERFORMANCE PENALTY	REFERENCE EVALUATE FOR COST OPTIMIZATION	• SIMPLEST LIQUID SYSTEM • ADAPTABLE TO REDUNDANCY • COST IS AN ISSUE • FLEXIBILITY IN INSTALLATION
ROCKET ENGINE SOLID PROPELLANT	300	REJECT DUE TO PERFORMANCE PENALTY	REJECT • NO ADVANTAGE RELATIVE TO STORABLE LIQUID	• RELIABILITY AS AN ISSUE • CATASTROPHIC FAILURE MODE • NOT ADAPTABLE TO REDUNDANCY • INSTALLATION IS INFLEXIBLE
SOLAR ELECTRIC POWERED ION ROCKET (SEP)	RANGE 3000 TO 9000	CONSIDER FOR COMBINED MISSION WITH AND WITHOUT OTV BOOST		• HIGH PERFORMANCE • LONG TRIP TIMES • CONCENTRATE ON COST, MANIFESTING ISSUES

Figure 4.2-1: Propulsion Systems Assessment

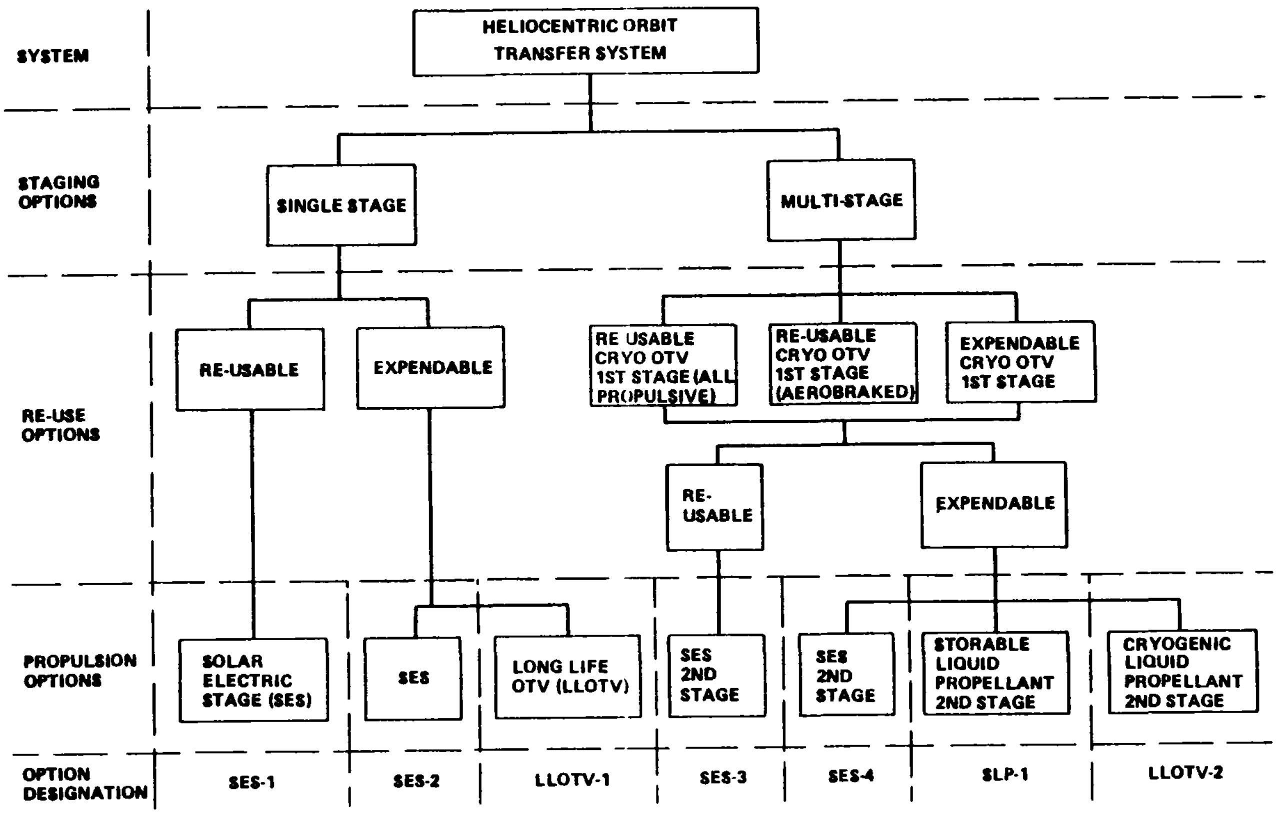


Figure 4.2-2. Identification of Candidate Orbit Transfer System for Space Disposal

as being difficult to characterize (particularly in cost) and of doubtful availability for 1995 IOC.

With candidate systems selected, a preliminary evaluation was conducted to assess propulsion systems for application to different mission phases. Results of the evaluation are illustrated in Figure 4.2-1.

Solid propellant rockets were rejected for further consideration. Solid rockets showed no advantage in specific impulse over storable liquid systems, suffer from reliability problems characterized by occasional catastrophic detonation on ignition, are not adaptable to redundancy, and are inflexible in terms of installation.

Storable liquid propulsion systems were assigned to the placement roll only. The performance penalty relative to cryogenic propellants for the injection mission led to their rejection in this application.

Cryogenic liquid propellants and electric propulsion are suitable for all mission phases. Cryogenic propellant systems are suitable for both injection and placement missions and for missions which combine the functions in a single stage expendable vehicle. Electric propulsion allows the additional option of vehicle return after the placement maneuver.

4.2.2 Identification of Options. A combination of the selected propulsion options by mission phase with the remaining options in the areas of staging and reuse provided a definition of the entire range of orbit transfer system options for space disposal of nuclear waste.

The results of this combination are illustrated in matrix form in Figure 4.2-2. Systems identified and their designations include: (1) a single stage reusable solar electric stage (SES-1), (2) a single stage expendable solar electric stage (SES-2), (3) a single stage expendable long life OTV (LLOTV-1), (4) a multistaged system using a cryogenic propellant injection stage and a reusable solar electric propulsion second stage (SES-3), (5) a multistaged system using a cryogenic injection stage and an expendable solar electric propulsion second stage (SES-4), (6) a multistaged system using a cryogenic propellant injection stage with an expendable, storable, liquid propellant second stage (SLP-1), and (7) a two staged system using a cryogenic liquid propellant injection stage and a cryogenic liquid propellant second stage (LLOTV-2).

Two stage options using storable liquid or cryogenic propellant upper stages were exercised with three injection stage reuse options:

1) All propulsive return
2) Aerobraked return
3) Use of an expendable injection stage

This provided a total of 13 options for characterization (option SES-3 was confined to use of an aerobraked injection stage for simplicity).

4.3 DEFINE CANDIDATE ORBIT TRANSFER SYSTEMS

Definition of candidate orbit transfer systems identified in Section 4.2 was achieved in two steps. In the first, the vehicles used in the candidate systems were defined, using the extensive data bases generated in both studies of SEPS and OTV and in the IUS production contract. Characteristics of the vehicles defined are described in Section 4.3.1; detailed mass statements for each vehicle are contained in Appendix G.

The second step was to use the vehicle definition of Section 4.3.1 and the mission parameters from Section 3 as inputs to the Boeing OTV PSMC (Payload and Sequential Mass Calculation) code. The output of the code provided a parametric characterization of the performance of each candidate system option.

4.3.1 Characterize Candidate Vehicles. The following types of vehicles are required for the different orbit transfer system options:

Injection Stages

1. Single launch LO_2/LH_2 OTV (SES 3,4, SLP-1)
2. Dual launch LO_2/LH_2 OTV (LLOTV-2)

Placement Stages

3. Storable propellant SOIS (SLP-1)
4. LO_2/LH_2 SOIS (LLOTV-2)

Combination Injection/Placement Stages

5. LO_2/LH_2 long life OTV (LLOTV-1)
6. Solar electric stage (SES) (SES-1, SES-2)

As the first step in determining the performance for the range of orbit transfer system options, point designs and parametric mass relationships were developed for the candidate vehicles. Definition of vehicle point designs was accomplished in three areas: LOX/LH_2 cryogenic propellant vehicles, storable liquid propellant vehicles, and solar electric propulsion vehicles.

For the LO_2/LH_2 stages the initial OTV defined in the Boeing/MSFC OTV concept definition study (Contract NAS8-33532) was used as a reference point design. It is very close to the optimum size for the single launch LO_2/LH_2 injection stage. Point designs for all other LO_2/LH_2 vehicles were developed as variations from this baseline. The emphasis was on analyzing and defining those areas of the LO_2/LH_2 vehicles that were different from the Phase A baseline.

The storable propellant placement stage was configured by combining existing engines (from the STS reaction control system) and avionics (from the IUS third stage) with an in-house design for structure and electrical power. The extreme simplicity of the vehicle allowed a reasonable level of definition to be achieved within the constraints of the study.

Definition of the solar electric stage drew heavily on the SEPS Phase B study effort and the previous Future Space Transportation Systems Analysis Study conducted by BAC for NASA/JSC (Contract NAS9-14323). A vehicle configuration developed for the space disposal mission in the JSC study was updated by using SEPS derived estimates for the key propulsion areas of arrays, power processors and thrusters. Structure, thermal, and propellant storage masses were scaled.

Other subsystems such as avionics and RCS were synthesized using components derived from the IUS and OTV programs.

The resulting point design vehicles are illustrated along with some of their key characteristics in Figure 4.3-1.

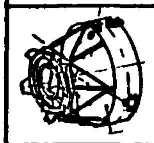
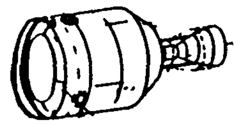
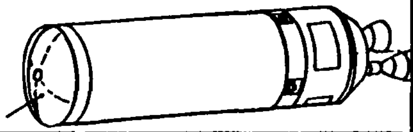
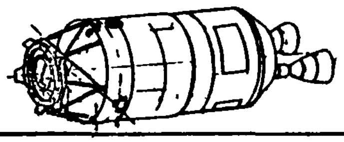
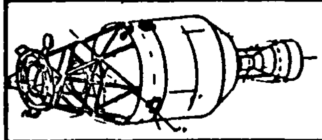
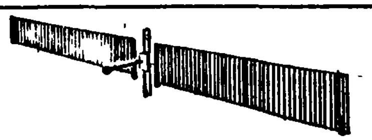
CONFIGURATION	USAGE	PROPELLANT	POINT DESIGN CHARACTERISTICS (Kg)			DIMENSIONS
			MASS AT BURNOUT	PROPELLANT MASS	MASS AT STARTBURN	
	PLACEMENT (SOIS)	STORABLE N_2O_4 N_2H_4	1091	3316	4407	4.7 M DIA X 1.52 M LONG
	INJECTION	CRYO LOX LH_2	2931	21,100	24,811	4.7 M DIA X 9.36 M LONG
	INJECTION	CRYO LOX LH_2	5,529	60,136	66,820	4.7 M DIA X 17.7 M LONG
	INJECTION + PLACEMENT (LLOTV)	CRYO LOX LH_2	3,919	28,122	32,736	4.7 M DIA X 10.9 M LONG
	PLACEMENT (SOIS)	CRYO LOX LH_2	1,823	12,383	14,461	4.7 M DIA X 7.7 M LONG
	INJECTION + PLACEMENT	ARGON	4,770	11,225	16,332	145 M X 50 M (ARRAY)

Figure 4.3-1· Candidate Vehicles and Characteristics

STORABLE PROPELLANT SOIS

The storable propellant SOIS is the first configuration shown in Figure 4.3-1. It is an MMH/N_2O_4 propellant stage that is spin stabilized during burns and three-axis stabilized during coast.

Structure. The structure is 7075 aluminum skin and stringer and consists of a circular shell section divided into six bays by radial panels and a center circular section. Upper and lower surfaces are closed out by skin panels. Propellant is stored in three N_2O_4 and three MMH 6 Al-4V titanium tanks at a maximum operating pressure of 1450 kPa. A design ultimate factor of safety of 4 allows the tanks to be designed without fracture mechanics validation.

Avionics. The storable SOIS uses a modified NASA IUS third stage avionics suit. A simplified block diagram of the system is shown in Appendix G.

Power Supply and Distribution. Electrical power is provided by a 14-m^2 solar array. This is a state of the art silicon array with 8 mil cells, 6 mil cover glass and 2 mil substrate. Batteries, power conditioning and switching equipment and wiring harness complete the power supply and distribution subsystem.

Main Propulsion. Main propulsion consists of six 445N thrust Aerojet AJ10-160 bi-propellant pressure feed engines. Three kevlar overwrapped bottles store GHe at 24,130 kPa to maintain pressure in the propellant tanks.

Attitude Control. Twelve STS orbiter 111N RCS vernier thrusters are used for attitude control. These are bi-propellant thrusters and use propellant from the main tanks.

A summary mass statement for the point design storable SOIS is shown in Appendix G.

LO_2/LH_2 OTV - SINGLE LAUNCH INSERTION STAGE

The single launch insertion stage is an LO_2/LH_2 propellant OTV that can employ the aerobraking option to accomplish the reduction in velocity to circularize upon return to LEO. The vehicle is the second configuration in Figure 4.3-1, and is illustrated in more detail with its ASE in Figure 4.3-2. The following sections present a summary description of the vehicle and its systems.

Because of the critical role of this vehicle as a standard from which configuration and mass estimates of all cryogenic stages were derived, the system descriptions are provided in detail, and can be applied to the derivative configurations. A complete summary mass statement for the vehicle is contained in Appendix F.

Primary vehicle systems include structure, thermal control, avionics, power supply and distribution, propulsion, attitude control, and the airborne support equipment (ASE).

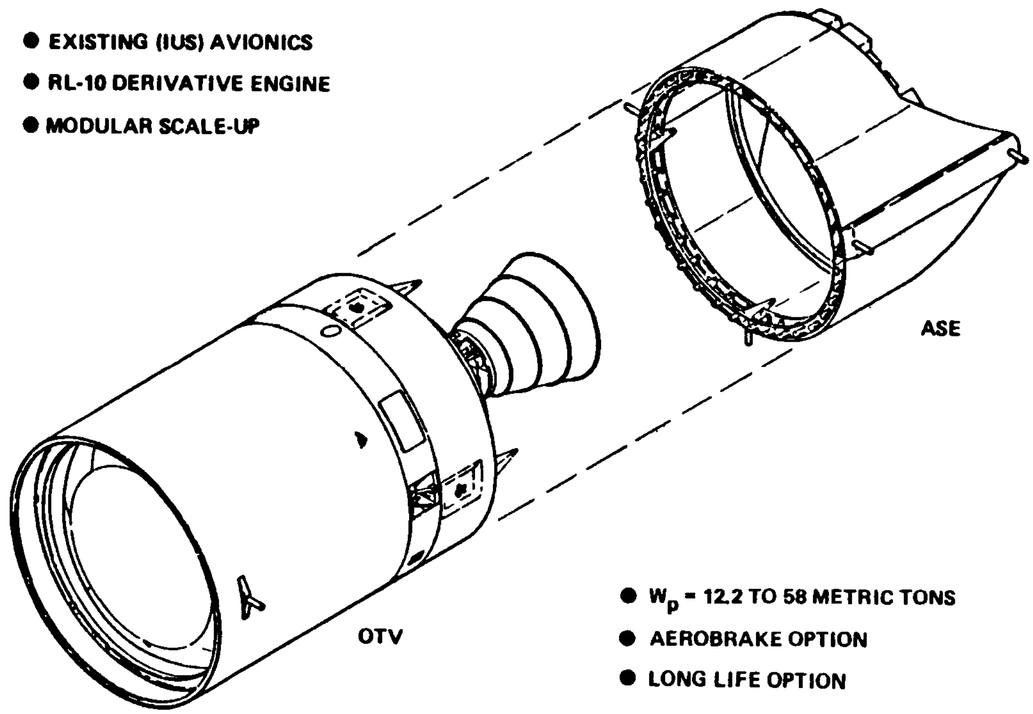


Figure 4.3-2: Single Launch LOX/LH$_2$ Injection Stage Vehicle

Structures. An exploded view of the OTV is shown in Figure 4.3-3. All of the vehicle and ASE external body shell except for the avionics ring are graphite/epoxy. The main propellant tanks are fabricated from 2219 aluminum

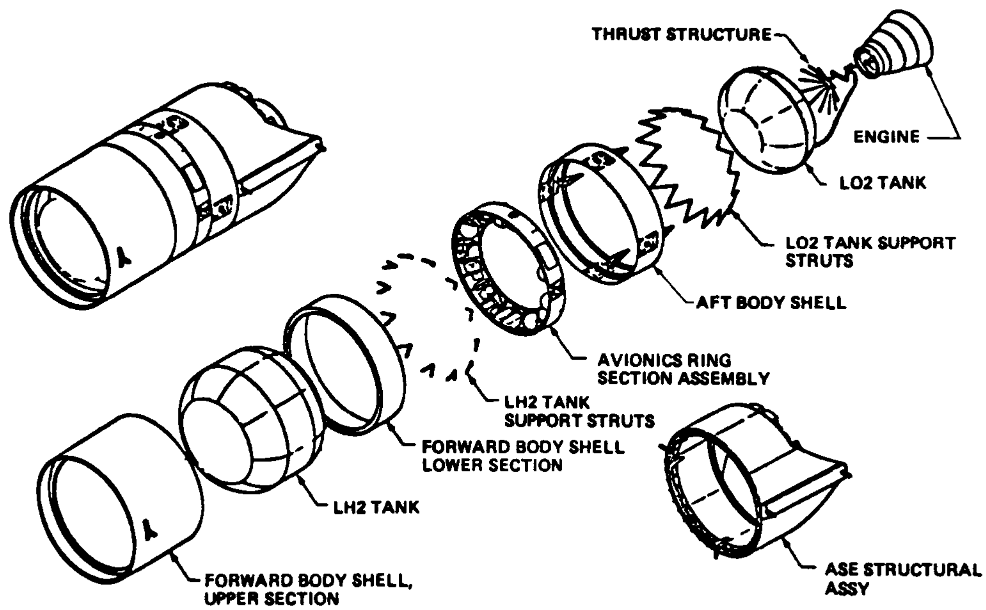


Figure 4.3-3 Exploded View of LOX/LH$_2$ OTV

and are designed for a 20 mission service life. Fiberglass struts are used to support the liquid hydrogen tanks, with graphite/epoxy struts used to support the liquid oxygen tank and the main engine. Trunnion fittings are made of titanium. Pneumatically actuated payload latch/release mechanisms are provided for payload separation. The vehicle side of the vehicle/ASE interface has a total of 36 receptacle fittings for the vehicle latch/release mechanisms mounted on the ASE.

A major portion of the electrical power, avionics, and attitude control system are mounted on the avionics/equipment ring. The external shell for this section is aluminum (for thermal control purposes) instead of the graphite/epoxy used for the remaining external body/shell.

The aerobraking capability described previously is achieved by the modular installation of the ballute deceleration subsystem on the aft body shell. It consists of the ballute, ballute inflation system, installation provisions, and pyrotechnic devices for the deployment and release of ballute prior to and after re-entry. A global positioning system (GPS) receiver/processor subsystem is added to the vehicle avionics to provide the precise position determination required for the aerobraking maneuver. These additions increase the dry mass by 83 kg.

Thermal Control. Thermal control of the OTV is accomplished by both active and passive techniques. The avionics are passively cooled. The avionics are mounted on an aluminum ring section with the components which operate during ascent in the orbiter located in the upper quadrant. The thicknesses of the mounting shelf and the external ring are tailored to accommodate component thermal requirements. Flexible optical solar reflector (FOSR) covers the external ring surface.

Waste heat rejection from the fuel cell system is accomplished using an active cooling loop with a space radiator mounted on the LO_2 tank support body shell. Freon 11 is used for the working fluid. Heaters are used on the attitude control subsystem storage tanks, feedlines, and thrusters and also for batteries and the fuel cell product water dump line.

The cryogenic tanks are covered with blankets composed of 23 layers of doubly aluminized kapton. To prevent air liquification and ice formation within the blanket, a ground purge is used during prelaunch activities and initial portions of ascent to LEO.

Avionics. A block diagram of the OTV avionics subsystem is shown in Figure 4.3-4. This subsystem accomplishes all guidance, navigation, and control functions; handles communications to the orbiter and ground; and with the orbiter mounted ASE, interfaces with the orbiter avionics. The avionics features redundant strings including two computers and is communications compatible with both STDN and TDRS. It is compatible with addition of GPS receivers to provide precise navigation for the aerobraking return maneuver. Arrangement

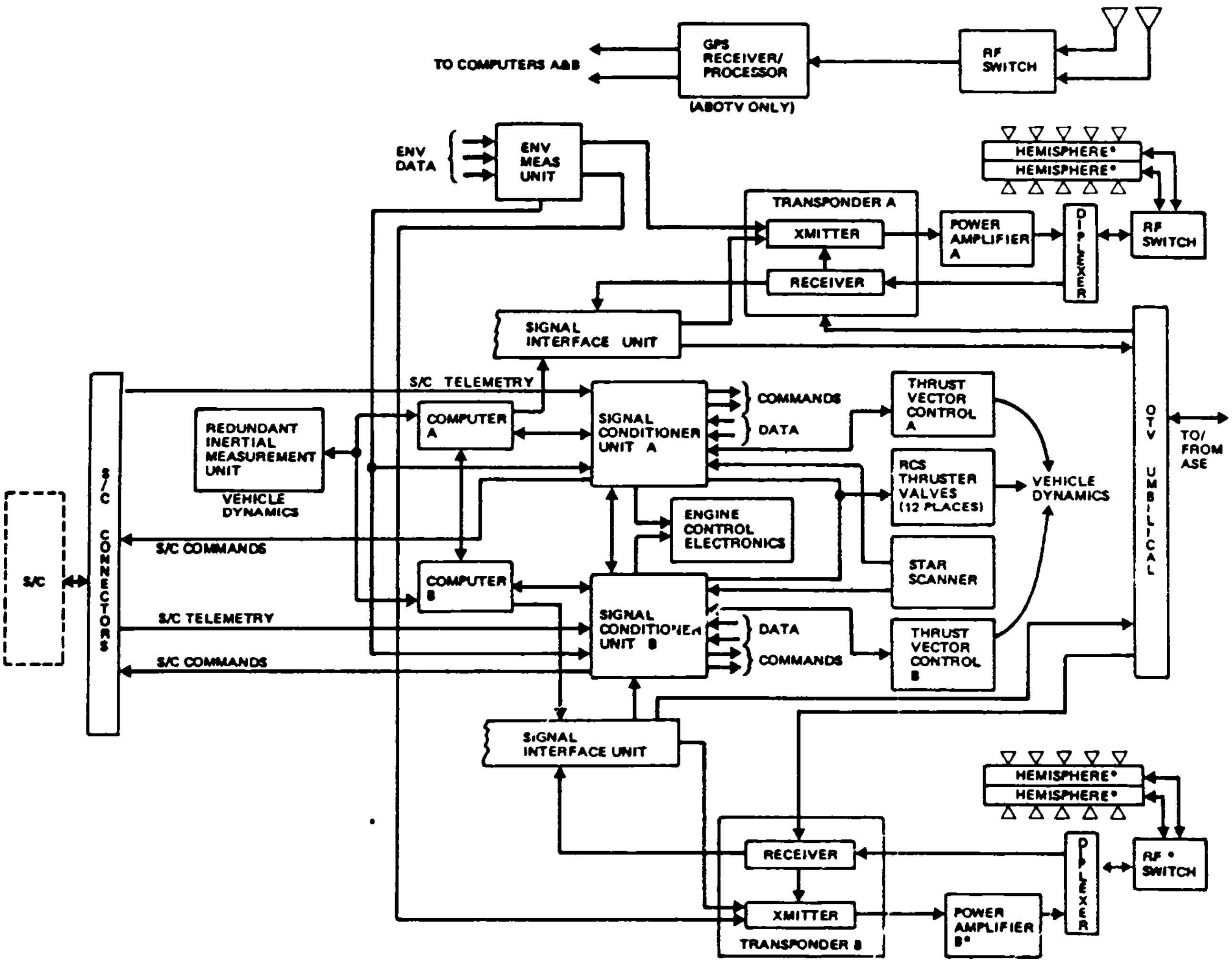


Figure 4.3-4: Avionics Block Diagram

of the avionics system components on the structure avionics ring is illustrated in Figure 4.3-3.

Power Supply and Distribution. The electrical power supply and distribution subsystem, designed for 28V operation, is powered by redundant, low pressure, modified orbiter H2/O2 fuel cells, each rated at 2.0 kW nominal/3.5 kW peak. Dedicated reactant storage tanks are used with reactant expulsion similar to the orbiter design. A 25 A/hr nickel/hydrogen utility battery is also provided. The distribution and control subsystem is based on IUS design. The system design provides for redundant power distribution units. The load demand on the power supply is approximately 2 kW during coast and 3 kW during main engine operation.

Propulsion. The schematic for the main propulsion subsystem is shown in Figure 4.3-5 with the general arrangement of propulsion system components illustrated in Figure 4.3-3. Main propulsion is provided by a single Pratt & Whitney

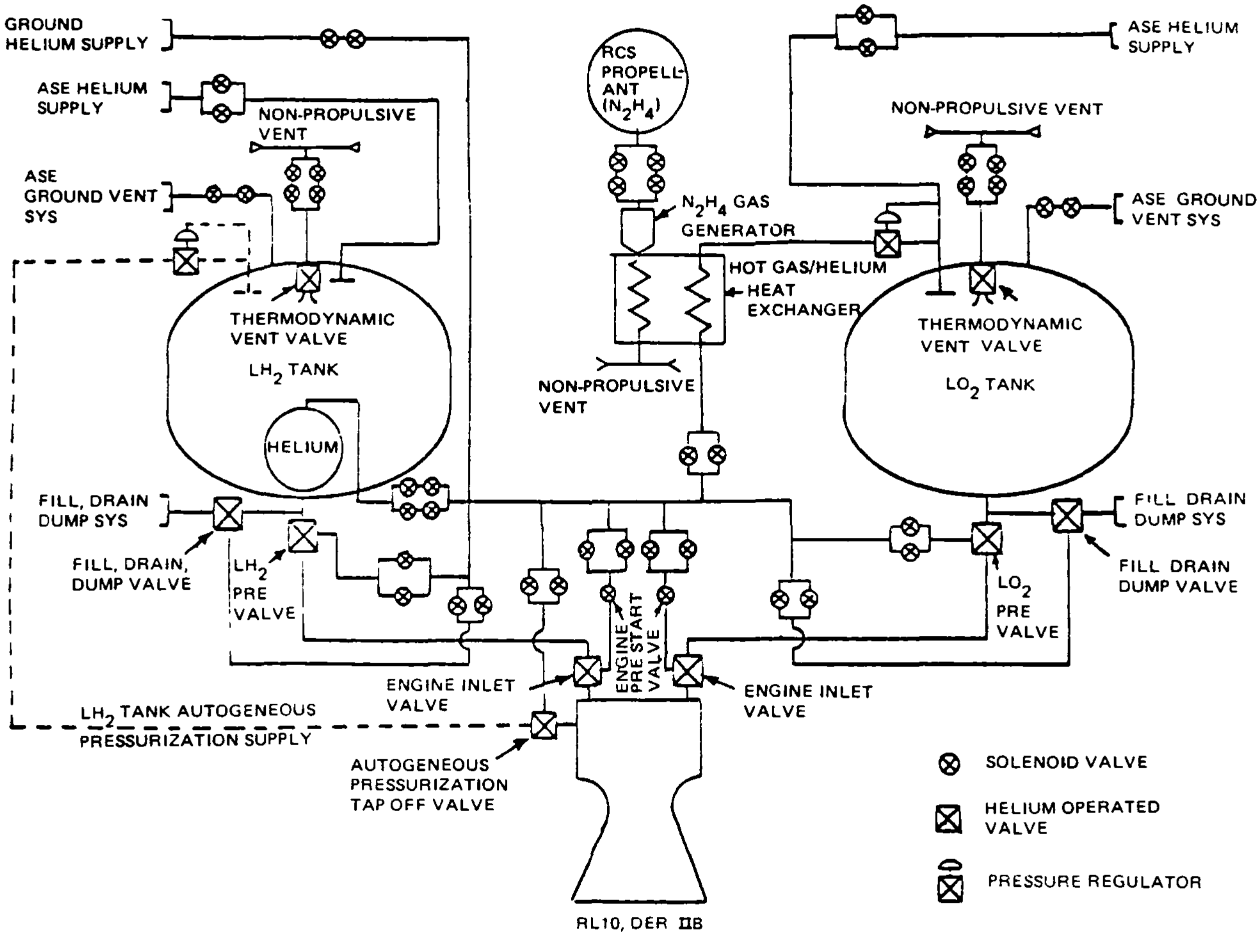


Figure 4.3-5. LOX/LH_2 OTV Main Propulsion System Schematic

RL10-IIB engine which has a stowed length of 1.778m and provides 66,720N of mainstage thrust. The main propellant tanks have usable capacities of 3130 and

17,970 kg of liquid hydrogen and oxygen respectively. The propellant delivery system features 0.057m delivery lines, tank sump-mounted prevalves, and 0.102m fill, drain, and dump lines with redundant parallel dump valves. Tank pressurization is accomplished using autogenous pressurization during engine mainstage. Separate space and ground (orbiter) vent systems are provided.

Attitude Control. The attitude control subsystem (ACS) uses hydrazine mono-propellant with pressure blowdown positive expulsion. The ACS uses 12 IUS reaction engine modules (REM) and propellant storage tank assemblies. Each of the three 0.533m diameter titanium tanks provides a usable propellant capacity of 54 kg. Propellant expulsion is accomplished using a flexible diaphragm and N_2 pressure blowdown from 2620 kPa to 690 kPa. Propellant isolation while in the orbiter is accomplished by using pyrotechnic valves which are opened after OTV deployment at LEO. The thrusters provide 133N of thrust with 2620 kPa inlet pressure and 36N at 690 kPa inlet pressure. Specific impulse is 235 and 230 at the 133N and 36N thrust levels respectively. Propellant tanks, REM's and all plumbing are mounted on the avionics ring section (Figure 4.5-3).

Airborne Support Equipment. The ASE provides for all electrical and fluid interfaces between the OTV and the orbiter. The ASE provides structural support as well as tilt and vehicle release mechanisms. The configuration and major features of the ASE are shown in Figures 4.5-2 and 4.5.3.

DUAL LAUNCH LOX/LH_2 OTV INJECTION STAGE

The dual launch injection stage is a stretched version of the baseline OTV (Figure 4.3-6). It shares many subsystems with the single launch injection stage. Avionics and electrical power subsystems are unchanged except for increased length in the wiring harnesses. The vehicle has a larger version of the structure and thermal control systems used on the smaller stage. The ACS has three additional propellant tanks but is otherwise unchanged. Two RL 10 IIB engines are used to provide the necessary thrust for the increased gross mass. The propellant system schematic is the same except that the final feed line splits to the two engines and the line and valve sizes upstream of the split have been increased to match the flow rate of the two engines. A point design summary mass statement for the stage is contained in Appendix F.

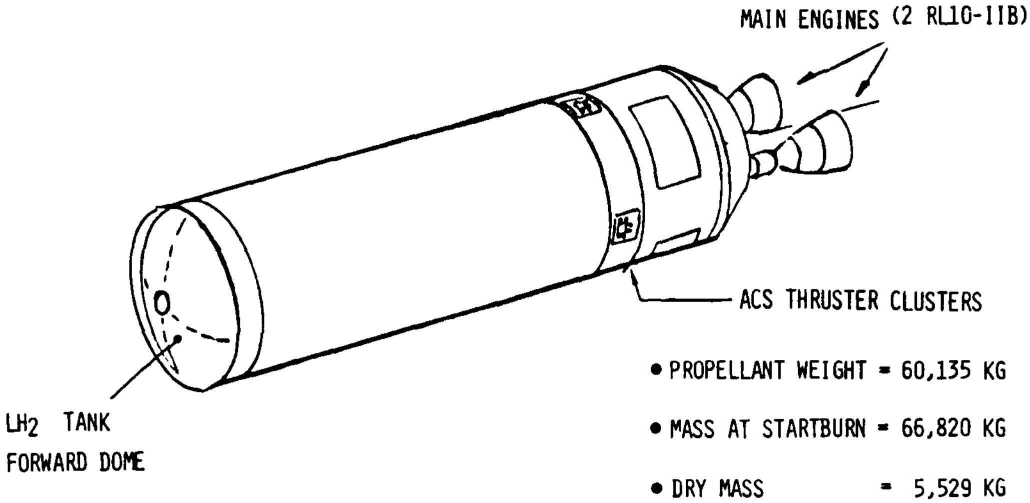


Figure 4.3-6: Dual Launch LOX/LH_2 Injection Stage Point Design

LOX/LH_2 LONG LIFE OTV (LLOTV)

The LLOTV combines the injection and placement functions in one stage. As such it must carry all subsystems peculiar to each type of stage. The stage general arrangement and features are illustrated in Figure 4.3-7. The structure, attitude control and main propulsion are those of the baseline OTV except for

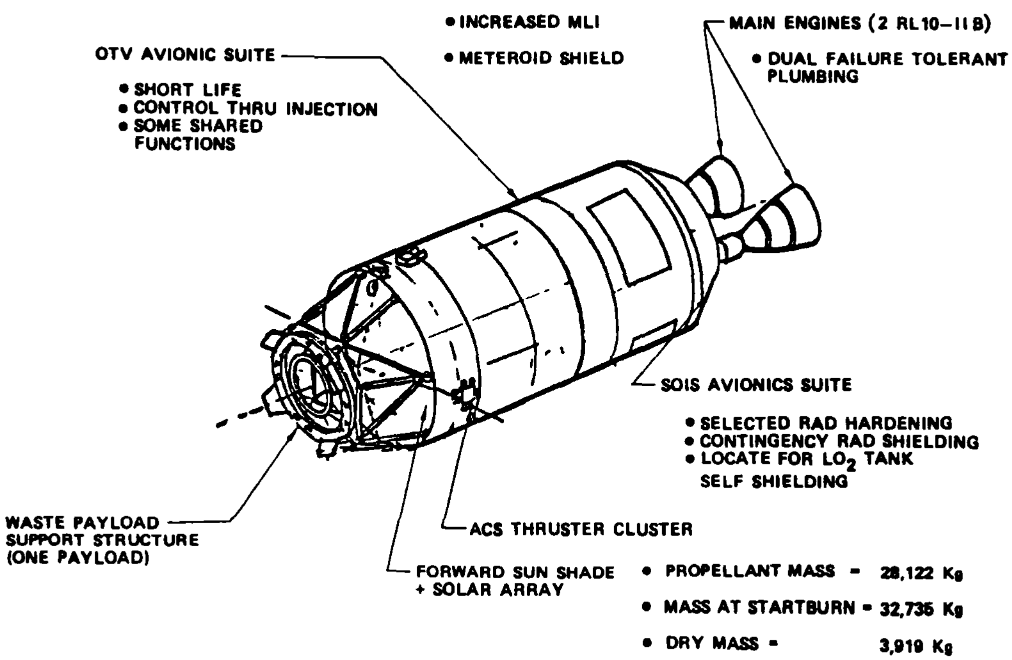


Figure 4.3-7: Long Life Orbit Transfer Vehicle (LLOTV) LOX/LH_2 Injection Plus Placement Stage Configuration

increased size for tankage and structure. Thermal control consists of the baseline OTV subsystem with MLI increased to 40 layers for the long duration mission and a forward sunshield added. The avionics subsystem includes both the baseline OTV avionics suit and the SOIS avionics suit. This is necessary because the baseline OTV avionics are not capable of long duration operation. The electrical power subsystem consists of the baseline OTV subsystem plus the solar array and other components of the SOIS subsystem. Electrical power during geocentric operations is provided by the OTV subsystem and during coast by the solar array. Primary batteries are switched in to provide additional power during the placement maneuver at 0.85 AU.

A summary mass statement for the LLOTV stage is contained in Appendix F.

LOX/LH_2 PLACEMENT STAGE (CRYO SOIS)

The CRYO SOIS is an expendable cryogenic propellant stage designed to support the waste payload during the 165 day post injection coast in transfer orbit and to perform the circularization or placement maneuver when the destination orbit radius of 0.85 AU is achieved. The general arrangement and key features of the stage are illustrated in Figure 4.3-8.

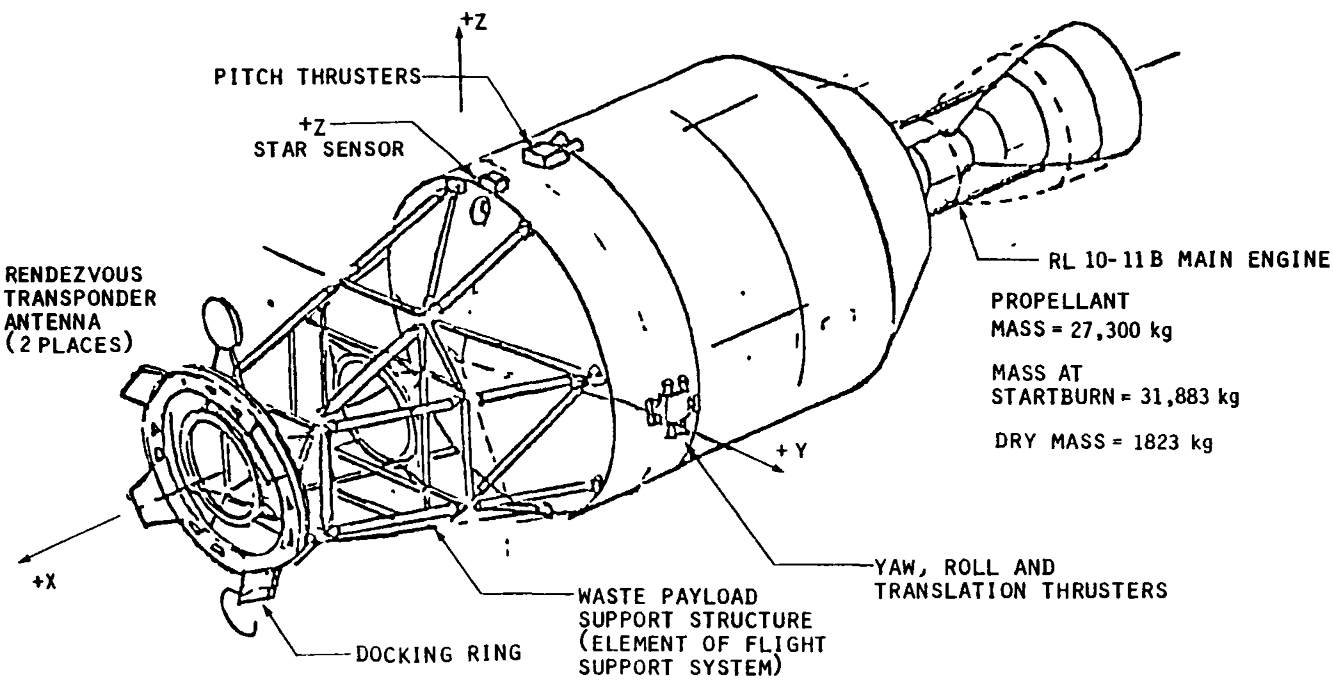


Figure 4.3-8: LOX/LH_2 Placement Stage (Cryo SOIS) Configuration and Features

The CRYO SOIS is essentially a smaller version of the basic injection OTV with changes in the thermal control, avionics and electrical power subsystems to handle SOIS functions.

Thermal Control. Changes in the thermal control subsystem are required because of the 162 day coast period from 1 AU to 0.85 AU. The stage remains oriented during coast in a head on attitude to the Sun. A sun shield mounted on the front end of the vehicle reduces the incident heat flux. The number of layers of MLI has been increased from 23 to 40 to prevent excessive boiloff.

No active thermal control is required because electrical power is provided by solar array and batteries instead of fuel cells.

Avionics. The LO_2/LH_2 SOIS has the same avionics suit as the storable SOIS except for the addition of two thrust vector controllers (TVC) and the main engine control unit required for control of the main engine. These units are from the baseline OTV.

Electrical Power. This subsystem is similar to the storable SOIS except for additional batteries which are used at the end of mission only to provide the higher power needed to drive the TVC actuators during the circularization burn into 0.85 AU orbit.

A summary mass statement for the CRYO SOIS is contained in Appendix F.

SOLAR ELECTRIC STAGE (SES)

Definition of the solar electric stage options shown in Figure 4.3-1 required more effort than that devoted to the chemical propellant stages because of the relative difficulty of analyzing the performance requirements imposed by the low acceleration, continuous burn trajectories used by the SES. The task was accordingly divided into three subtasks: trajectory and performance analysis (performed by a Boeing proprietary code and described in Section 3.2.3), initial SES technology characterization and option sizing, and SES point design definition.

Technology Characterization and Option Sizing. The objective of this task was to characterize solar electric propulsion (SEP) vehicles for transporting nuclear waste. The four mission scenarios considered are:

1. One-way LEO to 0.85 AU
2. Round-trip LEO to 0.85 AU and return
3. One-way high Earth orbit (HEO: 7000 nmi) to 0.85 AU
4. Round-trip HEO to 0.85 AU and return

Returned solar electric stages are recovered and refurbished for reuse. Ion propulsion using 50-cm argon thrusters is used. The technology is second generation with respect to the contemporary NASA SEPS stage.

Technology Characterization. The basic technology assumptions are shown in Figure 4.3-9. All have been demonstrated at the component level. The 50 cm thrusters chosen are similar to those presently under design by NASA/LeRC; the five power supply power processing unit (PPU) was first demonstrated in 1977; the capacitor-diode voltage multiplier (CDVM) screen supply has been demonstrated at BAC, Hughes Aircraft, and at NASA/LeRC. Space qualified 2 mil cells are in production.

THRUSTERS	POWER PROCESSING UNIT (PPU)	SOLAR ARRAY
DIAMETER TBD (50-cm REF)	5 POWER SUPPLIES	2 MIL SI CELLS
Ar, Xe, Hg, Cs	CDVM SCREEN SUPPLY	EFFICIENCY: 15%
MULTIPLE	SEPS THERMAL CONTROL	BLANKET: 190 W/M^2
LIFE – MISSION	NO ISP CONTROL	DEGRADATION: 50%/1 YR.
N_T = 0.71 @ 5000 SEC	N_p = 0.93 @ 11 KW	BLANKET: 2.166 KG/KW
		6 KG/KW: BOL

Figure 4.3-9: SES Technology Projections

Argon was selected as the propellant used in vehicle characterization because of the wealth of existing data, and because of its benign plume characteristics (minimum magnetic-atmosphere impact). If this proves to be less of a problem, alternate propellants such as Xe, Hg, or Cs offer the potential of higher performance.

PPU performance (weight and efficiency) benefits greatly from combination of power supply function, higher power rating, and an unregulated beam voltage. Thermal control for the PPU is accounted at 8.0 kg/kW radiated. The principle improvement in solar array technology is the adoption of 2 mil silicon cells at 15% (BOL) efficiency. Most of the array mass is attributable to launch packaging.

Option Sizing. Definition of mission profiles and technology allowed SES vehicles to be sized for the required mission options. Option 4 was dropped due to operational complexity, leaving one way and round trip missions to 0.85 AU from LEO, and an expendable mission from a 7000 nmi HEO to 0.85 AU as the remaining choices.

Sizing of vehicles for the four options used a proprietary code to select the specific impulse yielding the maximum payload for a 3-year trip time (chosen as representative of achievable system lifetimes for SES systems). An inital mass including payload of 34,000 kg was assumed as a constraint. The code used provided estimates of the initial power, number of thrusters, and propellant mass used.

Results of the sizing study for the three mission options are illustrated in Figure 4.3-10, which for each of the three options shows initial mass, initial

MISSION	① LEO / 85	② LEO / 85/ LEO	③ HEO/ 85	④ HEO/ 85 / HEO
INITIAL MASS kg (LBM)	34000 (75000)	34000 (75000)	13200 * (29105.)	TBD
INITIAL POWER Kw	600.	650	60	TBD
SPECIFIC IMPULSE (SEC)	9000	8000	7000	TBD
TRIP TIME – DAYS	1100	1073 (PROPULSION)	1090	TBD
PROPELLANT MASS (Kg)	4242	5830	1450	TBD
NUMBER OF THRUSTERS	8	12	3	TBD
DELIVERED PAYLOAD Kg / LBM	23600 / (52000)	21320 / (47000)	10460 / (23060)	TBD

* DELIVERED TO 11,268 km CIRCULAR @ 28.5°

Figure 4.3-10: SES Performance Summary power, specific impulse, number of thrusters, propellant mass, and deliverd payload for identical initial masses and trip times. A review of these results led to selection of the single stage expendable mission for development as a point design due to the following factors.

1. Highest delivered payload for single launch option.

2. Reusability for solar electric stages is questionable due to the difficulty of restowing the solar array; the long component life required; dependance on array annealing; and the operational complexity is also a negative factor.

3. The cost savings of reusability may not exist. Array costs are such a strong function of quantity that the smaller number of vehicles required for a reusable fleet might cost as much as the expendable fleet, which being more numerous benefits from economies of scale in solar array production.

In addition, the expendable mission, if attractive in comparison to chemical stages, would automatically establish the viability of a reusable SES if such a vehicle were to prove technically feasible.

Solar Electric Stage Point Design Definition. A further review of the SES performance led to selection of a stage sized to carry the maximum size fully shielded waste payload still capable of passive thermal dissipation. This allowed reduction in the array size to 270 kW and provided a substantial margin for mass growth.

The point design vehicle general arrangement and features are illustrated in Figure 4.3-11.

4.3.2 Candidate Systems Parametric Characterization. The candidate vehicles described in the previous section were assembled into candidate orbit transfer systems. Performance parametrics were developed for these candidate orbit transfer systems. These parametrics were combined with cost estimates in task 8 so that candidate systems could be compared on cost per kilogram of delivered waste.

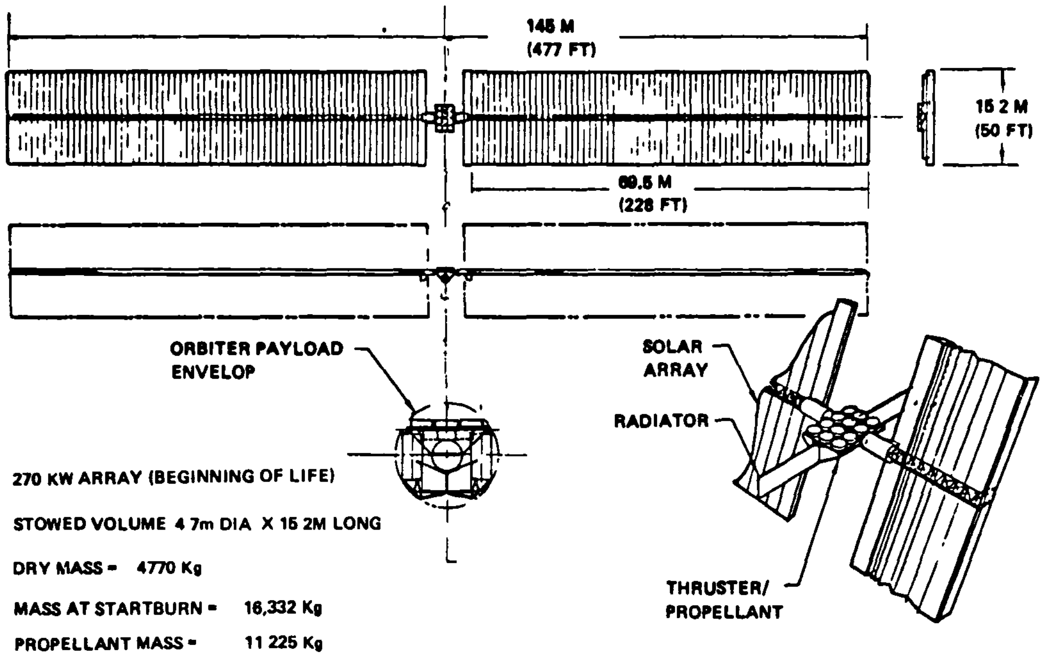


Figure 4.3-11. Solar Electric Stage (SES) Point Design Configuration and Key Characteristics

Candidate Orbit Transfer Systems. The following types of orbit transfer systems were identified in Section 4.2.2 for the nuclear waste disposal mission.

- Aerobraked recoverable OTV/Storable SOIS
- Aerobraked recoverable OTV/LOX/LH_2 SOIS
- Expendable OTV/Storable SOIS
- Expendable OTV/LO_2/LH_2 SOIS
- Long life LO_2/LH_2 OTV
- Aerobraked OTV/SEPS
- Solar Electric Stage
- All propulsive recoverable OTV/Storable SOIS
- All propulsive recoverable OTV/LOX/LH_2 SOIS

Parametric Characterization. Parametric payload versus weight relationships were developed to determine the maximum payload capabilities of the different orbit transfer systems for each of the launch options.

Trending curves for mass at burnout were generated from the point design mass statements described in the previous section. The burnout mass versus propellant mass relationships are shown for the OTV's and SOIS in Figures 4.3-12 and

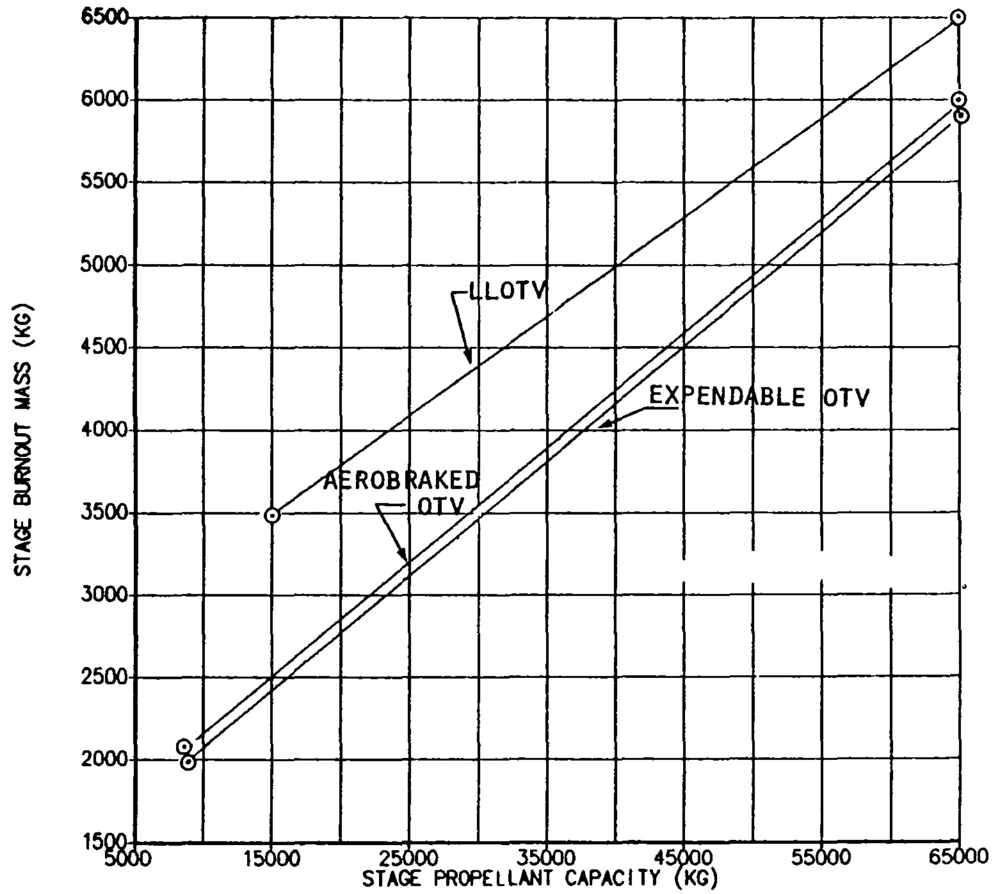


Figure 4.3-12: Injection Stage (OTV) Mass Trending Curves

4.3-13, respectively. A series of points were selected from these curves and used for the performance analysis.

The PSMC program was used to determine payload capabilities of the candidate orbit transfer systems. Given a stage burnout mass and propellant capacity, PSMC calculates propellant consumption, losses and stage mass for each event in the mission profile. Payload and start mission mass are iterated until calculated propellant consumption and burnout mass match the specified values. The program incorporates a complete mission profile of time and delta-V, for each event. The type of burn, either RCS or main engine, and corresponding start-stop losses can be specified. Boiloff and EPS losses are calculated from the timeline and specified loss rates. The loss rate is specified as a function of propellant capacity to handle different stage sizes. A detailed mission sequential mass statement listing event, delta-V, propellant usage, losses, and mass is printed along with a summary mass statement.

In addition to basic stage masses from the mass trending curves, a 254 kg inter-

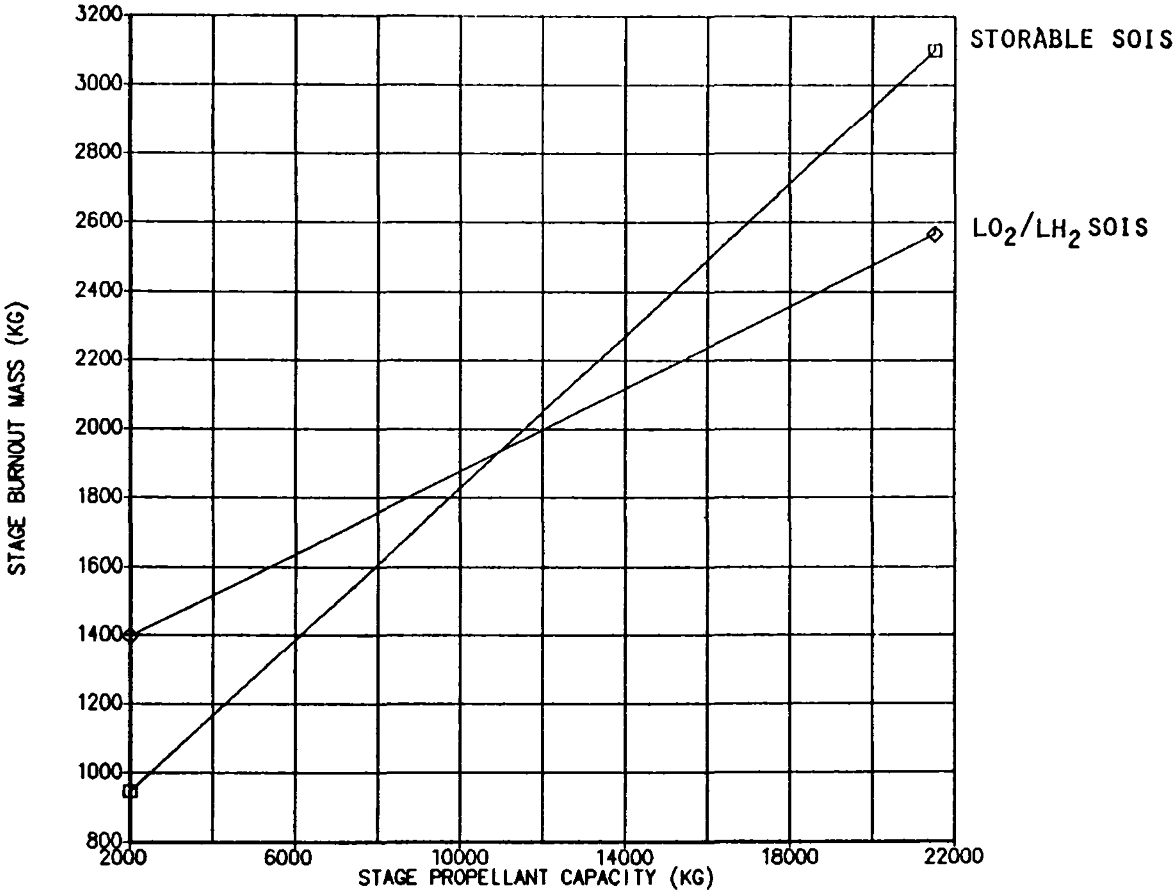


Figure 4.3-13: Placement Stage (SOIS) Mass Trending Curves

stage is carried by the OTV/LO_2/LH_2 SOIS combinations. This is jettisoned by the OTV after injection into heliocentric transfer. A payload adapter mass of 227 kg for single waste ball and 454 kg for double waste ball payloads was added to the SOIS burnout masses.

Figure 4.3-14 presents the mass required as a function of payload for the aerobraked OTV/storable SOIS. The mass lines plotted are start mission, OTV and SOIS, SOIS and payload, SOIS, and SOIS propellant from top to bottom, respectively. Also shown are the same lines for an aerobraked OTV/storable SOIS system in which the payload radiation shield is jettisoned at injection into the heliocentric transfer orbit.

The same data for the aerobraked OTV/LO_2/LH_2 SOIS and the LLOTV are shown in Figures 4.3-15 and 4.3-16, respectively. In all cases where the payload radiation shield is jettisoned, an additional mass of 770 kg is added to the SOIS burnout mass for a radiation shield to protect it from the payload.

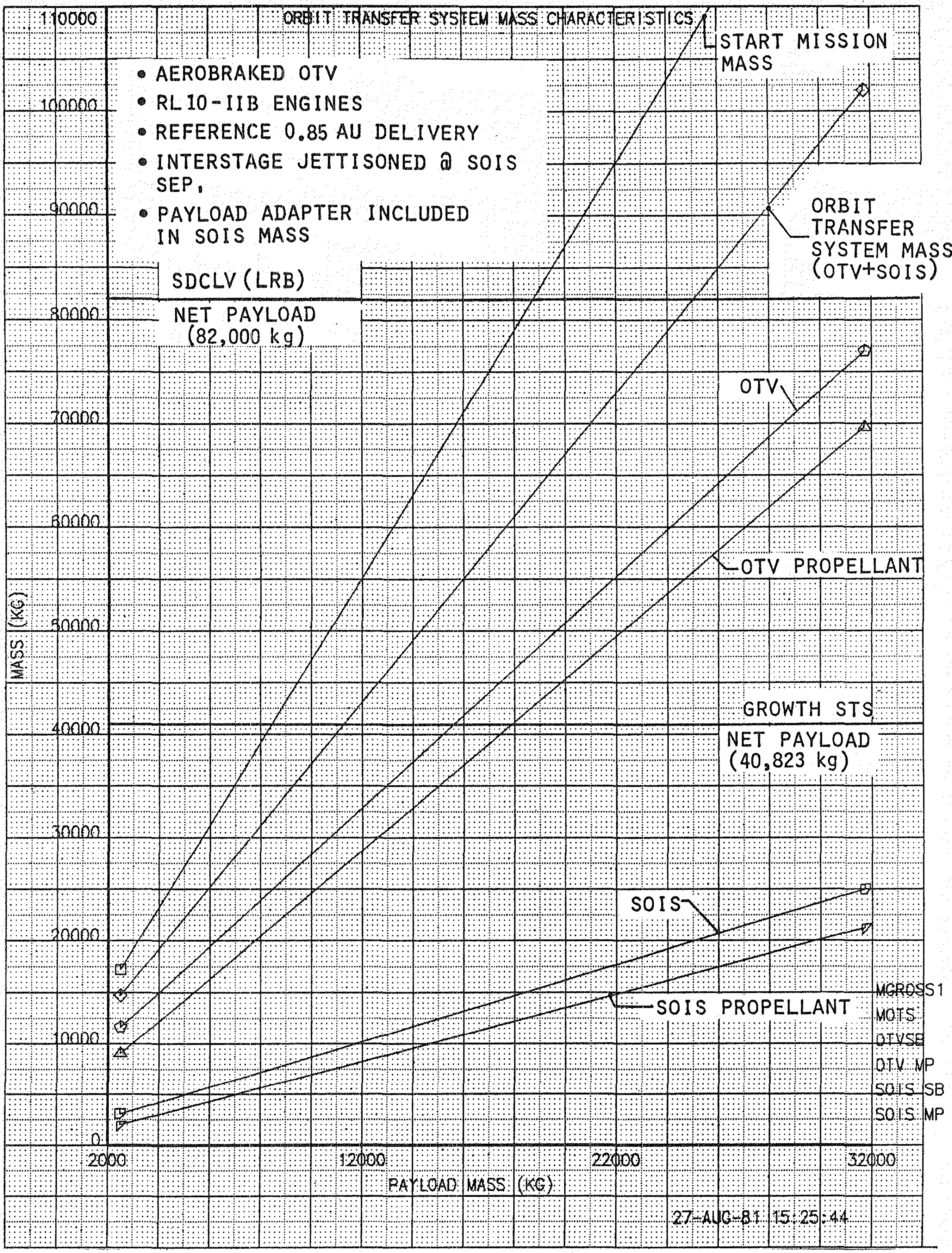


Figure 4.3-14: Parametric Performance Characterization for Aerobraked OTV/Storable SOIS

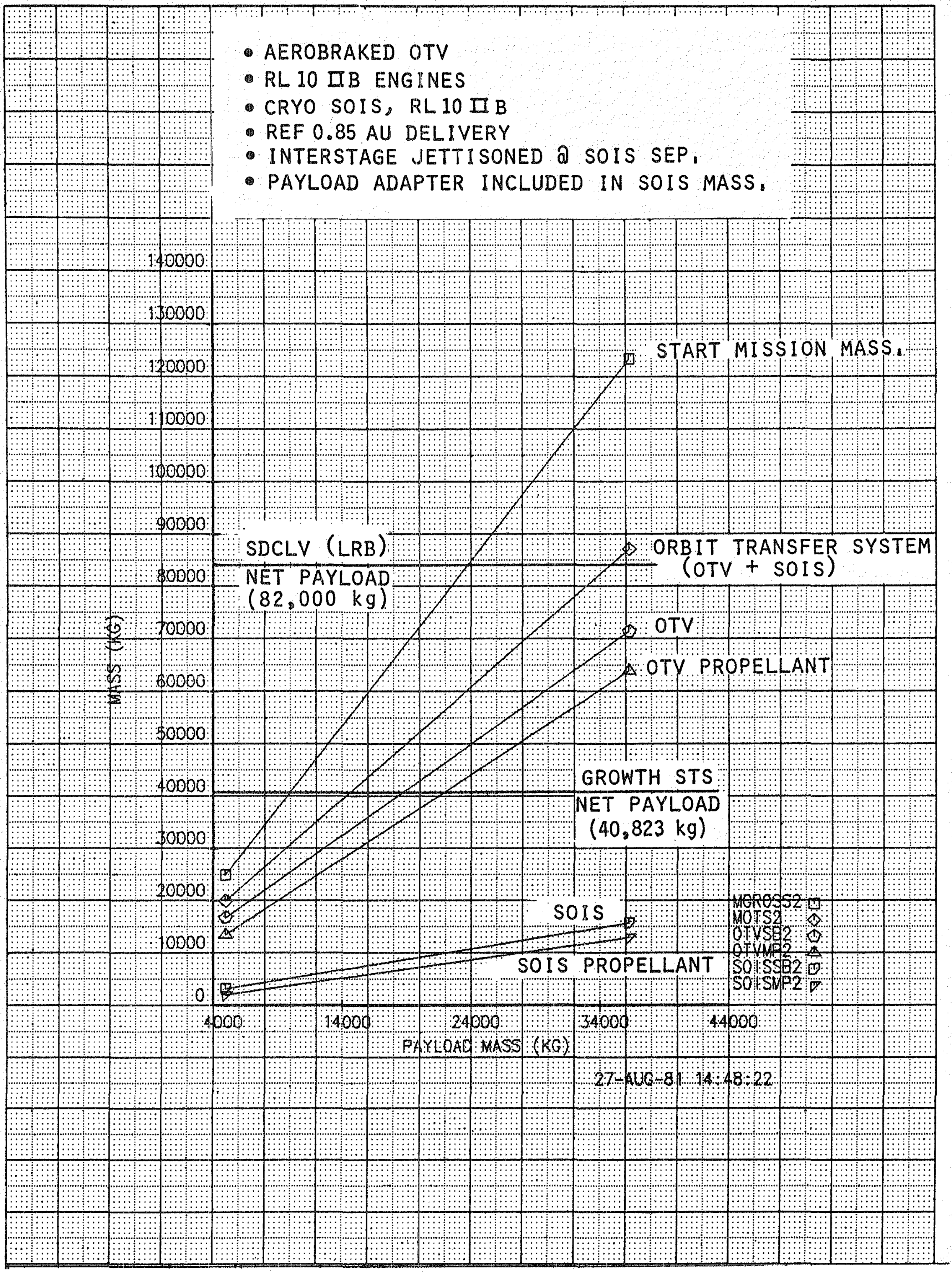


Figure 4.3-15: Parametric Performance Characterization for Aerobraked OTV/Cryogenic SOIS

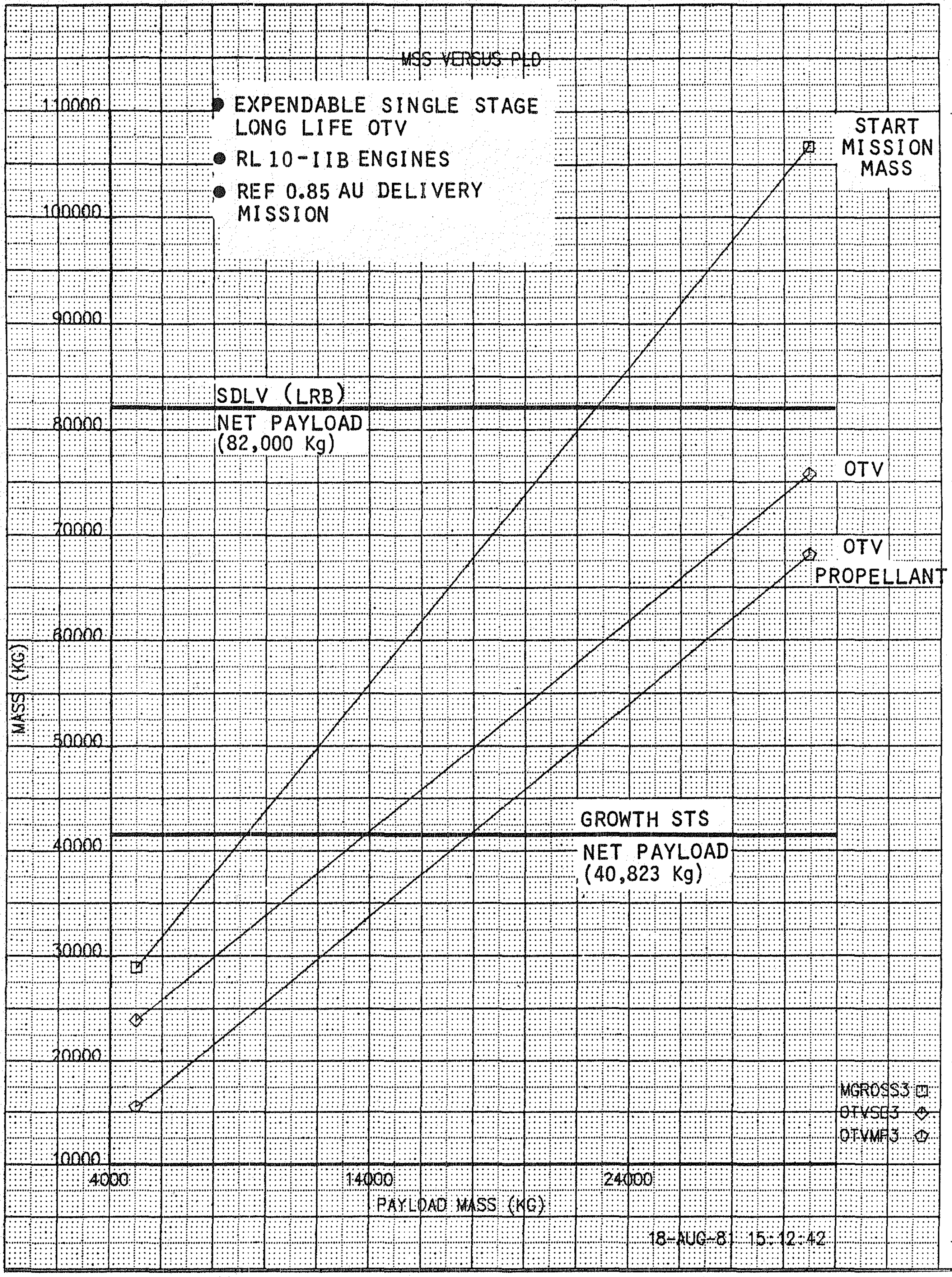


Figure 4.3-16: Parametric Performance Characterization for LLOTV

The points of interest on these figures are the net payload limits for the growth STS and the class 2 SDV launch vehicles. Single launch orbit transfer systems are constrained to a start-mission mass corresponding to the growth STS limit and the dual launch orbit transfer systems are constrained to an OTV and SOIS mass matching the class 2 SDV limit.

Data for the expendable OTV/SOIS systems are not shown. Preliminary investigation showed this type of system to have essentially the same performance as the aerobraked OTV/SOIS systems but with significantly higher cost per flight due to the OTV being expended. For this reason it was not considered further. The solar electric stage, being a single point design, was not characterized parametrically.

4.4 SELECT CANDIDATE ORBIT TRANSFER SYSTEMS

A parametric performance characterization of orbit transfer systems contained in Section 4.4 provided the basis for performance comparisons leading to selection of four orbit transfer system options for consideration in Task 8. Comparisons were carried out in two steps for chemical orbit transfer systems. In the first, six candidate two stage orbit transfer systems were compared to select the best two stage system. In the second step, the best two stage systems were compared to single stage systems. No performance trades were carried out for electric vehicles as only one option was selected.

4.4.1 Parametric Performance Comparisons. Figure 4.4-1 illustrates the parametric performance comparisons for six two stage orbit transfer systems. Orbit transfer system mass is plotted on the ordinate as a function of delivered payload mass plotted on the abscissa. Variables include both cryogenic and storable propellant SOIS propulsion and three injection stage options: all propulsive return for recovery, aerobrake return for recovery, and the option of extending the injection stage without recovery.

Comparisons were made for orbit transfer system masses of about 82,000 kg, typical of dual launch options where the waste payload is carried up in one launch and the orbit transfer system in a separate launch, and for orbit transfer systems of about 23,000 kg, typical of single launch options with waste payload and orbit transfer system carried to low orbit in the same launch vehicle. The

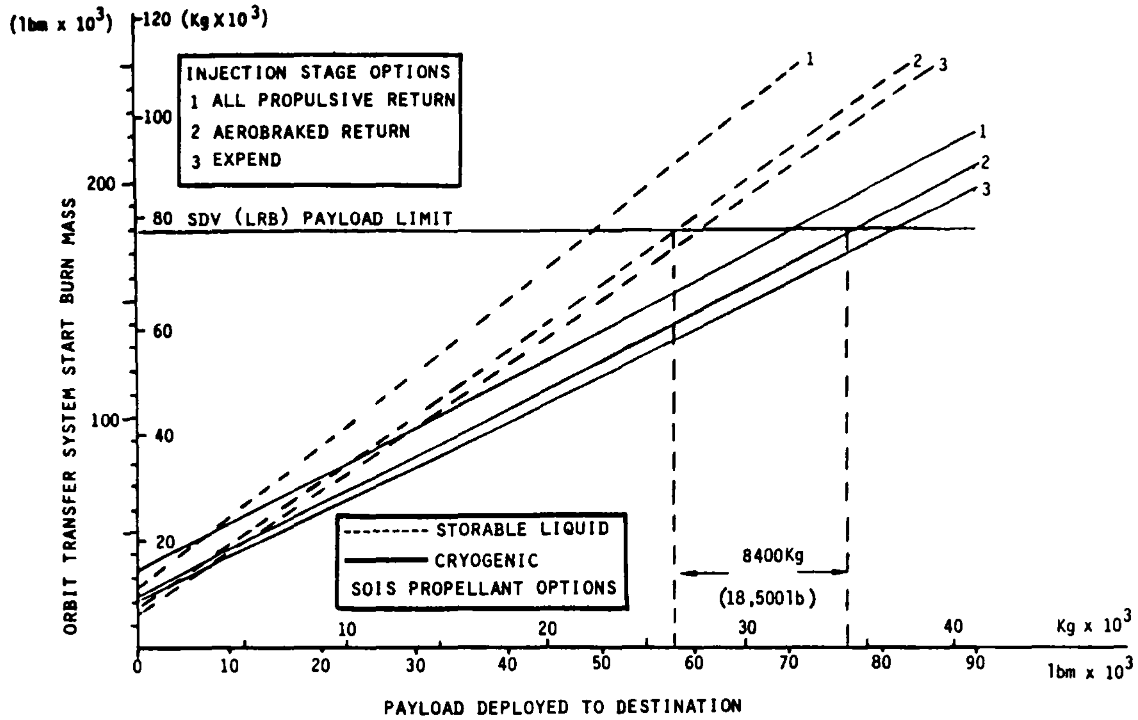


Figure 4.4-1: Parametric Performance Comparison for 2-Stage Systems

comparisons illustrated resulted in two selections: for the large system the selection of the cryogenic SOIS results in a payload gain of over 8200 kg. Use of the aerobrake return option in comparison to all propulsive return is worth a further 2000 to 3000 kg. Expending the injection stage will gain only 1000 to 2000 kg (about 4%) but the added cost associated with the expended injection stage led to its rejection in favor of the aerobrake option.

The choice for the single launch compatible system was not dominated by performance; even though the cryogenic upper stage still showed superior performance, STS cargo bay length constraints eliminated it from consideration. Evaluation of injection stage options for the small system lead once more to selection of the aerobraked injection stage for superior performance.

In summary the selected two stage options are:

1. For systems using dual launch, an aerobraked cryogenic propellant injection stage teamed with a crygenic propellant SOIS.

2. For smaller systems carring the orbit transfer system and payload in a single launch, an aerobraked cryogenic propellant injection stage using a storable propellant SOIS.

A comparison of the performance of the best two stage systems against single stage systems is illustrated in Figure 4.4-2. The performance of two stage systems using both storable and cryogenic propellant SOIS options is shown with the performance of single stage LLOTV options plotted on the same chart.

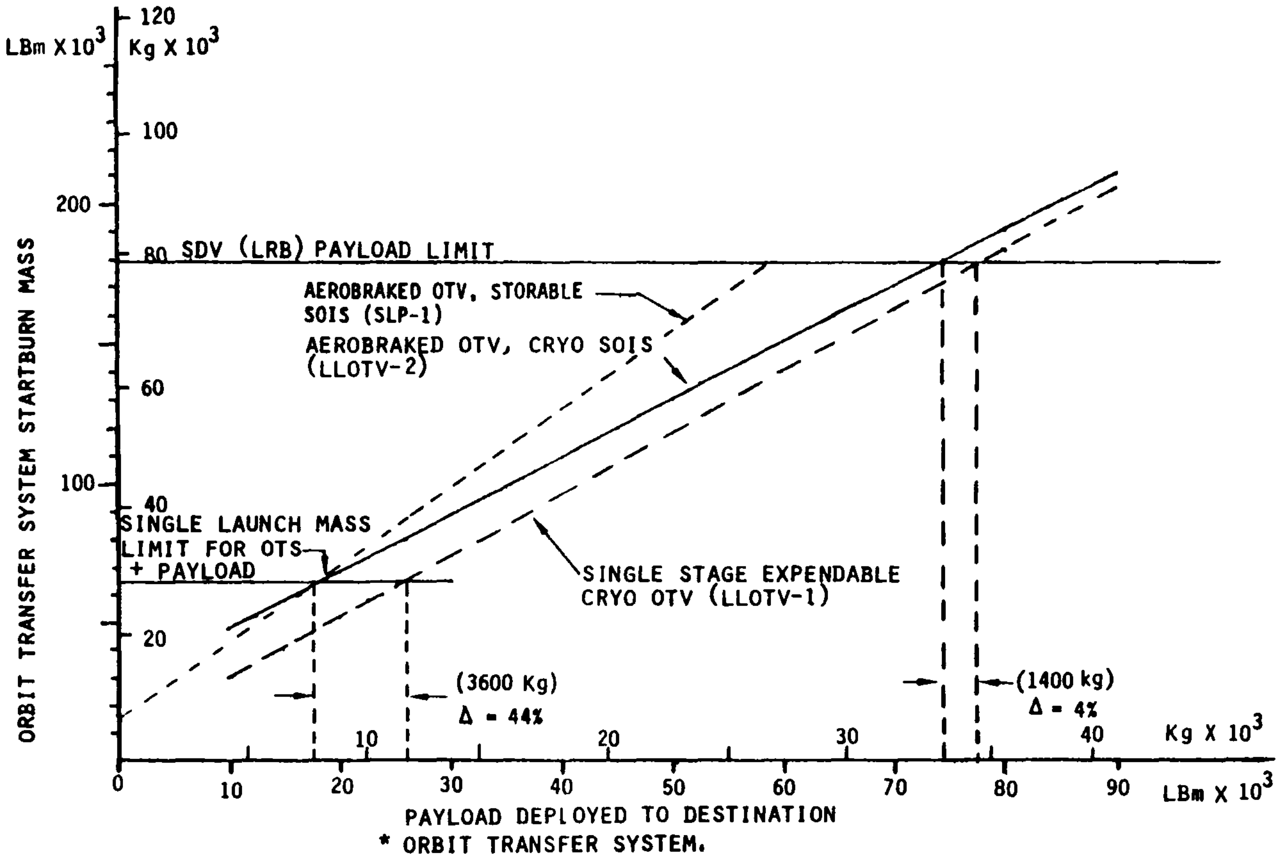


Figure 4.4-2: Parametric Performance Comparison of Staging Options

For the dual launch option for orbit transfer systems weighing about 82,000 kg, the performance of the single stage option is increased over the two stage option by about 4%. This was judged insufficient to warrant the increased cost per launch of the LLOTV option resulting in selection of the two-stage option for dual launch applications.

Evaluation for the single launch option traded the performance of the single stage LLOTV against the two stage system using the storable propellant SOIS. This comparison showed the single stage option as a clear winner, with a payload increase compared to the two-stage option of 44%.

Primary conclusions of the parametric performance comparison include:

1. The maximum performance dual launch option is the aerobraked injection stage used with a cryogenic propellant SOIS.

2. A maximum performance single launch option is a single stage cryogenic propellant LLOTV.

4.4.2 Selected Option Definition. Final orbit transfer systems selection for consideration in task 8 was based on performance and additional factors.

Three options compatible with single launch missions were chosen and are illustrated in Figure 4.4-3.

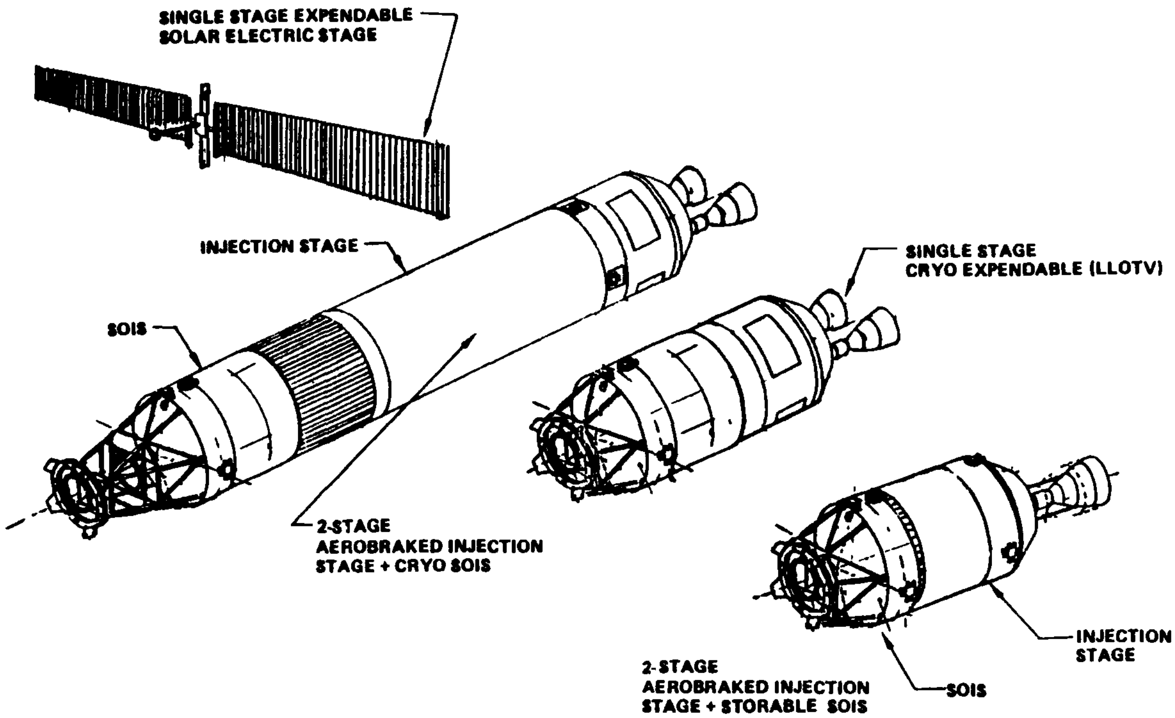


Figure 4.4-3: Selected Orbit Transfer Systems

1. The single stage cryogenic expendable LLOTV is the highest performance single launch chemical propellant option.

2. The two stage system using a storable propellant SOIS and an aerobraked injection stage is inferior in performance to the cryogenic propellant LLOTV but is closest to the reference system and was carried as a standard for evaluation of alternate systems.

3. The solar electric expendable stage has performance superior to the best chemical stage for the single launch option but is sufficiently different in cost to require cost comparisons at the total system level in Task 8.

4. The sole dual lauch option considered is also illustrated in Figure 4.4-3. It uses a large aerobraked recoverable injection stage and a expendable cryogenic propellant SOIS.

This combination offers the best overall performance in this size range. Electric systems were not considered for dual launch as efficient manifesting of multiple payloads and deployment of 81 metric ton electric stages would require extensive space based operations, which were considered beyond a scope of this study.

4.5 RESCUE SYSTEM DEFINITION

Definition of the rescue system was required to allow calculation of rescue mission capabilities of the basic orbit transfer system delivery vehicles.

Study of the MSFC reference system indicated potential mission cost savings in deletion of rescue peculiar hardware from the standard SOIS. Our study of rescue mission requirement in Task 3 indicated that the only SOIS rescue peculiar component required on every vehicle was a rendezvous radar transponder. Additional rescue provisions basically consisted of providing for a 3-year minimum life for beacon transponder and attitude control system. This provided minimum cost for most missions but left the basic SOIS capablities short of rescue mission requirements.

The approach evolved was to kit the basic SOIS to provide a rescue vehicle with required capabilities. This section describes the rescue kit installed on the SOIS to provide rescue vehicle capabilities. Integration of the rescue vehicle with the basic orbit transfer system is also described. The rescue system described is applicable to the dual launch large orbit transfer system but the basic kit approach is applicable to all orbit transfer systems considered.

4.5.1 Rescue System Requirements. Rescue mission requirements for the nominal rescue mission, derived from task 3 analysis of rescue mission profiles, are shown in Figure 4.5-1. Navigation to within terminal acquisition range of the target vehicle is accomplished by using the deep space network to track the onboard beacon transponder. Target vehicle and rescue vehicle relative

- NAVIGATE TO WITHIN TERMINAL ACQUISITION RANGE OF TARGET VEHICLE
- ACCOMPLISH TERMINAL RENDEZVOUS AND DOCKING
    - COMPUTER
    - IMU
    - RENDEZVOUS RADAR
    - MONITOR/COMMAND CAPABILITIES
- TRANSFER PAYLOAD TO RESCUE VEHICLE
    - EFFECTORS FOR PAYLOAD TRANSFER
- ORIENT SOIS FOR PLACEMENT MANEUVER

Figure 4.5-1: Rescue System Requirements

positions are monitored and the maneuvers required for closing are calculated on the ground and up linked to the rescue vehicle. The initial navigation phase is completed when the rescue vehicle arrives within 1000 km of the vehicle to be rescued.

The next requirement is accomplishment of terminal rendezvous and docking. The basic sequence of operation involves: (1) power up of the rescue kit, (2) a rescue radar scan to acquire the target vehicle transponder, (3) a range and bearing link to the ground where the closing trajectory for the rescue vehicle is calculated and uplinked back, (4) a closing maneuver from the initial acquisition range to 10 km range involving a total delta-V of approximately 40 m/sec and requiring about 10 hours, (5) initation of terminal rendezvous operations with an initial closing to 300m range followed by station-keeping and visual inspection by ground control using the onboard television camera and the high data rate downlink, (6) initation of the automatic rendezvous and docking sequence followed by completion of docking and waste payload transfer to the rescue vehicle, (7) undocking and a 5 m/sec clearance burn followed by orienting of the rescue vehicle to the final SOIS burn heading and loading of the SOIS coast timer, and (8) initation of the SOIS autonomous operations followed by the jettison of the rescue kit and sunshield. The SOIS then performs a normal placement to complete the deployment mission. Total duration for operations is approximately 18 hours.

This sequence dicates requirements for computer capability, an inertial measurement unit, rendezvous radar, and monitor/command capabilities including closed circuit television and a high data rate downlink to allow ground monitoring of terminal rendezvous operations. Payload transfer to the rescue vehicle involves requirements for docking provisions on both rescue vehicle and the vehicle to be rescued and effectors to accomplish payload transfer. The final requirement is to orient the SOIS for the placement maneuver and initiate the SOIS autonomous operations.

4.5.2 Chemical Propellant Rescue System. The rescue system evolved to meet these requirements consists of a rescue vehicle plus a standard delivery mission orbit transfer system. The rescue vehicle is assembled from a standard SOIS and a rescue kit.

Rescue Kit Configuration. The rescue kit is illustrated in Figure 4.5-2. Primary components include the reaction control system, propellant tankage, commu-

- EQUIPMENT COMPLEMENT
  1 N_2H_4 TANK
  2 TRANSPONDER
  3 20 WATT POWER AMP
  4 COMPUTER
  5 DATA BUS/SIGNAL CONDITIONER
  6 SIGNAL INTERFACE UNIT
  7 REDUNDANT INERTIAL MEASUREMENT UNIT (RIMU)
  8 RENDEZVOUS RADAR ELECTRONICS
  9 CCTV ELECTRONICS
  10 POWER DISTRIBUTION UNIT

- GROSS WEIGHT (Kg) 1582
  DRY WEIGHT 1153
  RESIDUALS 15
  RESERVE PROP 38
  NOMINAL PROP 376

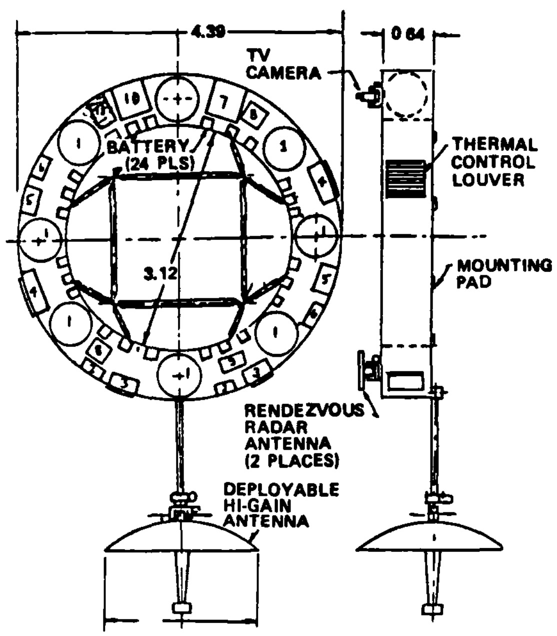


Figure 4.5-2: Rescue Kit Configuration

nication subsystem, a redundant inertial measurement unit, rendezvous radar electronics, computers and closed circuit television electronics unit which includes a high data rate RF subsystem and a deployable high gain antenna. Components are mounted in an equipment support ring which provides structural support and thermal control; the general arrangement is similar to the avionics equipment ring described in Section 4.3 for the standard injection stage. A 3.1m diameter hole in the center of the equipment section provides for transfer of the waste payload. Outboard mounting provides the widest possible field of view for the rendezvous TV camera, the gimbal mounted rendezvous radar antennas and a boom mounted high gain antenna. Additional structure consists of the struts used to interface the rescue kit with the SOIS mounted waste payload support structure.

Gross weight of the rescue kit is about 15,082 kg. The kit dry mass is about 11,053 kg. Consumables, primarily propellants for the reaction control system, amount to 426 kg. This propellant loading is adequate for all SOIS rendezvous and docking operations involved in rescue.

Rescue Vehicle Configuration. Assembly of the rescue kit and SOIS into a rescue vehicle is illustrated in Figure 4.5-3. The rescue kit is strut mounted to the SOIS payload support structure. Additional components include an aft sun-

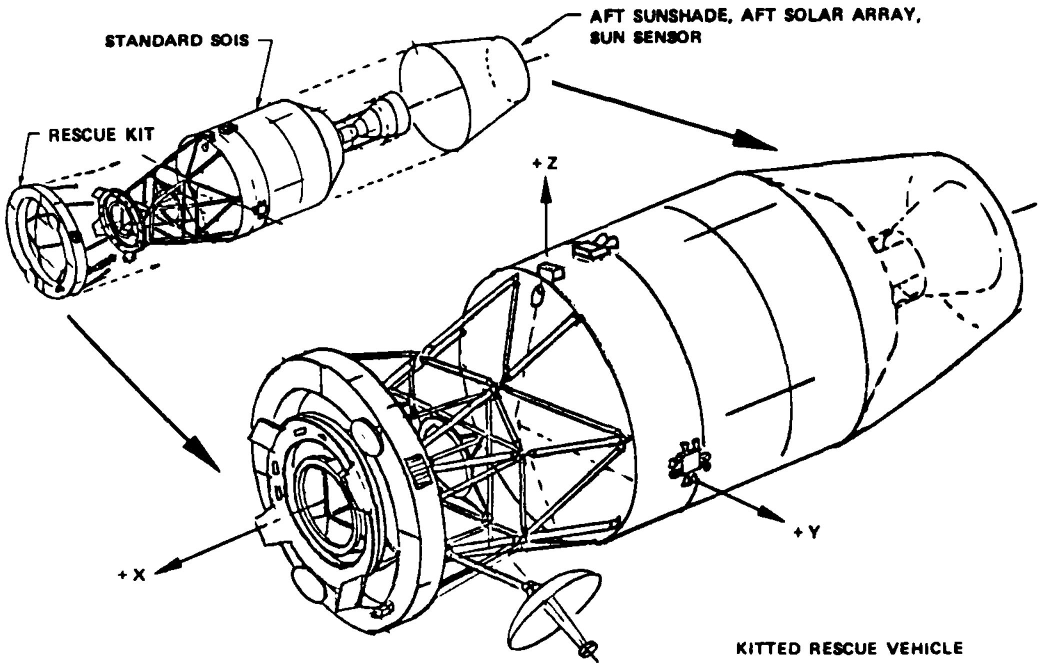


Figure 4.5-3: Rescue Vehicle is Modified from Standard SOIS

shade which mounts a solar array and Sun sensor used for vehicle attitude control and power during cruise and coast in the pursuit mode. At the conclusion of rescue operations the rescue kit and sunshade are jettisoned, converting the rescue vehicle back to a standard SOIS.

Key features including a summary mass statement for the complete rescue vehicle are shown in Figure 4.5-4.

	MASS, kg
STANDARD SOIS	14461 kg
–MAIN PROPULSION	
–ACS THRUSTERS	
–STAR SENSORS	
–PLUS–	
RESCUE KIT	1582
–TARGET ACQUSITION	
–RENDEZVOUS AND DOCKING	
–NAVIGATION	
–GROUND MONITORING	
–ADDED RCS PROPELLANT	
–ADDED ELECTRICAL POWER	
–JETTISONABLE	
ACTIVE DOCKING RING	95
–IMPACT ATTENUATION	
PAYLOAD TRANSFER MECH.	23
AFT SUNSHADE	78
–THERMAL CONTROL	
–SUPPLEMENTARY POWER	
–SUN SENSORS	
RESCUE VEHICLE GROSS	16239

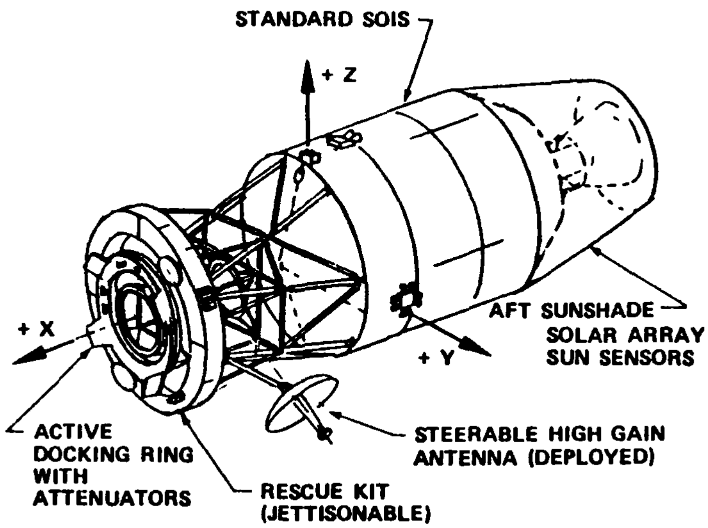


Figure 4.5-4: Rescue Vehicle Configuration and Summary Mass Statement

The aft sunshade provides thermal control for the vehicle cryogenic propellant during the Sun oriented coast portion of the pursuit mission when the X-axis of the rescue vehicle is pointed at the Sun. The sunshade also mounts a solar array and redundant Sun sensors used for vehicle power and pointing during the cruise portion of the mission. A steerable high gain antenna, shown in the deployed position, is used for the high data rate downlink required for closed circuit television monitoring of the rescue operations.

An active docking ring with shock absorbers to mitigate docking loads on the system is carried. The standard SOIS carries a passive docking ring hard mounted to the waste payload support structure. The docking ring is surrounded by the jettisonable rescue kit.

The mass of the standard SOIS is increased from approximately 14,400 kg to about 16,200 kg by the addition of rescue provisions. The largest increment is provided by the 1582 kg rescue kit. The active docking ring weighs 95 kg, mechanization for payload transfer an additional 23 kg, and the aft sunshade with its thermal control, supplementary power, and Sun sensors weighs 78 kg.

Rescue Orbit Transfer System. The rescue orbit transfer system consists of a pursuit configuration which carries the rescue vehicle to the target after injection and an injection configuration which injects the pursuit configuration to its initial transfer orbit.

The basic mission profile for rendezvous of the failed SOIS in heliocentric orbit is described in detail in Section 3.3.3 and illustrated in Figure 3.3-4. The initial injection inpulse injects the pursuit configuration into a transfer orbit (leg 3 to 4). A phasing maneuver by the standard SOIS component of the pursuit configuration applied at point 4 phases the pursuit vehicle for target interception at point 5. A second velocity inpulse provided by the standard SOIS matches the velocity of pursuit and target vehicle leaving the rescue vehicle within terminal acquistion range of the target. The expended standard SOIS is separated and the rescue vehicle (now in the configuration illustrated in Figure 4.5-4) completes the mission.

The pursuit configuration described is illustrated in Figure 4.5-5 and consists

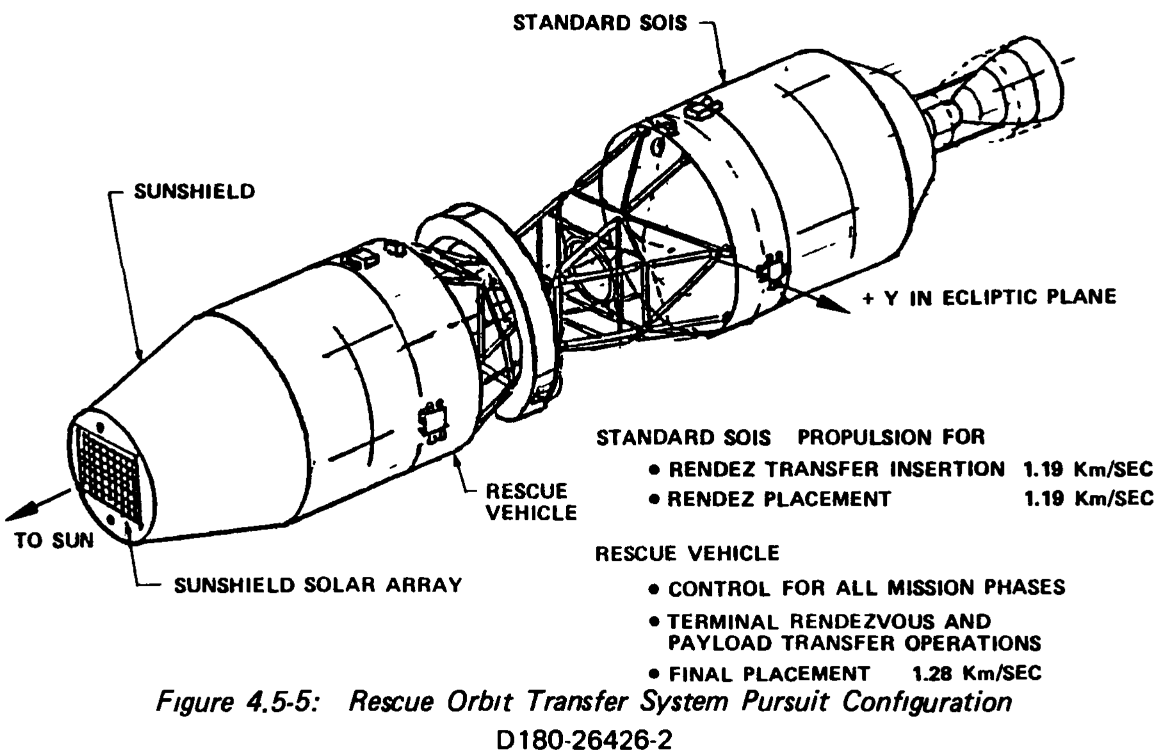


Figure 4.5-5: Rescue Orbit Transfer System Pursuit Configuration

of the rescue vehicle docked to the waste payload support structure of a standard SOIS. After injection and between maneuvers, the pursuit configuration flies with the rescue vehicle sunshade pointed at the Sun, allowing the rescue vehicle to shade the standard SOIS. All the control functions are provided by the rescue vehicle; the standard SOIS serves as a propulsion module only. Its propulsive capabilitity is adequate for the two 1.19-km/sec maneuvers required for the pursuit phase of the mission with the rescue vehicle itself performing the final 1.28-km/sec placement maneuver.

The same system is used with a slightly different delta-V split for the Earth orbit rescue mission illustrated in Figure 3.3-3.

The injection configuration of the rescue orbit transfer system is illustrated in Figure 4.5-6. The injection configuration is assembled on orbit from a

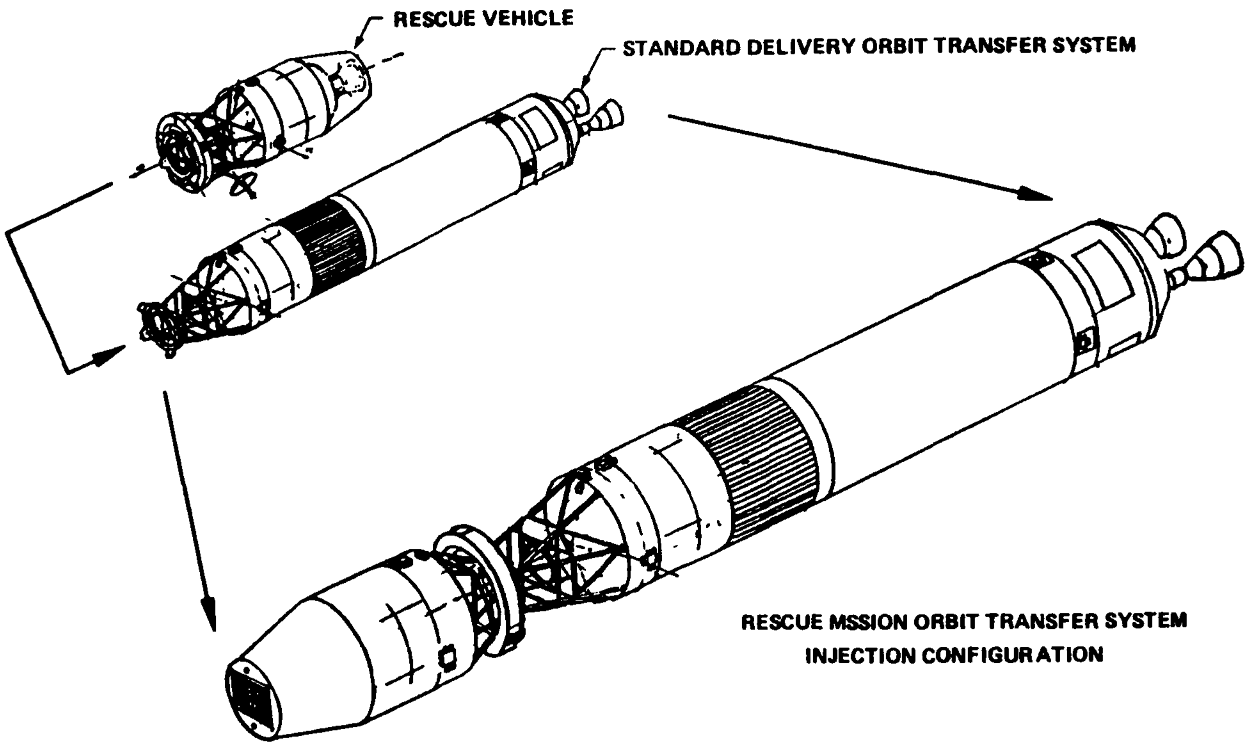


Figure 4.5-6: Rescue Orbit Transfer System: Injection Configuration

standard delivery orbit transfer system delivered to LEO by a shuttle derived cargo launch vehicle and a rescue vehicle which is carried up in the uprated shuttle orbiter. After orbiter rendezvous with the previously deployed standard delivery orbit transfer system, the rescue vehicle is deployed from the orbiter and in the first exercise of its functions, which serves as a final checkout, rendezvouses with and docks with the standard delivery orbit transfer system. Injection of the pursuit configuration to its initial transfer orbit is then accomplished by the large cryogenic injection stage which uses an aerobraking maneuver to return to low orbit for recovery by the orbiter.

4.5.3 Electric Propulsion Rescue Systems. Due to the relative lack of maneuverability of the electric orbit transfer system, the rescue kit for the electric vehicle is fitted with thrusters and used as a free flier capable of leaving the rescue electric vehicle and rendezvousing with the failed vehicle. The waste payload is then transfered from the failed vehicle to the free flier rescue kit which undocks from the failed vehicle and returns and docks with the rescue vehicle. Following a hard dock which attaches the waste payload to the rescue vehicle, the free flier rescue kit is jettisoned.

4.5.4 Contingency Rescue Study Areas. For reasonable system reliability, contingency rescues will comprise less than 1% of the total number of rescue missions carried out (see Section 3.3). Evolution of concepts for dealing with contingency rescue mission requirements is still required. Primary study areas include:

1. Tracking, acquisition and rendezvous with noncooperative failed vehicles.

2. Despin, docking and waste payload transfer from failed and randomly tumbling vehicles.

Various methods for dealing with these contengencies have been studied (Ref. 14) but a definite resolution of these issues requires further study.

5.0 LAUNCH VEHICLE SYSTEMS

5.1 INTRODUCTION

The primary objective of the launch vehicle systems task was to trade a full range of launch vehicle options in a systematic manner to define the optimum launch system for the space disposal option. Secondary objectives included definition of unique requirements imposed on launch vehicle systems by the space disposal mission.

The task began with a preliminary screening of launch vehicle options which resulted in selection of solid and liquid rocket boosted versions of the space shuttle orbiter and shuttle derived cargo launch vehicles as candidate configurations. Applicable references were reviewed to define key characteristics of the candidate vehicles in the areas of performance and cost. The candidate concepts were then traded in the areas of risk and cost to select the best candidate for the space disposal mission. These trades resulted in selection of two candidate systems.

1. For single launch missions, the liquid rocket booster uprated space shuttle.

2. For dual launch missions, the liquid rocket booster uprated space shuttle was teamed with a liquid rocket booster version of the shuttle derived cargo launch vehicle. This system showed significant life cycle cost saving over the reference launch mode for a wide range of assumptions on DDT&E and recurring costs, while preserving the low risk characeristics inherent in the winged orbiter.

5.2 CANDIDATE SYSTEM SELECTION

5.2.1 Launch System Survey. A survey of the available data on launch vehicles and past studies of space disposal of nuclear waste identified six categories of launch system candidates illustrated in Figure 5.2-1. Four major categories are represented.

(1) REFERENCE (29,500 KG PAYLOAD) SPACE TRANSPORTATION SYSTEM (STS)
(2) UPRATED (45,000 KG PAYLOAD) STS
(3) SHUTTLE DERIVATIVE CARGO LAUNCH VEHICLE USING SOLID ROCKET BOOSTERS [SDV(SRB)]
(4) SHUTTLE DERIVATIVE CARGO LAUNCH VEHICLE USING LIQUID ROCKET BOOSTERS [SDV(LRB)]
(5) HEAVY LIFT LAUNCH VEHICLE (HLLV) CONCEPTS
(6) SINGLE STAGE TO ORBIT (SSTO) LAUNCH VEHICLE CONCEPTS

Figure 5.2-1: Candidate Launch Systems

1. Reference and uprated versions of the present winged orbiter. These systems team the orbiter with an external tank and boosters, either solid rocket boosters as used on the present shuttle configuration or uprated liquid rocket boosters.

2. Shuttle derivative cargo launch vehicles which replace the winged orbiter with a recoverable propulsion pod and an expendable cargo shroud. The external tank and booster configurations used with the shuttle derived cargo launch vehicles are identical to those used with the winged orbiter, but deletion of the orbiter inert weight leads to significantly increased payload capabilities.

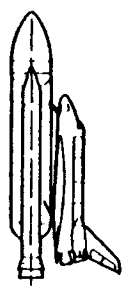
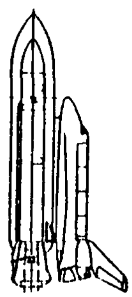
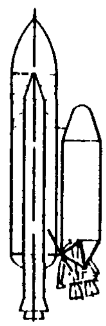
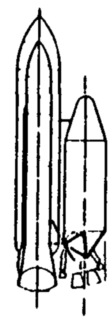
3. A variety of heavy lift launch vehicle (HLLV) concepts have been defined primarily in support of studies of the solar power satellite. The latest HLLV concepts defined involved winged fully reusable orbiters and boosters used in two stage configurations with gross lift off weights in the region of 5000 tons. These vehicles are capable of orbiting payloads on the order of 250 tons per launch.

4. Single stage to orbit (SSTO) concepts have also been investigated for the space disposal of nuclear waste. These concepts typically orbit relatively small payloads and depend on a high degree of reusability and greatly simplified mission operations to lower launch costs.

5.2.2 Preliminary Launch System Screening. A preliminary screening of the selected concepts rapidly eliminated the HLLV and SSTO concepts from further consideration.

While the HLLV offers attractively low launch costs per kilogram of payload, the scale of the vehicle and its development costs (estimated on the order of $10 billion) are sufficient to make it nonviable except in the context of support of an active solar power satellite program. Further study of a concept dependent for viability on the existance of a solar power satellite program was not considered to be consistent with the ground rules and assumptions governing the present study. A preliminary assessment of the utility of the HLLV for support of the space disposal mission was conducted by the Marshal Space Flight Center in Reference 20.

5.2.3 Candidate Launch System Description. The candidate launch vehicles resulting from the preliminary screening are illustrated along with key characteristics in the areas of risk, cost, and performance in Figure 5.2-2. Major elements of each candidate are listed.

KEY CHARACTERISTICS / LAUNCH VEHICLE	STANDARD SHUTTLE	UPRATED SHUTTLE	SHUTTLE DERIVATIVE (SOLID ROCKET BOOSTER)	SHUTTLE DERIVATIVE (LIQUID ROCKET BOOSTER)
CONFIGURATION				
MAJOR ELEMENTS	• ORBITER • EXTERNAL TANK • SOLID ROCKET BOOSTER (2)	• ORBITER • EXTERNAL TANK • LIQUID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • SOLID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • LIQUID ROCKET BOOSTER (2)
APPLICATION	CREW AND CARGO	CREW AND CARGO	CARGO	CARGO
CARGO BAY (M) DIA x LENGTH	4 57x18.3	4.57 x 18.3	7x24	7x24
PAYLOAD TO LEO Kg @ 370 Km	29,500 (5)	47,000	67,700	84,000
DDT&E	-0-	2 08 B	1.2 B	1 0 B (4)
PROD COST (1) (2)	0	460 M	335 M	790 M
COST/FLT (3)	28 6 M	27 0 M	22 0 M	18 7 M

ALL COST IN 1980 DOLLARS
(1) ORBITERS NOT INCLUDED
(2) FOR FLIGHT RATE 23/YR
(3) WHERE APPROP INCLUDES ET AND P/A MOD
(4) WHEN DEVELOPED IN ASSOCIATION WITH DEVELOPMENT OF UPRATED SHUTTLE.
(5) TO 270 Km

Figure 5.2-2: Key Characteristics Candidate Launch Vehicle System

Risk is expressed in terms of whether or not the vehicle possesses an intact abort capability. Winged orbiters in the event of a malfunction are able in most cases to jettison the external tank and glide back for a landing at the launch site or at an alternate field. Shuttle derivative vehicles do not possess this capability.

Performance is expressed in terms of payload bay size and the payload that the candidate vehicle can lift to a 28.5 deg inclination orbit at an altitude of 260 km.

Cost is expressed in terms of the design, development, test, and engineering required to implement the candidate, the production cost per unit, and the cost per flight.

Immediately apparent is the increased payload to low orbit capability of the shuttle derived cargo launch vehicles and the decreased unit cost due to deletion of the winged orbiter.

5.3 LAUNCH SYSTEMS SELECTION

Launch systems selection was accomplished in two steps: in the first, candidate launch systems were ranked for risk; the second step compared launch system concepts on the basis of life cycle cost. The risk evaluation led to the selection of the shuttle derived cargo launch vehicle as a supplement to the winged orbiter for dual launch mission scenarios. Cost trades led to the selection of two launch systems for further evaluation in Task 8.

5.3.1 Risk Assessment. Risk characteristics of the candidate launch systems are illustrated in Figure 5.3-1. Immediately apparent is the lack of abort capability for the shuttle derived cargo launch vehicles. The logical conclusion is that the orbiter will always provide lower risk for the waste payload.

The significantly lower cost per pound of the shuttle-derived cargo launch vehicle can still be used as part of a dual launch scenario with the waste payload carried in the winged orbiter and the orbit transfer system carried to low earth orbit by the shuttle-derived vehicle. This concept preserves the risk benefits of the winged orbiter but uses the increased cost efficiency of the shuttle-derived vehicle to lift the heavier part of the space system.

LAUNCH VEHICLE	INTACT ABORT MODES	COMMENTS
REFERENCE STS	RTLS, AOA, ATO	+ CONTINGENCY ABORTS
UPRATED STS	RTLS, AOA, ATO	+ AUGMENTED CONTINGENCY ABORT (LRB SHUTDOWN)
SDV (SRB)	NONE	NO IDENTIFIED ABORT CAPABILITY
SDV (LRB)	NONE	NO IDENTIFIED ABORT CAPABILITY

- A PRELIMINARY ASSESSMENT OF ESCAPE SYSTEM RELIABILITY INDICATES THAT AN ESCAPE SYSTEM RELIABILITY OF 0.9 IS OPTIMISTIC
  (F-111 ESCAPE CAPSULE DATA)
- SDV IS TOTALLY DEPENDENT ON ESCAPE SYSTEM FOR WASTE PAYLOAD RECOVERY AFTER ABNORMAL MISSION TERMINATION

Figure 5.3-1: Launch System Risk Characteristics

5.3.2 Cost Assessment. As a result of the risk assessment, four space systems were carried into the cost assesment. Two single launch options include the reference space shuttle and the uprated space shuttle using liquid rocket boosters. Two dual launch options team the reference space shuttle with the solid rocket booster version shuttle derived cargo launch vehicle and team the uprated space shuttle with the liquid rocket booster version of the shuttle derived vehicle.

Candidate system life cycle launch system costs for the 10 year reference mission were compared by calculating the number of flights required for each candidate system to lift the full mission cumulative mass to low Earth orbit. Cumulative mass requirements for transport to low orbit calculated for the Marshal Space Flight Center reference system are illustrated in Figure 5.3-2. The life cycle cost was calculated by multiplying the launch cost from Figure 5.2-2 by the total number of flights. This total was added to the DDT&E cost estimate to derive an estimate of each system's life cycle cost for the mission.

Figure 5.3-3 compares launch system life cycle costs and shows some of the key

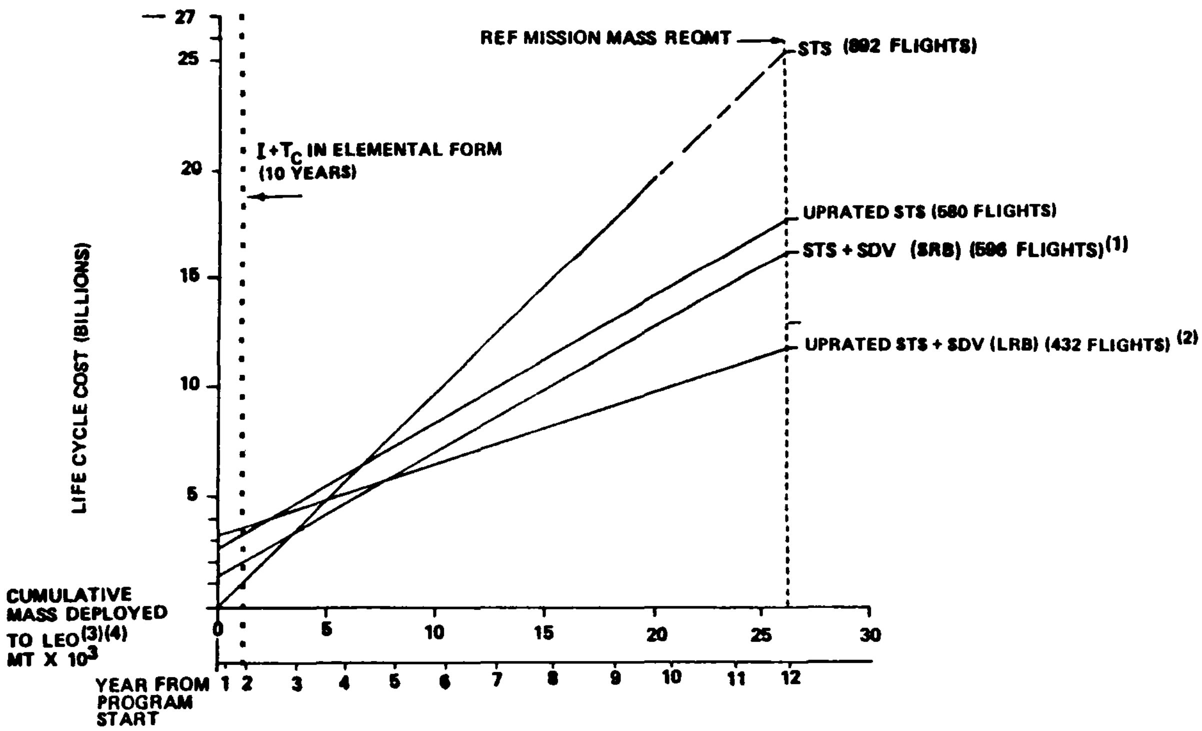
	YEAR AFTER PROGRAM START											
	1	2	3	4	5	6	7	8	9	10	11	12
NUMBER OF [1] UPRATED SHUTTLE FLIGHTS	10	20	50	50	50	50	50	60	60	60	60	60
EQUIVALENT [2] MASS TO [3] LEO (MT)	470	940	2350	2350	2350	2350	2350	2820	2820	2820	2820	2820
CUMULATIVE MASS TO LEO (MT.)	470	1410	3760	6110	8460	10810	13160	15980	12800	21620	24440	27260

[1] BASED ON MSFC REF. SYSTEM (200 GWE INSTALLED, PW-4b WASTE MIX)

[2] BASED ON RE-USABLE CRYO OTV FOR INJECTION, STORABLE PROPELLANT SOIS. MSFC WASTE PAYLOAD SYSTEM

[3] UPRATED SHUTTLE CAPACITY 47 MT/LAUNCH

Figure 5.3-2: Space Disposal of Nuclear Waste Launch Requirements



(1) INCLUDES 298 ORBITER AND 298 SDV FLIGHTS TO SUPPORT 298 MISSIONS
(2) INCLUDES 216 ORBITER AND 216 SDV FLIGHTS TO SUPPORT 216 MISSIONS
(3) BASED ON MSFC REF SPACE SYSTEM CONCEPT
(4) INCLUDES WASTE PAYLOAD AND PROTECTIVE SYSTEM, FLIGHT SUPPORT SYSTEM, ORBIT TRANSFER SYSTEM

Figure 5.3-3: Life Cycle Cost Comparison for Candidate Launch Vehicles assumptions used in their calculation. The ordinate shows life cycle cost in billions of dollars. Cumulative mass in thousands of metric tons is plotted on the abcissa along with years from program start for the reference mission scenario.

Launch costs for the four candidate systems are represented by the four lines running from left to right. The slope intercept representes DDT&E for initial deployment of the launch system; values range from 0 for the reference shuttle to about 3.2 billion for the uprated shuttle teamed with the liquid rocket booster version of the shuttle derived cargo launch vehicle. The slope of each line is proportional to the cost per flight.

Despite the highest initial DDT&E costs, the uprated shuttle/shuttle-derived vehicle team shows the lowest life cycle cost by a significant amount. This finding was tested by a series of sensitivity studies which (1) varied the mission model by adding non-waste disposal space missions; (2) let the shuttle derived cargo launch vehicle cost per flight equal cost per flight for the winged orbiter; (3) increased development cost for the shuttle derived vehicle and the liquid rocket booster by a factor of 2; and (4) imposed various combinations of these assumptions listed in Figure 5.3-4.

CASES EXAMINED

1) ADDED LaRC FOTV NOMINAL MISSION MODEL TO SDNW MODEL
2) SDV COST/FLT = ORBITER COST/FLT
3) SDV DEV COST = 2 X NOMINAL
4) LRB DEV COST = 2 X NOMINAL
5) 1 + 2 + 3
6) 1 + 2 + 4
7) 1 + 2 + 3 + 4

CONCLUSION:

- UPRATED SHUTTLE & SDV (LRB) SHOWS LOWEST LCC FOR ALL CASES
- LARGER PAYLOAD MODEL INCREASED ADVANTAGE
- PAYLOAD MODEL MAY BE VERY CONSERVATIVE (IT IS NOT VERY LIKELY THAT THE SPACE DISPOSAL MISSIONS WOULD BE TERMINATED AFTER 10 YEARS)

Figure 5.3-4. Launch System Life Cycle Cost Sensitivity Studies

Conclusions of the cost study are also summarized in Figure 5.3-4. The uprated shuttle teamed with the shuttle derived cargo launch vehicle using liquid rocket boosters shows the lowest life cycle cost for all cases studied. This advantage increases with increasing payload models and is maintained with smaller payload models down to about 20% of the reference mission studied. This concept was selected as the reference dual launch/launch system configuration for evaluation in Task 8.

The uprated shuttle using liquid rocket boosters was selected as the candidate for single launch vehicle mission scenarios in Task 8. This system shows a life cycle cost about two-thirds that of the reference space shuttle which uses solid rocket boosters and is identical to the launch system used in the reference space system.

5.4 LAUNCH VEHICLE UNIQUE REQUIREMENTS

Launch vehicle unique requirements identified for launch vehicles carrying waste payload ejection systems are listed in Figure 5.4-1. No specific unique requirements beyond strengthening for full payload weight landing were identified for orbiters carrying waste payload systems not requiring ejection.

- JETTISONABLE/QUICK CYCLE PAYLOAD BAY DOORS
  - FOR WASTE PAYLOAD JETTISON
  - MAY OCCUPY ONLY PART OF DOOR LENGTH
- ORBITER DESTRUCT SYSTEM (PREVENT HIGH SPEED IMPACT)
- BLAST CHANNEL/FIREWALL TO CONTAIN PLUME FROM WASTE PAYLOAD EJECTION SYSTEM MOTOR (MAINTAIN ORBITER STRUCTURAL INTEGRITY AFTER EJECTION)
- CREW ESCAPE PROVISIONS
- STRUCTURE AND LANDING GEAR STRENGTHENING FOR MAXIMUM PAYLOAD WEIGHT LANDING.

Figure 5.4-1: Unique Requirements for Launch Vehicles

6.0 WASTE PAYLOAD SYSTEMS

6.1 INTRODUCTION

The waste payload system consists of those systems providing payload protection functions that provide the necessary level of safety. Protection system concepts must provide protection against accident possibilities during all mission phases; launch site, ascent, low Earth orbit, and heliocentric transfer to the final destination. Primary systems involved include:

- o Containment
- o Thermal control
- o Radiation protection (gamma and neutron)
- o Thermal protection (reentry)
- o Impact protection (land and water)
- o Other auxiliary systems

Accordingly, the primary objective of the waste payload system study effort was to trade a full range of waste payload protection system options in these areas in a systematic manner to define the optimal waste payload protection system for the space disposal mission. Both removable and integral (non-removable) radiation shield containment systems were studied.

Key elements of the approach used included use of the containment/accident requirements from the 1979/80 Marshall/Battelle study. These requirements are documented for reference in Appendix E. Requirements were reviewed and modifications recommended. The results formed the basis for the remainder of the study effort. Trade studies were run to evaluate protection options in the area of radiation shielding and containment, waste payload ejection, and radiation shield removal. Results of these trade studies provided the basis for definition of reference waste payload systems for use in Task 8, total system integration effort.

Key results include:

1. Recommendation of additional requirements in the areas of radiation shielding and impact protection.

2. Adoption of a composite steel and graphite radiation shield used in both fixed and removable shield configurations.

3. Characterization of waste payload configurations and parametric weight estimating relationships for both integral and removable shields.

Two primary risk issues were identified:

1. Ground impact at terminal velocity. This problem can be assessed in future studies aimed at verification of the ability of the waste payload system concept to withstand ground impact.

2. Post-burial meltdown following ground impact. This problem appears unavoidable with waste payloads possessing thermal loadings comparable to the PW-4B waste mix used in this study. The only solution may be a waste mix with reduced thermal loading.

6.2 DERIVATION OF PROTECTION SYSTEM REQUIREMENTS

Protection system requirements were derived in three steps. The first was establishment of a classification system for accident events. Accident end conditions were then identified and the results used in reviewing the safety requirements defined for the reference system. Recommended changes included reduction in the waste payload radiation specification from 2 rem/hr at 1m to 1 rem/hr and the addition of orbiter crash conditions to the waste payload design conditions. These additions, plus the recommendations of the previous study, provided the basis for design of the waste payload protection system.

6.2.1 Accident Effect Classification. Accident classifications used in the study along with their defining criteria are illustrated in Figure 6.2-1. The classification and criteria were derived in modified form from Reference 21. Four classes of accidents were established. Class 1, the catastrophic accident, is defined as an accident producing an immediate radiological hazard

CLASS	DEFINING CRITERIA	NECESSARY CONDITIONS	PRODUCED BY
1 CATAS-TROPHIC ACCIDENT (CTA)	IMMEDIATE RADIOLOGICAL HAZARD TO POPULATION AND ENVIRONMENT; POTENTIAL FOR 10^2 - 10^3 WORLDWIDE EFFECTS OR SEVERE LOCAL DAMAGE, POTENTIAL LONG TERM EFFECTS	BREACH OF CONTAINMENT FOLLOWED BY ATMOSPHERIC RELEASE OF >10 KG OF WASTE FORM	MECHANICAL RUPTURE: BLAST OVERPRESSURE, HI-SPEED GROUND IMPACT, FRAGMENT IMPACT; MELTING: RE-ENTRY HEATING, THERMAL RADIATION, FIRE, MELTDOWN FOLLOWING BURIAL
2 CRITICAL ACCIDENT (CRA)	POTENTIAL RADIOLOGICAL HAZARD TO MARINE ENVIRONMENT	BREACH OF CONTAINMENT IN DEEP OCEAN	OCEAN IMPACT, SINKING FOLLOWED BY SLOW CORROSION OF CONTAINER
	SHORT TERM LOCAL NON-RADIOLOGICAL HAZARD TO POPULATION, ENVIRONMENT	GROUND IMPACT IN POPULATED OR FLAMMABLE AREA; NO LOSS OF CONTAINMENT	ATTAINABLE GROUNDTRACK NOT OVER OCEANS
3 NON-CRITICAL ACCIDENT (NCA)	NO DANGER TO POPULATION, ENIVRONMENT	OCEAN IMPACT WITH SUCCESSFUL FLOTATION AND RECOVERY OR LAND IMPACT IN UNPOPULATED, NON-FLAMMABLE AREA. NO LOSS OF CONTAINMENT	SHUTTLE CATASTROPHIC FAILURE FOLLOWED BY GROUND IMPACT IN SAFE AREA OR OCEAN IMPACT WITH SUCCESSFUL FLOTATION, RECOVERY
4 ABNORMAL MISSION TERMINATION (AMT)	MISSION NOT SUCCESSFUL	• ORBITER EXECUTES INTACT ABORT	STS OR WASTE PAYLOAD SYSTEM FAILURE

Figure 6.2-1: Space Disposal Accident Classification

to population or the environment with the potential for hundreds or thousands of world-wide effects or severe local damage. Class 2, the critical accident, may pose a potential long-term radiological hazard to the marine environment caused by a breach of containment in deep ocean or a short-term local nonradiological hazard to population and environment caused by ground impact or by the setting of fires. Class 3, the noncritical accident, represents no danger to either population or environment. Class 4, the abnormal mission termination, is a classification applied to an unsuccessful mission followed by an intact abort. Necessary conditions and agents producing accidents in these classes are summarized in the figure.

6.2.2 Protection System Objectives. The primary objective identified for the waste payload system is that of restricting the worst accident consequences to the Class 2 or critical accident category, or phrased differently, the elimination of the Class 1 or catastrophic accident. In Class 2 accidents the marine hazard can be eliminated by correct design of the waste payload system or by making provision for waste payload system recovery. The nonradiological hazards, crash and fire, in this category of accident are similar to those posed by the operation of commercial aircraft and cannot be completely eliminated from any launch system. Primary implications of this category of accident limitation include planning for land and ocean recovery.

6.2.3 Accident End Conditions. Accident end conditions identified for the launch vehicles considered in this study are illustrated in Figure 6.2-2 for systems having the capability for ejection of the waste payload in the event of an accident and for systems lacking ejection capability or the case where the ejection system failed. Possible accident environments and the primary environ-

EJECTION OPTIONS	POSSIBLE ACCIDENT ENVIRONMENTS	PRIMARY ENVIRONMENTAL STRESS RESULTING
EJECT	RETARDATION SYSTEM WORKS	LOW SPEED IMPACT; CONNECTION COOLING ASSURED
	RETARDATION SYSTEM FAILS	HIGH SPEED DIRECTIONAL GROUND IMPACT
NO EJECTION OR EJECTION SYSTEM FAILS	LAUNCH PAD ACCIDENT	BLAST, FRAGMENT, RADIATION HEATING, FIRE
	ORBITER INFLIGHT BREAKUP	BLAST, FRAGMENT, RADIATION HEATING, RE-ENTRY, HI-SPEED GROUND IMPACT
	ORBITER CRASH	HI-SPEED IMPACT, CRUSHING, POST-CRASH FIRE, COOLING?
	ORBITER LAND - INTACT, RUNWAY - DITCH OCEAN, LAKE - BELLY NO RUNWAY	ORBITER CRASH LOAD CONDITION. POST LAND COOLING?

Figure 6.2-2: Accident End Conditions

mental stress on the waste payload resulting are tabulated as a function of the type of accident. The environmental stresses identified were compared with those tabulated in Appendix E for the reference system accident requirements. As a result of this comparison we are recommending an addition to paragraph 2.5.1.3.3, On- or Near-Pad or Ascent Booster Accident. We would recommend adding to this a provision for shuttle crash resulting in an impact velocity distribution as shown in the impact speed and angle diagram illustrated in Figure 6.2-3, followed by a TBD crushing load imposed by the orbiter structure. This addition will provide for withstanding all orbiter crash environments and will contribute to making the accident environment list comprehensive.

The second recommendation was that the radiation limits imposed by 49CFR173.393 (surface radiation limit: 1000 mrem/hr, 1m from external surface of container) be applied for all mission phases when the possibility of personnel coming into contact with the payload during contingency situations exists. This would require a modification to paragraph 2.5.2.4, Flight Radiation Shielding, of the existing specification by changing the first paragraph to read "from 2 rem/hr at 1m" to "1 rem/hr at 1m."

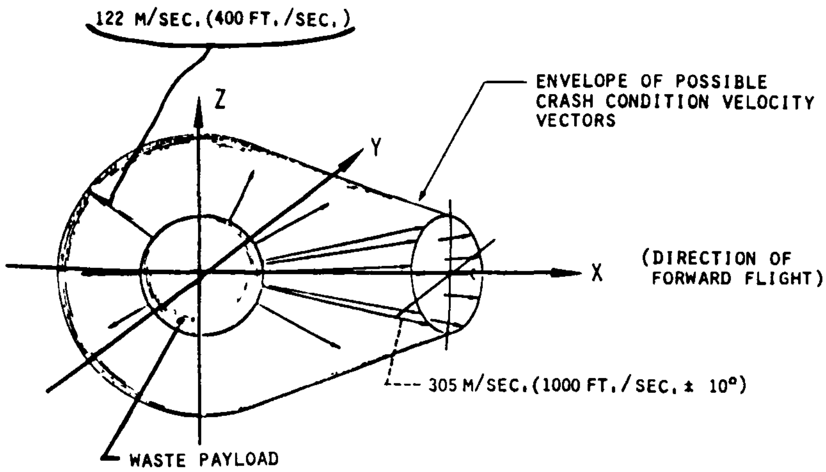


Figure 6.2-3: Recommended Design Criteria for Impact Angle

6.3 WASTE PAYLOAD PROTECTION TRADE STUDIES

Waste payload protection trade studies were conducted to determine the best approach to meeting the protection system requirements defined in Section 6.2. The effort began with definition of protection system options in key areas and proceeded with analyses conducted in the areas of radiation shielding, ejection from the launch vehicle, and shield removal, backed up by waste payload thermal analyses.

6.3.1 Protection Options. Basic options for protection of the waste payload are illustrated in Figure 6.3-1. Options exist in the area of radiation shielding material with the prime choice being between the use of uranium or steel for radiation protection. A second primary option is the ability to remove the radiation shield at some point in the mission to reduce the requirements for payload carried all the way to the destination. A final primary option is providing the capability to eject the waste payload in the event of a launch vehicle malfunction. Studies were conducted in each of these areas to determine which of these options would be carried to the total system trades in Task 8.

6.3.2 Containment/Radiation Shielding Trades. Primary objective of the shielding and containment study was to determine whether shielding and contain-

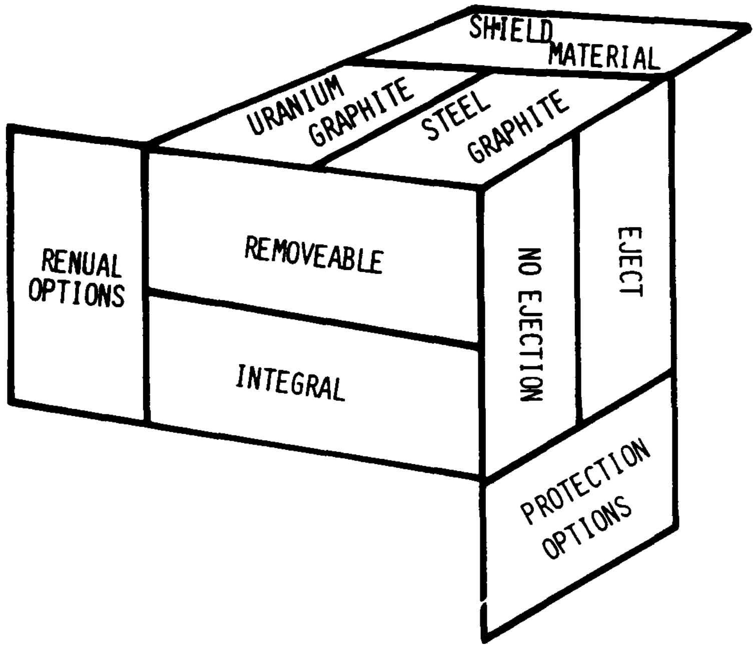


Figure 6.3-1: Waste Payload Protection Options

ment could be integrated in a synergistic manner to maximize protection of the waste payload while minimizing total system mass. A series of trades were conducted early in the study to evaluate the characteristics of candidate shielding/containment configurations.

The first step was the evaluation of concepts used in past studies. A survey of packaging arrangements used in past studies led to the selection of a single shape, or one piece waste payload. Considerations involved in the trade are illustrated schematically in Figure 6.3-2. The single shape arrangement offers the highest packaging density, fair cooling, and the simplicity of passive deployment.

A brief consideration of the full range of shapes considered in past studies led to selection of a spherical shape. Primary considerations in the trade illustrated in Figure 6.3-3 include ratio of packaging volume to surface area which is highest in the spherical configuration and the nondirectional nature of the spherical configuration in terms of sensitivity to ground impact or reentry heating.

A survey of past studies also led to the selection of the candidate shield materials illustrated in Figure 6.3-4 along with their key properties. Properties of the cermet waste form illustrated were derived by contacting Oak Ridge

	Packaging Density	Passive System Cooling	System Deployment
Hexagonally packed cylinders	Low	Good	Active
Telescoping Cylinders	Med	Good	Active
Sphere with multisphere	Low	Poor	Passive
Single Shape	High	Fair	Passive

Figure 6.3-2: Waste Payload Packaging Arrangement Trade

	Impact Sensitivity	Passive Cooling	Reentry Sensitivity
Long cylinder	Directional	Good	Directional
Short cylinder	Directional	Good	Directional
Disc	Directional	Good	Directional
Hemispherical	Directional	Good	Directional
Spherical	Nondirectional	Good	Nondirectional
Square	Directional	Good	Directional
Rectangular	Directional	Good	Directional
Cone	Directional	Good	Directional

Figure 6.3-3: Waste Payload Shape Trade

	CERMET (Hastelloy G)	304 SS	Steel (4130)	Uranium (Depleted)	Tantalum	Graphite
Density (g/cc)	6.7 (8.3)	7.98	7.8	18.7	16.6	2.25
Melting temperature (C^o)	1200 (1260)	1399	1482	1133	2996	-
Thermal conductivity (w/m-e)	14 (13.0)	16.4	29.7	30.0	55.0	~200
Coefficient thermal expansion ($x10^{-6}$ cm/cm/C^o)	- (16.5)	18.7	11.8	14.0	6.5	~1.3
Yield strength ($x10^{-6}$ n/cm^2)	- (32)	29	41	24	33	2.7
% elongation (in 5 cm)	- (61)	55	25	5	~30	<2
Corrosion resistance	Excellent	Excellent	Poor	Poor	Good	Excellent

Figure 6.3-4: Candidate Shield Materials and Properties

National Laboratories (ORNL). ORNL indicated that the cermet properties were approximated by the material properties of the superalloy, hastelloy G.

The candidate materials chosen were evaluated for shielding effectiveness in homogeneous and compound (gamma plus neutron shield) configurations. The evaluation was conducted in three steps.

1. Characterization of source terms for the baseline PW-4B waste mix incorporated in the cermet waste form.

2. A criticality estimate to verify the safety of the reference PW-4B/cermet waste form in the size range considered.

3. Shielding transport analyses used in the sizing and selection of candidate radiation shields.

<u>Baseline Waste Mix Parameters</u>. The baseline cermet parameters for this study were:

1. Total cermet mass, 5,075 kg
2. Cermet mass density of 6.7 g/cm^3
3. Cermet diameter of 113.1 cm

4. 58.7% waste loading with waste composition PW-4B (Table 1, Appendix H)
5. 41.3% metal phase composition as follows: 22.49% Fe, 6.13% Ni, 7.36% Cu 5.32% Mo

Characterization of Source Terms (Neutron, Gamma, and Thermal). The ORIGEN-79 computer code was used to calculate the radiation and thermal source terms for the baseline cermet. The radio-isotopic composition of the cermet (Table 2, Appendix H) was input to ORIGEN as a waste mass and a 100-day decay calculation was performed. Output was obtained for 10 intervals within the 100-day period.

It was found that some decay time was required to reach the secular equilibrium that would be characteristic of some of the isotopes in a 10-year-old waste mix. This was significant to the source calculations since one of these isotopes, Ba 137 metastable, is a primary contributor to the gamma source term but is not explicitly included in Table 2 (Appendix H). Other isotopes that were observed to reach secular equilibrium in the 100-day period were Pu 240, Np 239, and Pa 233. These made a slight contribution to the gamma spectrum.

Gamma spectrum from ORIGEN are output in three tables for three classes of nuclides: (1) light elements, (2) fission products, and (3) actinides. The total spectrum was obtained by summing over all three classes. Due to the equilibrium buildup of the isotopes noted above, there were some slight variations in the spectra over the 100-day decay time. To remain conservative in the calculations, the worst case spectrum from each nuclide class was used in adding up the total spectrum listed in Table 3 of Appendix H. This spectrum is regrouped for input to ANISN in Table 4 of Appendix H. The amount of overestimate possible by adding worst case spectrum is less than 2%.

The neutron spectrum could not be obtained directly from ORIGEN since ORIGEN only calculates the total neutron production rate and the contribution from each isotope. However, it was possible to approximate a neutron spectrum by noting that 98% of the neutron production calculated by ORIGEN is from spontaneous fission of Cm 244. Information on the Cm 244 fission spectrum from 0.3 to 12.8 MeV was obtained from Reference 22 and is listed in Table 5 of Appendix H.

For energies above and below 0.3 to 12.8 MeV, the spectrum was estimated by fitting an equation of the form,

$$N(E) = \left(\frac{2}{{}^{1/2}\ T^{3/2}} \right) kE^{1/2}\ e^{-E/T}\ \frac{n}{MeV}$$

where T = 1.265 MeV and k is a normalization constant. The above spectral information was normalized to the total rate given by ORIGEN of 3.05×10^{n} n/sec and restructured into a histogram spectrum for input to ANISN as listed in Table 6 of Appendix H.

The decay power calculated by ORIGEN is calculated for each of the three nuclide classes in terms of:

1. Total power (thermal plus gamma) for the nuclide class
2. Contribution from each nuclide to the total power of the class
3. Gamma power for the class
4. Contribution from each nuclide to the gamma power of the class

Summing the total power over the three material classes yielded a total cermet output of 97.9 kW. The gamma power contribution to this was 36.7 kW; the remainder was thermal. ANISN calculations indicate that up to 14% of the gamma power can escape the cermet but that this comes from the outer 5 cm of the cermet and is absorbed in the first 5 cm of any shielding material used. The result is that all the gamma rays are converted to heat in or near the cermet and the entire 97.9 kW represents a thermal source.

Criticality Estimate. A criticality estimate was performed for the 5075 kg cermet configuration using the XSDRN computer code. XSDRN is a discrete ordinates spectral code which uses a 123 group neutron cross section set to solve several types of neutron production transport problems one of which is the calculation of k effective.

Inputs to XSDRN which are important to the k effective calculation are (1) material compositions, (2) fission spectrum, and (3) medium surrounding the fissile material.

The material composition of the cermet is given in Table 13 of Appendix H. The composition as input to XSDRN is given in Table 14 of the same appendix. Some of the elements present in the cermet could not be included in the input to XSDRN because they were not contained in the XSDRN cross section set. The exclusion of these elements, which are predominantly fission products, was a conservative assumption that would result in a slightly higher estimate of k effective.

The neutron spectrum used for this calculation was the U 235 fission spectrum stored in the XSDRN cross section set. This spectrum was specified by utilizing the mixing table fission spectrum option. While the cermet spectrum is more closely a Cm 244 fission spectrum, the differences between the Cm 244 and U 235 fission spectra are not significant to the calculation.

The medium surrounding the cermet was assumed to be water since this would increase the moderated neutrons in the outer regions of the cermet and increase k effective.

The result of this XSDRN calculation was a criticality estimate of k_{eff} = 0.025.

Shielding Calculations. Shielding calculations for this study were performed using ANISN.

Cross sections used in ANISN were the CASK 40 group coupled neutron and gamma-ray cross section set (DLC-23E) obtained from the Radiation Shielding Information Center of Oak Ridge. To reduce the storage and run time requirements of ANISN, the CASK element cross sections were blended into material cross sections and stored on tape using the TAPEMAKER routine. The element cross sections available in CASK are shown in Table 7 of Appendix H. The element compositions of the materials to be used in ANISN are given in Tables 8 and 9 of Appendix H. As is evident in the tables, not all the elements present in the cermet are available in CASK. To account for some of the "missinq" elements, other elements of comparable (or conservative) neutron and gamma cross sections were substituted. The substitutions were as follows:

Si substituted for P

Zr substituted for Rb, Sr, and Y

Mo substituted for Tc, Ru, Rh, and Pd

Sn substituted for Ag, Cd, and Te

No substitutions were made for Cs, Ba, La, Ce, Pr, Nd, Pm, Sm, Eu, Gd, Np, Am, U 236, Pu 241, or Cm. They constitute 21% of the total cermet weight. It was believed to be more practical within this study to accept a conservative calculation by excluding these elements than it would be to attempt to approximate their cross sections by elements or mixture of elements in the CASK set.

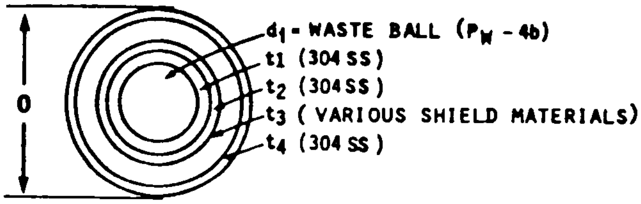
The flux to dose conversion factors used in ANISN were the Snyder-Neufeld factors for neutrons and the Henderson factors for gammas. These are listed in Table 10 of Appendix H.

When performing ANISN calculations, the following program conditions were used:

Spherical geometry	IGE	=	3
Reflection at left boundary	IBL	=	1
Vacuum at right boundary	IBR	=	0
Fixed source eigenvalue	IEVT	=	0
Distributed source	IQM	=	1
Zero flux guess	3* all	=	0

For the bare ball cases, an S4 quadrature was used. For the shielding calculations, an S8 quadrature was used. These are listed in Table 11 of Appendix H. Baseline configurations for a compound (neutron and gamma) cermet shield were also developed using ANISN. Various shield materials and cermet payloads were input to ANISN to determine the shield mass required to achieve a 1 rem/hr dose rate at 1m from the surface. While an attempt was made to choose the gamma and neutron shield thicknesses that would minimize the shield mass, a complete mass optimization calculation was not performed. Informal estimates by the BAC physics staff indicate that such an optimization could conceivably reduce the shield mass by as much as 5%.

Results of the analysis of isotropic and compound shields are presented in Figure 6.3-5. The results are all based on the reference 5075 kg cermet waste



Shield Material	Dose Rate		d (cm)	t_1	t_2	t_3	t_4	Total Wt. (kg)
Uranium	η (1.0)	t (cm)	-	.64	.13	12.7	.13	-
(18 7 g/cc)	γ (0 0)	d (cm)	113.1	114 38	114 64	140.04	140.3	-
	$\eta+\gamma$ = 1.0	W (kg)	5075	209	43	12138	64	17529
Steel	η(82)	t (cm)	-	-	-	27.0	-	
(7 98 g/cc)	γ(.18)	d (cm)	113.1	-	-	167 1	-	-
	$\eta+\gamma$ = 10	W (kg)	5075	-	-	13450	-	18525
Tantalum	η(.89)	t (cm)	-	.64	13	11.8	.13	-
(16 6 g/cc)	γ(.09)	d (cm)	113 1	114.38	114 64	138.24	138.50	-
	$\eta+\gamma$ = .98	W (kg)	5075	209	43	4867	62	15256
Uranium /Graphite	η(.52)	t (cm)	-	.64	.13	U 6.13 S .13 GR 10	.5	-
(18 7/2 25 g/cc)	γ(48)	d (cm)	113 1	114 38	114.64	126.40 127 16 147 16	148.16	-
	$\eta+\gamma$ = .98	W (kg)	5075	209	43	U 5257 S 53 GR 1339	275	12251
Steel /Graphite	η(0.5)	t (cm)	-	-	-	S 22 45 Gr 5.0	.5	-
(7 98/2 25 g/cc)	γ(48)	d (cm)	113.1	-	-	158.0 168.0	169.0	-
	$\eta+\gamma$ = .98	W (kg)	5075	-	-	10436 939	356	16806
Tantalum /graphite	η(.32)	t (cm)	-	.64	13	Ta 8.5 Gr 10	.13	-
(16 6/2 25 g/cc)	γ(58)	d (cm)	113.1	114.38	114.64	131 64 151.64	151.90	-
	$\eta+\gamma$ = .90	W (kg)	5075	209	43	6732 1421	75	13555
Uranium /H_2O	η(.18)	t (cm)	-	.64	.13	U 6.3 S .13 H_2O 8	.5	-
(18 7/1 0 g/cc)	γ(73)	d (cm)	113.1	114 38	114.64	127 24 127.50 143.5	144.5	-
	$\eta+\gamma$ - .91	W (kg)	5075	209	43	5418 53 462	260	11520
Steel /H_2O	η(15)	t (cm)	-	-	-	S 22.5 H_2O 5 0	.5	-
(7 98/1 0 g/cc)	γ(68)	d (cm)	113.1	-	-	158.1 168 1	169.1	-
	$\eta+\gamma$ .83	W (kg)	5075	-	-	10467 418	356	16316
Tantalum /H_2O	η(06)	t (cm)	-	64	.13	Ta 8.5 H_2O 7 0	.5	-
(16 6/1.0 g/cc)	γ(73)	d (cm)	113 1	114 38	114 64	131 64 145 64	146 66	-
	$\eta+\gamma$ - .99	W (kg)	5075	209	43	6732 423	268	12750

Figure 6.3-5: Candidate Radiation Shield Comparison

form. A preliminary optimization of each shield configuration has minimized the shield mass to the specified 1 rem/hr at 1m dose rate. Several conclusions are apparent:

1. Compound shields are lighter than the isotropic shield due to their increased efficiency in reducing the neutron dose rate.

2. Water is the most effective moderator considered.

3. The lightest configuration by a narrow margin is the uranium water composite shield at 11,520 kg. The next lightest is the uranium graphite composite shield at 12,251 kg. The heaviest shield at 16,806 kg is the steel graphite composite shield, which is also judged as the safest.

The final shield evaluation considered survivability, mass, and availability of shield material. Survivability considerations ruled out water as a moderator. Any accident condition which breached the outer shell containing the water would lead to loss of moderation and immediate increase in neutron dose rate. Tantalum was ruled out as a serious competitor by availability. The mass required to fabricate a tantulum shield for the reference 5075 kg waste form is worth about $5 million.

A comparison of the remaining steel graphite and uranium graphite candidates shows that manifesting constraints and impact tolerance both tend to favor the steel shield.

In the case of a dual payload launch where a single launch vehicle carries two payloads, sizing of the waste payload for passive thermal dissipation at the reference destination keeps the payload under the uprated shuttle payload constraints for both steel and uranium shield concepts.

Crash survivability considerations are an even more important influence. Past studies indicate orbiter crash load attenuation requirements for the uranium shield are undefined and the pyrophoric nature of uranium is an unresolved problem. Similarly, the ability of uranium to tolerate fragment penetration is undefined. In contrast high strength steel provides maximum resistance to impact requirements, is not flammable, and possesses an extensive literature on fragment tolerance for a variety of steel formulations and heat treats.

In conclusion, survivability considerations must dominate, resulting in adaptation of the steel graphite composite shield as the sole candidate for further consideration in the study.

6.3.3 Ejection From Launch Vehicle. The ejection option effects on launch conditions are immediately apparent. Staying with the orbiter requires surviving the most rigorous possible environments. Ejection enormously simplifies the survival problem but is not 100% reliable.

The effect of ejection system reliability was examined using a review of data on USAF's experience with escape capsules used for aircraft crew escape. The study illustrates that ejection systems on the whole are very reliable, but problems cannot be totally eliminated.

Systems can be damaged through blast overpressure or through fragment impact. These possibilities are unavoidable. An alternative possibility is the failure of components such as squibs and parachutes, or human error, for instance the misrouting of shielded mild detonating cord (SMDC) lines. These factors can be

minimized but not eliminated. The conclusion is clear: a system which depends on ejection to eliminate Class 1 or catastrophic accident events cannot guarantee that no catastrophic event will occur.

Even though not 100% effective, the effect of an ejection system can still be substantial.

Risk to the waste payload is a function both of accident magnitude and the frequency of occurrence. An ejection system which is not 100% reliable cannot be relied on to decrease the magnitude of insults delivered to the waste payload, but can be a valid means of reducing the frequency of occurrence of insults, and thus the probability of a catastrophic accident event. The need for such a reduction cannot be established until the reliability of launch systems is more firmly established.

In conclusion, the waste payload system must survive all accident environments without ejection to ensure against catastrophic accident events; but ejection can still contribute to overall risk reduction. Evaluations in Task 8 have shown that the cost of the ejection option for selected options ranges between zero and 20%. A final evaluation or balancing of this cost against reduction in risk can be accomplished only after the launch vehicle operation reliability is firmly established. Until that time the waste payload protection system providing for ejection should be carried as an alternate in all systems considered.

6.3.4 <u>Shield Removal Trades</u>. Primary issues in shield removal are illustrated in Figure 6.3-6. The integral or nonremovable shield is clearly superior in every area except performance.

	REMOVEABLE	INTEGRAL
STRUCTURAL INTEGRITY	MASS PENALTY FOR IMPACT CONDITION NEEDS ASSESSMENT	HIGHEST; BEST TOLERANCE TO IMPACT
RADIATION EFFECTS ON AVIONICS	ADD 770 KG OF SHADOW SHIELDING FOR RESCUE CRITICAL AVIONICS PROTECTION (PLACEMENT STAGE ONLY)	NO RADIATION EFFECTS
THERMAL EFFECTS	CORE MELTING IN CONTINGENCY SITUATIONS, NO MELT THRU	NO CORE MELTING ONCE ON ORBIT FOR ANY CONTINGENCY ONCE CARGO BAY DOORS OPEN
EVA ACCESS	TERMINATES WITH SHIELD REMOVAL	AT ANY TIME IN EARTH ORBIT
MECHANIZATION	SHIELD REMOVAL MECHANIZATION RELIABILITY IS CRITICAL-NO EVA ACCESS FOR CONTINGENCY REPAIR	NO CRITICAL MECHANIZATION. EVA BACKUP AVAILABLE AT ALL TIMES IN LEO
SIZING	LIMITED TO 5075 KG OF REF. CERMET WASTE FORM	ALLOWS 10% INCREASE IN WASTE FORM MASS WITH PASSIVE HEAT DISSIPATION
PAYLOAD DEPLOYED PER MISSION	HIGHEST	44-50% OF REMOVEABLE SHIELD PAYLOAD (DEPENDS ON SYSTEM OPTIONS)

Figure 6.3-6: Shield Removal Factors

The total performance penalty may not be as great as it first appears. An initial look would indicate the penalty appears to be on the order of 250%, but when quantified at the system level for the dual launch reference system considered here, the results, as shown in Figure 6.3-7, show a less drastic decrease in performance. The figure, for both integral and separable shields, shows the number of orbiter launches, shuttle derived cargo launch vehicle launches, orbit transfer vehicle flights and expended stages for deploying six payloads to the destination. When all factors are taken into account the

	PAYLOADS DEPLOYED TO DESTINATION	ORBITER LAUNCHES	SDV LAUNCHES	OTV FLIGHTS	SOIS STAGES EXPENDED	TOTAL COST	COST PER PLD
INTEGRAL SHIELD CONCEPT	6	3 @ 27.0 M	3 @ 18.7 M	3 @ 5.0 M	3 @ 6.0 M	170.1 M	28.35 M
		81.0 M	56.1 M	15.0 M	18.0 M		
SEPERABLE SHIELD CONCEPT	6*	3 @ 27.0 M	1 @ 18.7 M	1 @ 5.0 M	1 @ 6.0 M	110.7 M	18.45 M
		81.0 M	18.7 M	5.0 M	6.0 M		

● ACTUAL COST/PLD IS APPROXIMATELY 30 - 50% HIGHER FOR INTEGRAL SHIELD

* MAXIMUM, MAY DECREASE DUE TO RADIATION SHIELDING FOR AVIONICS

Figure 6.3-7: Manifesting Mismatch Decreases the Cost Advantage of Shield Removal

actual penalty ranges from about 30% to 50% rather than 250%. These costs must be balanced against the decreased risk involved in the non-separable shield.

In conclusion, shield removal deserves consideration at the total system level. At the total system level the removable shield will be used only with the ejectable waste payload protection system in view of its decreased survivability. The intregal shield will be evaluated both with and without the ejectable waste payload protection system.

6.3.5 Protection System Option Evaluation Conclusions. In conclusion, the following ground rules have been adapted for the remainder of the study as a result of the protection system option trade studies.

1. Steel graphite composite radiation shields have been adapted for all options.

2. Both integral and removable shields will be evaluated for total system impact in Task 8.

3. Integral shield options will be evaluated both with and without ejectable waste payload protection systems to determine total system impact in Task 8; removable shield options will be evaluated only in conjunction with use of the ejectable waste payload protection system.

6.3.6 Thermal Analysis. It was decided very early in the study that the waste package should be spherical to facilitate fabrication, shielding, design, and thermal analysis. One of the first questions to be answered was how large can the waste sphere be? Due to the high density, the limitation was not shuttle cargo bay volume so the maximum allowable from the thermal standpoint was determined. It was assumed that active cooling required would be available during ground and in-shuttle operations. Therefore, the first sizing was done based on deployment of the waste sphere at 0.85 AU with no shielding. Surface properties, especially emissivity, are very important since, in space, all heat rejected is by radiation. Various coatings were investigated for suitability of application to the hot (up to 982°C) cermet surface. The surface treatment selected was flame sprayed aluminum oxide (Al_2O_3) which melts at 2037°C and has absorptivity/emissivity values of 0.21/0.8 which degrade to 0.26/0.8 after 7-1/2 months based on Telstar I experience. Another criterion which affected the allowable waste sphere size was that the sphere center temperature must not exceed the cermet melting temperature of 1200°C. The surface temperature of a sphere in space far removed from any other body is given by:

$$T_s = 155.4 \left(\frac{q_i}{4 \pi r^2} + \frac{111}{N^2} \frac{\alpha}{\varepsilon} \right)^{\frac{1}{4}} - 460 \qquad (1)$$

The internal dissipation (q_i) is a function of waste volume and hence increases by the cube of the radius. From Carslaw and Jaeqger (Reference 23) the temperature rise from surface to center of a sphere with internal heat generation is given by:

$$\Delta T = \frac{Q}{6k} r^2$$

where Q = 0.13 W/cm^3, k = cermet conductivity, and the allowable surface temperature is

$$T_s = 1200^{\circ}C - \frac{Q}{6k} r^2 \qquad (2)$$

The intersection of equations (1) and (2) for T_s defines the maximum radius as shown in Figure 6.3-8 and is about 600 mm.

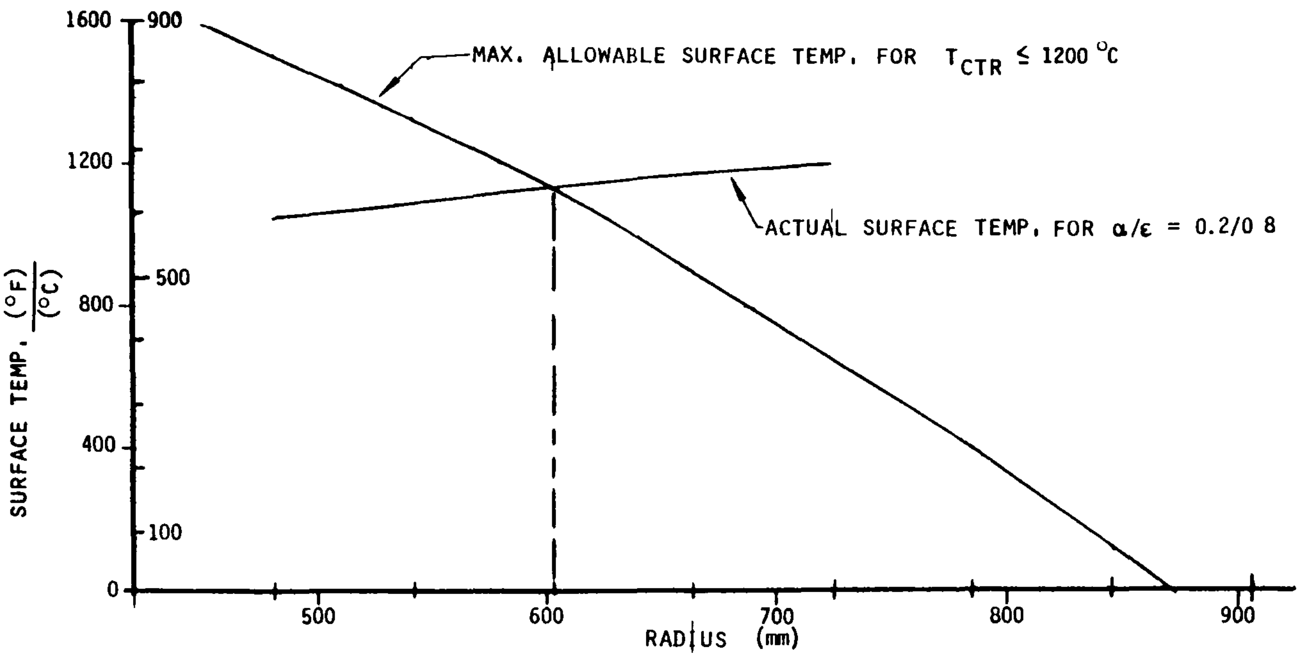


Figure 6.3-8: Unshielded Waste Sphere Radius Limit Determination at 0.85 A.U.

The next step in the thermal analysis was to determine how shielding would affect the temperatures at the center and of the other materials involved. A large number of shielding schemes were examined. Various combinations of such materials as uranium, steel, tantalum, graphite, and water were evaluated. Two concepts were selected: one an integral or nonremovable shield and another which features a removable shield. In the interests of conservatism a waste sphere radius of 560 mm was selected for the shielding studies.

The integral shield concept remains with the waste package. It features cooling tubes in the steel just outboard of the cermet/steel interface which will provide cooling until the waste payload separates from the shuttle. The cermet/steel interface will be fused together during fabrication to enhance thermal conductance. The graphite will be mechanically fastened to the steel to maximize contact conductance. Based on data available, a contact conductance of 88 W/m^2-^{o}C was selected.

The removable shield incorporates cooling tubes in the shield portion near the cermet interface. Contact conductance can be maintained high (88 W/m^2-^{o}C) except at the steel/cermet interface. In this area it was assumed a gap will exist with emissivity equal to 0.8 on the cermet side and 0.9 on the shield side. At the temperatures involved, the equivalent conductance was 63 $W/m^2\text{-}^oC$.

Steady state temperature profiles for both the removable and integral shield approaches are shown in Figure 6.3-9. The solar heating rate at 0.85 AU and

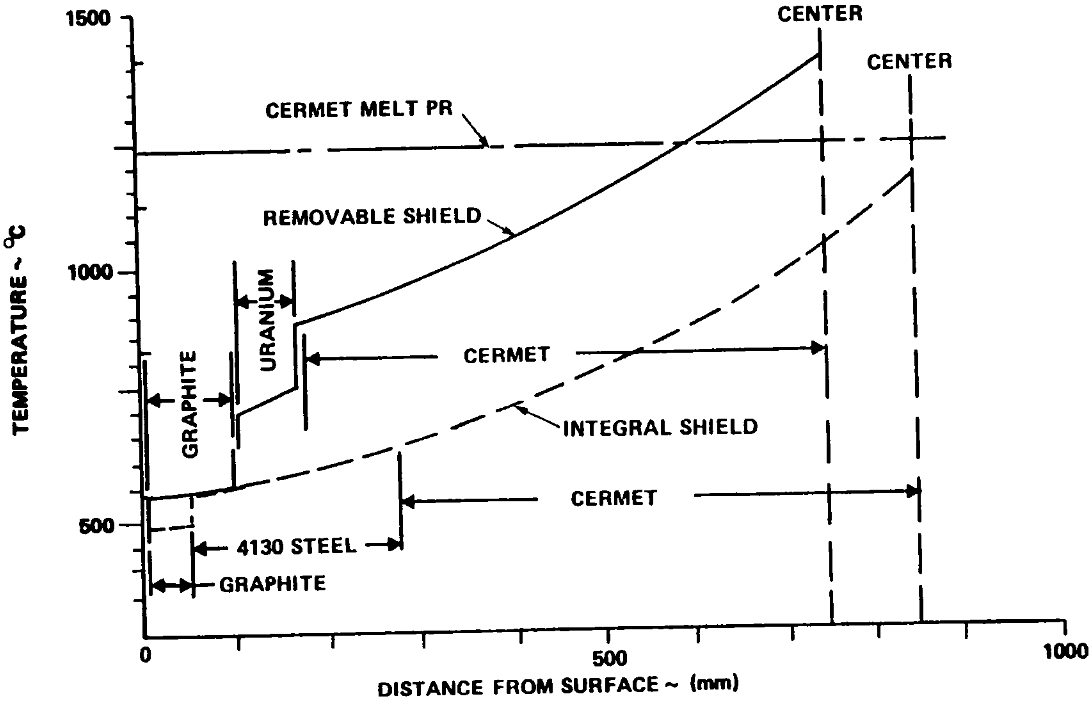


Figure 6.3-9: Removable & Integral Shield Temperature Profiles for Leo & 0.85 A.U.

the average of the solar, albedo, and Earth radiation heating rate at LEO for spherical shapes with absorbtivity/emissivity = 0.26/0.8 are sufficiently similar (3.15 and 3.2 MJ/hr, respectively for this case) to result in nearly identical temperatures; thus, the temperature profiles shown in Figure 6.3-9 are applicable for both. The LEO case would represent a contingency situation where the waste sphere was stranded in LEO: e.g., following a failure of the orbit transfer vehicle. Therefore, with either shielding approach, there is sufficient heat loss from the waste through the shield in LEO or at 0.85 AU to assure containment even though there is some cermet melting for the case of the removable shield.

A final evaluation was conducted of the effect of increasing cermet radius for the integral shield. If no cermet melting is to be permitted, the cermet radius cannot exceed 500 mm.

Reentry. Various shuttle mishaps could result in reentry of the waste sphere into the atmoshere. Two approaches to cope with reentry were examined.

The Aero-Shell (Waste Payload Protection System). One reentry approach studied was to enclose the waste sphere in an aero-shell which would be ejected under certain conditions of shuttle failure. The aero-shell protects the waste sphere from the heat of re-entry and, at a predetermined altitude, would be separated from the waste payload by a parachute which would then carry the waste to a soft landing. This scenario is similar to that used by the Viking lander when entering the Martian atmosphere. Two aero-shell designs were developed; one having a 45 deg conical section half-angle and the other 60 deg. The aero-shell and afterbody configurations selected were determined based on the requirement that they be stable in only one orientation; i.e., conical section leading regardless of initial entry orientation.

Re-entry temperature histories and temperature profiles were calculated for these shapes using the Boeing CHAP computer code. Profiles were based on use of phenolic silica ablator, chosen for its combination of toughness and low thermal conductivity. The two aero-shells perform similarly from the thermal standpoint but, for a diameter-limited configuration (imposed by the shuttle), the 45 deg aero-shell provides greater volume. A number of locations on the aero-shells were evaluated to determine required ablator thickness. The variations from the selected thickness were minor and, since a constant ablator thickness of 18 mm does not pose a significant weight penalty and simplifies the design, that thickness was chosen.

Integrally Shielded Waste Sphere. An alternate re-entry approach studied takes advantage of the high temperature capabilities of the integrally shielded waste sphere concept. A re-entry profile was determined for the waste sphere using the drag coefficient curve from Figure 6.5-1. During the re-entry process 0.53 inches of graphite are lost due to ablation. Note that the thermal capacitance of the 8.84 inches of steel under the graphite is such that, at the steel/cermet interface, there is almost no increase in temperature. Therefore, it is apparent that the integrally shielded waste sphere can survive the reentry heating environment.

Post Entry

Parachute Landing Case. When the waste package is soft landing via parachute, the waste payload will be held up above the ground, regardless of orientation, by a framework to maximize heat loss by convection and radiation. The framework will offer minimum blockage to air movement and radiation interchange with the surroundings.

The heat lost from a large hot sphere in air, assuming free convection and average sink temperature for radiation equal to air temperature, can be expressed as:

$$q_{tot} = A_s \quad \epsilon\sigma(T_s^4 - T_A^4) + 0.19\,(T_s - T_A)^{5/4}$$

In the steady state condition, the heat given up to the surroundings must equal the heat dissipated by the waste; in this case 100 kW. If an air temperature is assumed, the surface temperature can be determined for a given radius sphere. Figure 6.3-10 presents the temperature profiles for the integral and removable waste spheres for these conditions. It is apparent that the steady state temperatures do not impose a containment concern.

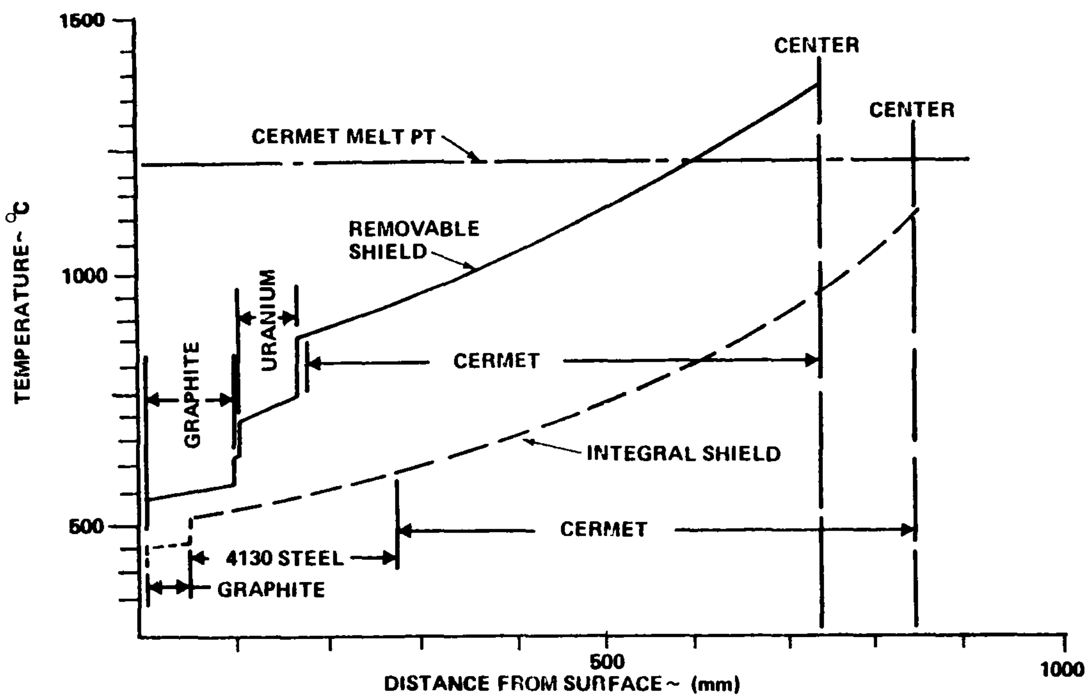


Figure 6.3-10: Removable & Integral Shield Temperature Profiles for Free Convection Cooling in 100°F Air at Sea Level

High Speed Impact. The reentry case where the waste sphere is allowed to impact the Earth's surface at high speed (454 m/sec) presents a critical thermal design problem. The problem results from impact on and subsequent burial in soft, dry soil. Using the conduction shape factor, S, approach to determine allowable sphere burial depth for steady state where $q_i = kS\,(T_S - T_O)$

$$k = 0.35\ W/m^2\text{-}^oC$$

$$\text{and } S = \frac{4\pi r}{1 - \frac{r}{2\rho}}$$

it was immediately evident that, if the sphere is buried beneath the surface at all, the dissipation rate must be less than or equal to 5 kW, far below the actual 100 kW. An analysis of the transient response was then undertaken to determine the rescue time available before loss of containment. Figure 6.3-11 illustrates the sphere temperature response after a burial of from 4 to 8 hr.

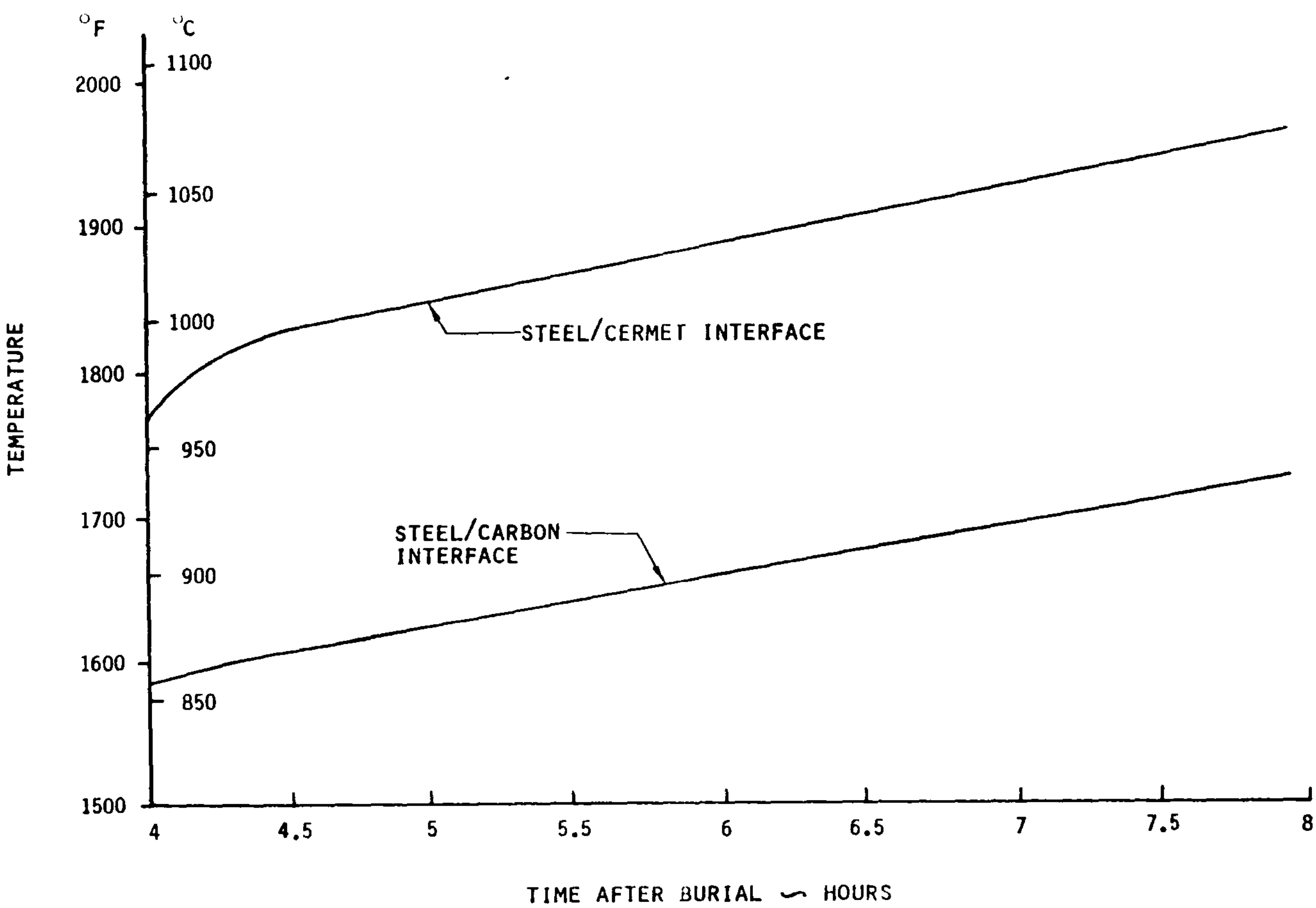


Figure 6.3-11 Integral Shield Temperature History after Post-Entry Serial

Extrapolating the linear portions of the curve leads to the conclusion that approximately 27.5 hr are available for rescue before containment is lost.

Thermal Analysis Conclusions. Conclusions reached include:

1. Leaving the radiation shield on the waste sphere indefinitely has very little thermal impact on the cermet.

2. The integrally shielded waste sphere can survive reentry.

3. Removable or integrally shielded waste spheres can survive indefinitely on the ground, cooled by free convection and radiation.

4. The integrally shielded waste sphere will remain intact for 27.5 hr after burial in dry soil following reentry.

6.4 WASTE PAYLOAD CONFIGURATION DESCRIPTION

Using the results of the waste payload system trades in the areas of radiation shielding, shield removal and the results of the thermal analysis task, integral and removable shield configurations were characterized by drawings and parametric weight estimating relationships. A dual waste payload system was also defined for use in dual launch concepts.

6.4.1 Integral Shield Configuration. The integrally shielded waste payload general arrangement is illustrated in Figure 6.4-1 for the reference 5075 kg cermet waste form. A primary feature is the high strength steel gamma radiation shield and primary container. The shield is welded into a one piece integral shell around the cermet waste form. The fit is tight enough to assure intimate thermal contact between waste form and shield; integral cooling tubes for active cooling are incorporated in the shield. The shield is 225 mm thick and weighs 10,436 kg.

The graphite neutron shield is fabricated as over 100 individual tiles which are fastened to the steel shield with 1.3 cm high strength steel bolts. The tiles are overlapped to prevent leak paths for neutron emission. The steel bolts provide sufficient tension to preload the graphite against the shield t

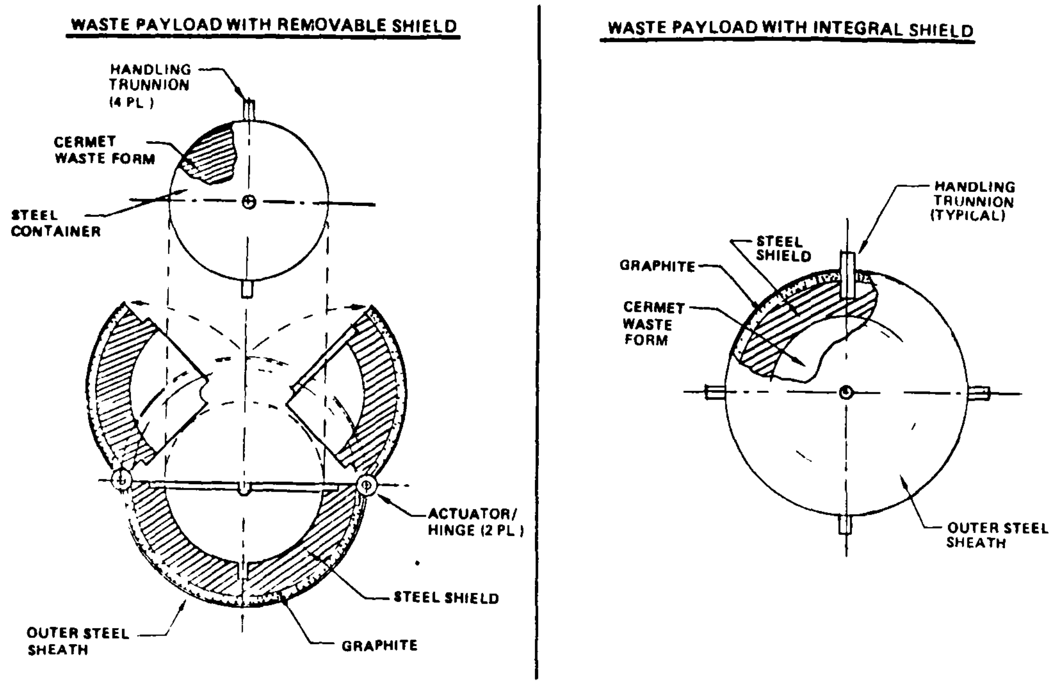


Figure 6.4-1: Candidate Waste Payload Configurations

a level adequate for high thermal conductivity between steel and graphite. Graphite plugs over the steel bolts are used to prevent the bolts from becoming conduits for neutron emission. In addition to its neutron shielding function the graphite serves as contingency entry protection to protect against inadvertent entry. Total graphite thickness is 50 mm and the graphite shield weighs 939 kg.

The outer sheath is 5 mm of corrosion resistant steel which is bonded to the outside of the graphite tiles. The sheet is divided into individual plates for each tile. A flame sprayed aluminum oxide thermal control coating is applied to the outside of the shield to provide corrosion resistance and the correct ratio of absorbitivity to emissivity for thermal control during long term space storage.

Six 50-mm diameter handling trunnions fabricated from high strength steel provide for mechanical interfaces with the flight support systems and the waste payload transfer mechanisms. The trunnions are interfaced with the steel shield for structural integrity, and pass through the carbon neutron shield, posing a neutron leakage problem which requires further investigation.

The total individually shielded waste payload system is approximately 1500 mm in diameter and weighs 16,806 kg.

6.4.2 Removable Shield Configuration. The removable shield configuration is also illustrated in Figure 6.4-1. The composite radiation shield is similar in thickness to the integral shield. The cermet waste form is enclosed in a separate 64 mm thick reinforced stainless steel container; trunnions are set in the cermet and protrude through the radiation shield for independent support of the waste form during shield removal. Shield removal is mechanized by offset hinges which allow the two articulated segments to swing open, providing sufficient clearance for waste form removal. The shield is overlapped at the joints to contain radiation leakage. Overall size and shape of the removable shield are similar to the integral shield. The diameter of the waste form without the shield is approximately 1140 mm.

6.4.3 Dual Waste Payload. The dual waste payload configuration is illustrated along with key dimensions in Figure 6.4-2. Two of the integrally shielded

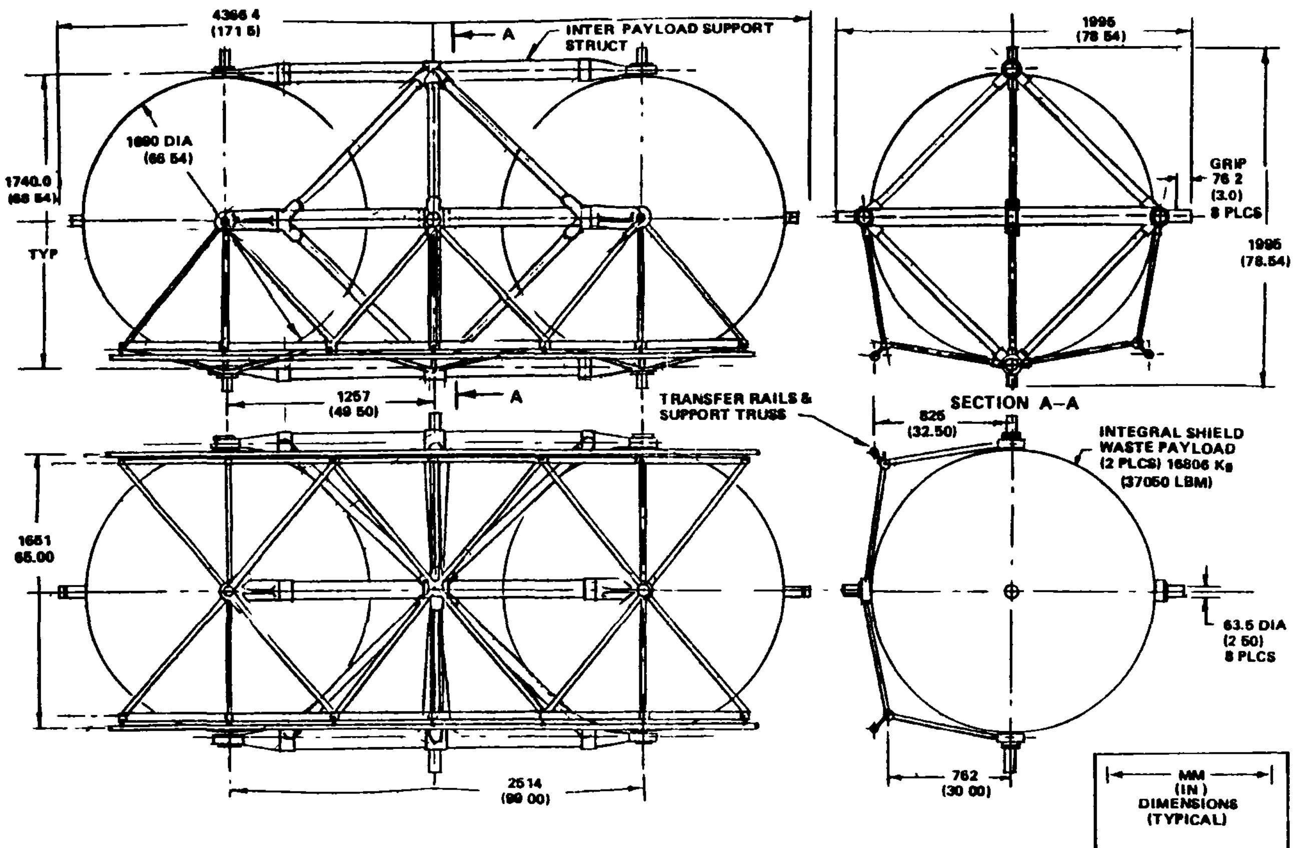


Figure 6.4-2 Dual Waste Payload Configuration

waste payloads described in Section 6.4.1 are connected by a titanium inter-payload support structure.

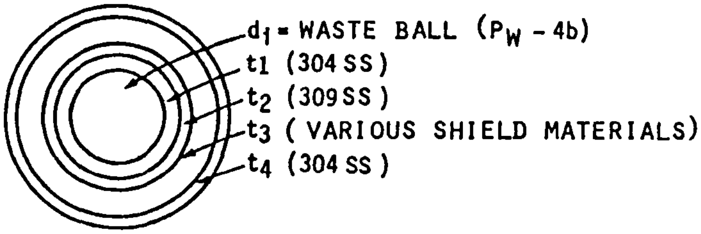
The interpayload support structure provides pickups for the trunnions of the individual waste payloads and accomplishes three primary functions: (1) it supports the waste payload in association with the dual waste payload flight support system during launch in the cargo bay, (2) it carries guide rails and other provisions for waste payload transfer from the flight support system to the orbit transfer system and between a failed solar orbit insertion stage and a rescue vehicle, and (3) the space frame structure provides a free field of view for passive waste form thermal dissipation. The titanium struts are sized to carry all loads at the full waste form equilibrium temperature.

Additional support trunnions mounted on the inter payload support structure in combination with the outboard trunnions on each of the individual waste payloads provide the structural interface between the support structure and the flight support system. Total mass of the inter payload support structure, including guide rails and associated hardware is calculated at 136 kg.

6.4.4 Waste Payload Weight Estimating Relationships. Weight estimating relationships for the waste payloads including waste form containment and shielding were prepared by optimizing shields for three waste form masses and using the resulting data points to prepare a curve showing the weight estimating relationship between waste form mass and total waste payload mass.

Waste payload point designs were established for waste form masses of 2500 and 10,000 kg by using the ANISN code to optimize the selected steel graphite composite shield configuration.

Results of the optimization are illustrated in Figure 6.4-3. Total masses range from 10,126 kg for the 2500 kg waste form to 27,853 kg for the 10,000 kg waste form. The penalty in shielding efficiency involved in smaller waste forms is evident. As a percent of total waste payload system mass, the waste form represents only 25% for the 2,500 kg waste form as compared to 30% for the reference 5075 kg waste form and 36% for the 10,000 kg mass waste form.



SHIELD MATERIAL	STEEL/GRAPHITE (7.98/2.25 G/cc)			STEEL/GRAPHITE (7.98/2.25 G/cc)		
DOSE RATE @/HR,1 MTR	η.39 γ.58	$\eta+\gamma$ (.97)		η.55 γ.43	$\eta+\gamma$ (.98)	
	t (cm)	d (cm)	w (kg)	t (cm)	d (cm)	w (kg)
d_1 (cm)	-	89.32	2500	-	141.8	10,000
t_1 (cm)	-	-	-	-	-	-
t_2 (cm)	-	-	-	-	-	-
t_3 (cm) S	21.5	132.32	6703	23.0	187.8	15,762
t_3 (cm) Gr	5.0	142.32	667	6.0	199.8	1,593
t_4	.5	143.32	256	.5	200.8	
TOTAL WT(kg)	-	-	10,126	-	-	27,858
			22,324			

Figure 6.4-3: Waste Payload Characteristics for Alternate Waste Form Sizes

These point designs were the basis for the weight estimating relationship curves plotted in Figure 6.4-4. The ordinate shows the mass of the composite waste payload including containment and radiation shield. The mass of the waste form is plotted on the abscissa. Individual curves are shown for the removable shield and the integral shield. Waste payload mass is slightly higher for the removable shield due to mass included to account for hinges and structural closure.

Using these curves the total waste payload mass can be derived for any given waste form mass. These curves were the basis for the payoff estimates in Task 8 where the waste form mass per mission was derived given the initial waste payload mass delivered to the destination.

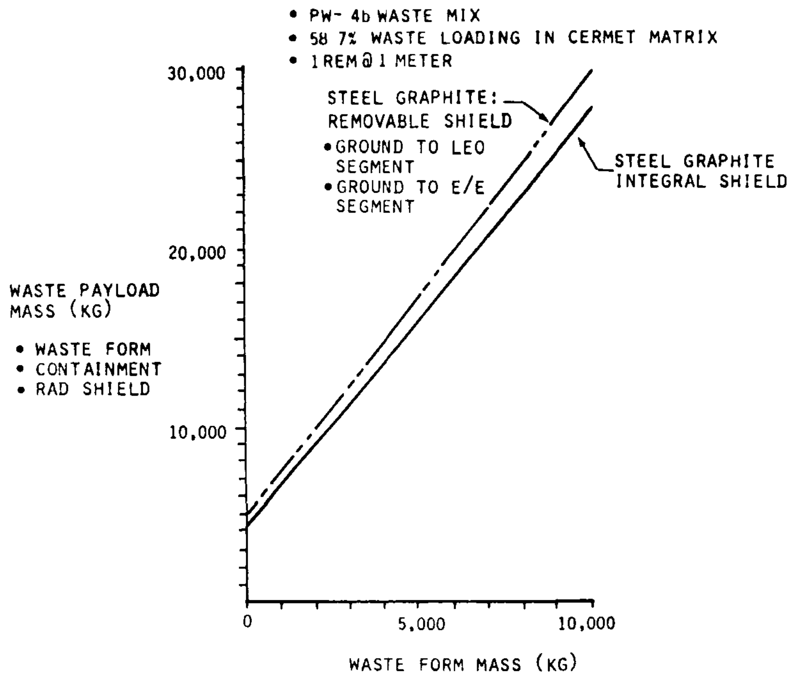


Figure 6.4-4: Waste Payload Weight Estimating Relationships for Selected Shield Options

6.5 WASTE PAYLOAD PROTECTION SYSTEM DEFINITION

The objective of the waste payload protection system is: (1) to provide intact recovery of the waste payload, restricting the consequences of any space transportation system accident to be non-critical action category, (2) to minimize the expense of waste payload reflight by preventing damage to the waste payload due to transportation system accidents; and (3) to decrease the probability (however low) of a Class 1 or critical accident.

The basic waste payload protection system has been well characterized in previous studies by the Marshall Space Flight Center, in particular Reference 19, and our objectives in this study were limited to: (1) characterization of the system for use in Task 8 total system trades and (2) evaluation of an alternate aerodynamic configuration offering better characterization of reentry properties as a result of extensive use on programs such as Galileo and Pioneer Venus. Additional benefits of the alternate configuration included: more internal volume, provisions for entry wake clearance, compatibility with the Boeing CHAP reentry heating analysis code, and perspective on the sensitivity to shape of the waste payload protection system in general.

6.5.1 Aerodynamic Configuration Trades. In order to provide a comparison of the alternate shape with the shape used in the reference system, the following constraints were applied to the alternate configuration:

1) Length to be less than or equal to the Marshall Space Flight Center reference system length.

2) Diameter less than or equal to 4.57m for compatablity with the diameter of the STS cargo bay.

3) Center of gravity distance from the extreme nose of the vehicle to be greater than or equal to that of the Marshall reference system.

4) Demonstrated static stability through hypersonic, supersonic, and subsonic flight regimes.

5) Wake clearance on the entry wake of 30 deg minimum.

Two candidate shapes were considered. The first, a blunt cone with a half angle of 60 deg and a ratio of nose radius to base radius of 0.68; the second, a blunt cone with a half angle of 45 deg and a ratio of nose radius to base radius of 0.68. Concepts were evaluated to determine whether alternate shapes would provide reduced terminal velocity or reduced ablator mass and were investigated for the possibility of providing increased stability margin.

Curves illustrating the drag coefficient for both shapes as a function of mach number are illustrated in Figure 6.5-1. At low mach numbers, the drag coefficient is almost identical. Computer trajectory runs indicated that the terminal velocity was not a strong function of shape for the blunt nose conical entry bodies.

Ablator mass was calculated using the CHAP reentry thermal analysis program. Ablator thickness of 18 mm was determined as adequate for both systems using the selected phenolic silica ablator. Comparison of the ablator mass for the two configurations indicated no significant difference in total system mass between the two concepts, with a maximum difference of approximately 147 kg for total system masses on the order of 20,000 to 30,000 kg.

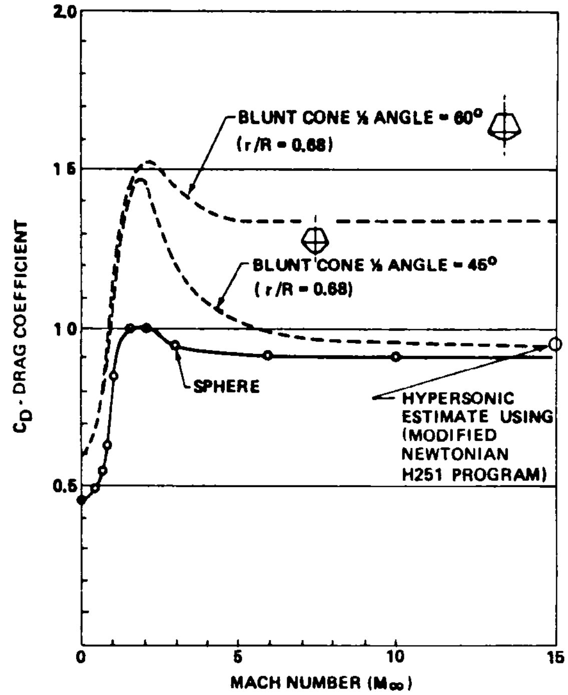


Figure 6.5-1 Drag Coefficient vs. Mach No. for Candidate Shapes

A comparison of the stability margin of the two candidates is illustrated in Figure 6.5-2 which shows the aft CG limit for static stability measured in

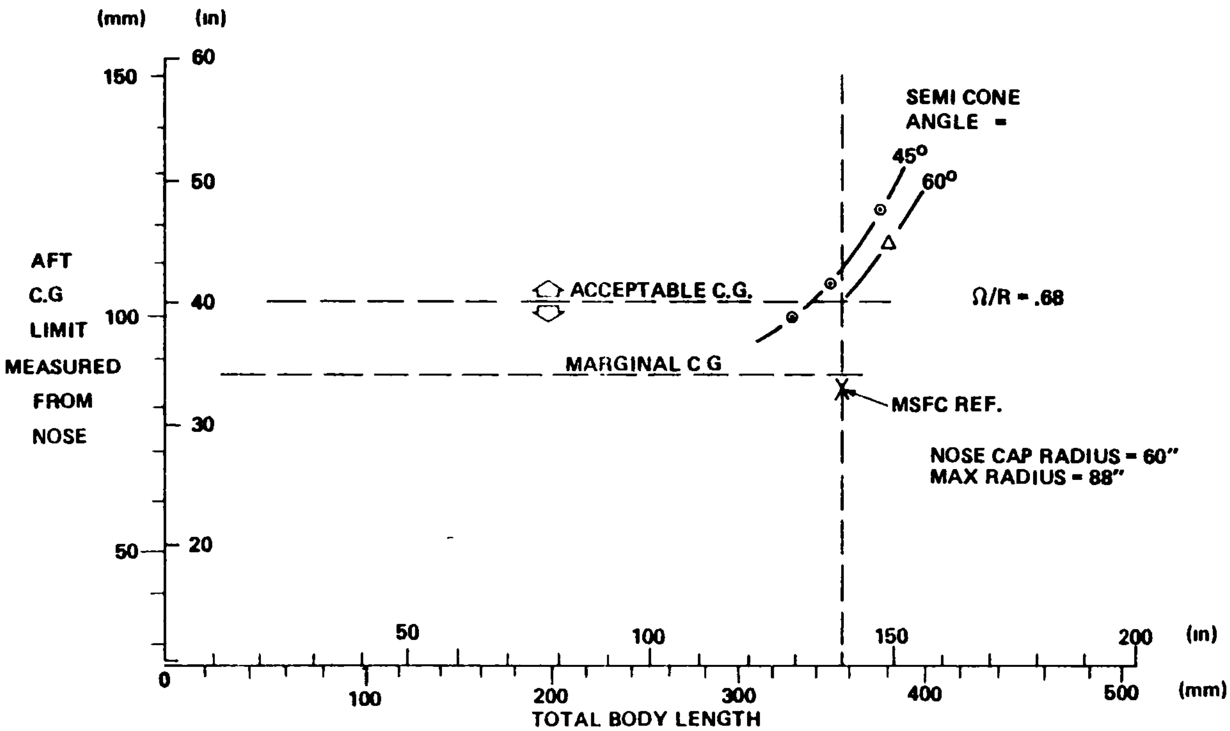


Figure 6.5-2. Hypersonic Aft C.G. Limit/Body Length Trade

millimeters from the extreme nose of the entry body as a function of total body length. With total body length limited to the Marshall reference system value, the 45 deg half angle blunt cone shows a clear superiority in the distance aft of the nose for the CG when compared with the 60 deg half angle cone.
As a result of the equivalence in terminal velocity, the minor differences in ablator mass, and the superior stability margin, the 45 deg half angle blunt cone was selected as a reference for further evaluation.

Results of the evaluation conducted to verify the minimum aft body angle for wake clearance are shown in Figure 6.5-3 which plots the hypersonic pitching moment coefficient against the angle of attack for selected aft body angles. The 30 deg aft body angle established for reentry wake clearance shows a positive stability margin for angles of attack ranging from 0 to 180 deg, allowing the option of some aft movement of the system CG if required.

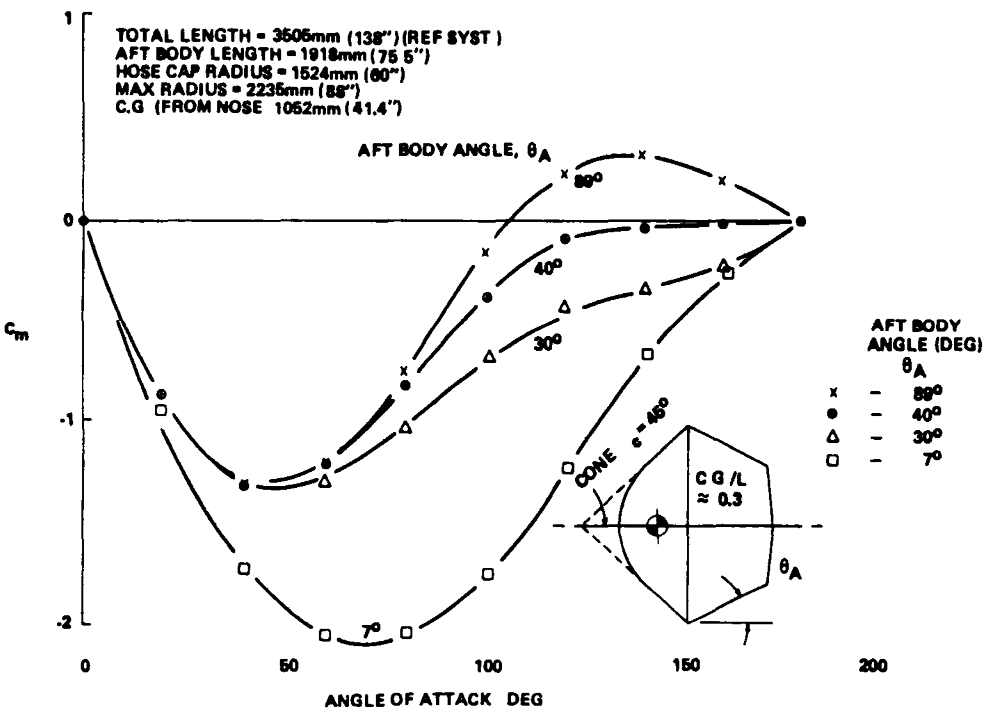


Figure 6.5-3: Hypersonic Pitching Moment/Aft Body Angle Trade

The configuration of the resulting waste payload protection system is illustrated in Figure 6.5-4 as it would be installed in the orbiter cargo bay. This shape offers advantages over the reference concept in the areas of internal volume for accommodation of subsystems and in terms of the data base available for reentry analysis; further development is strongly recommended.

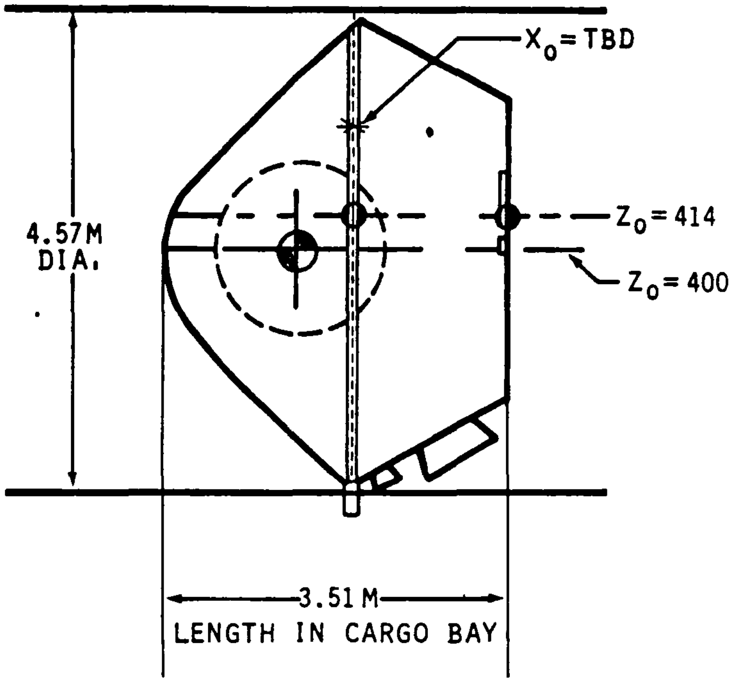


Figure 6.5-4: Waste Payload Protection System Configuration and Envelope Dimensions

6.5.2 Waste Payload Protective System Weight Estimating Relationships. A review of past studies of space disposal, in particular Reference 19, and of the space disposal reference system has indicated the complexity of the waste payload protection system. An indication of this complexity is indicated by the list of key elements illustrated in Figure 6.5-5. A preliminary investigation in several of these areas indicated no reason to differentiate from the concepts shown in the Marshall Space Flight Center Study. In accordance with the ground rules specifying maximum use of past studies, the MSFC generated data for the waste payload protective system from Reference 19 was used to estimate parametric weight relationships for the waste payload protective system.

Results are shown in Figure 6.5-6 which plots the total mass of the waste payload protective system against the mass of the waste payload carried from ground to low Earth orbit. Two curves are shown, the bottom one being the mass of the waste payload protective system by itself and the top curve being the mass of the waste payload protective system plus its associated flight support system. The mass of the reference system when plotted falls exactly on the bottom line. These curves were used for characterization of waste payload protective system mass in the total system studies in Task 8.

SYSTEM ELEMENTS (SUPPORT)

- EJECTION SYSTEM BLAST SHIELD
- SEPERATION SYSTEM
- THERMAL CONTROL
- SHIELD RECOVERY (REMOVEABLE SHIELD ONLY)
- INTERFACE ELECTRONICS
- WASTE PAYLOAD TRANSFER

SYSTEM ELEMENTS (WPPS)

- STRUCTURE
- EJECTION
- RE-ENTRY THERMAL PROTECTION
- RETARDATION
- SEPERATION
- WASTE PAYLOAD TRANSFER
- FLOTATION
- RECOVERY AID AND BEACON
- POWER + DISTRIBUTION
- AVIONICS
- CONTINGENCY IMPACT ATTENUATION
- FLUID AND PNEUMATIC

Figure 6.5-5: Waste Payload Protection System Elements

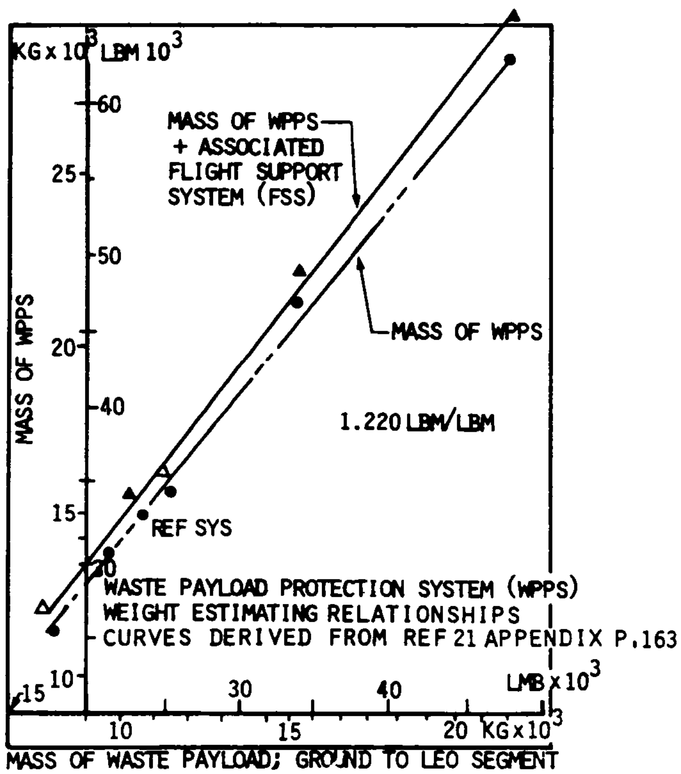


Figure 6.5-6: Waste Payload Protection System Parametric Mass Characterization

7.0 FLIGHT SUPPORT SYSTEMS

7.1 INTRODUCTION

Primary objectives of the flight support system task were to identify system requirements for support of waste payloads in the launch vehicle cargo bay; to define and evaluate flight support systems concepts; and to characterize the resulting flight support systems. Because of the earlier work documented in Reference 19 on flight support systems applicable to waste payloads protected by a waste payload protection system during launch, the effort in this study was concentrated on defining flight support systems for waste payload systems not requiring an ejectable waste payload protection system.

The task began with a review of flight support system requirements from past studies, aimed at identifing the peculiar requirements imposed by alternate waste payload concepts and launch vehicles. Consideration of these factors lead to identification of concepts for and characterization of flight support systems for (1) the hardened waste payload not using ejection, (2) the dual payload cargo manifest required by dual launch options, and (3) the area of payload support and contingency payload transfer for orbit transfer systems. The resulting concepts were characterized by drawings and preliminary mass statements for use in the total system evaluations conducted in Task 8.

7.2 FLIGHT SUPPORT SYSTEM FUNCTIONAL REQUIREMENTS

Primary functional requirements are summarized in Figure 7.2-1. With the exception of support of waste payload protection system ejection functions and support of radiation shield recovery after remote shield removal, these requirements are shared by the full range of flight support system concepts. Particular emphasis in this study was placed on mediation of payload transfer functions, which were accomplished on the reference system by the waste payload protection system, and on support of orbit transfer system docking with the orbiter required for the multiple launch scenarios considered in this study.

- PROVIDE MECHANICAL INTERFACE FOR WASTE PAYLOAD/WASTE PAYLOAD PROTECTIVE SYSTEM IN ORBITER CARGO BAY
- MEDIATE ALL PAYLOAD TRANSFER FUNCTIONS
- SUPPORT ORBIT TRANSFER SYSTEM DOCKING WITH ORBITER (MULTIPLE LAUNCH SCENARIOS)
- PROVIDE SUPPORT FOR CONTINGENCY OPERATIONS (TRANSFER AND RESCUE)
- PROVIDE COOLANT STORAGE FOR ONCE-THROUGH ACTIVE COOLING
- PROVIDE REMOTE DISCONNECT ELECTRICAL AND FLUID DISCONNECTS WITH PAYLOAD
- PROVIDE ELECTRICAL AND FLUID INTERFACES WITH ORBITER SYSTEMS
- PROVIDE FOR FLIGHT CREW MONITORING OF WASTE PAYLOAD STATUS (AFD)*
- PROVIDE FOR FLIGHT CREW CONTROL OF TRANSFER AND SHIELD (AFD) RECOVERY OPERATIONS
- SUPPORT WASTE PAYLOAD PROTECTION SYSTEM EJECTION FUNCTIONS
    - CONTINGENCY SEPARATION OF WPPS
    - BLAST SHIELD FOR ORBITER
- SUPPORT RADIATION SHIELD RECOVERY AFTER REMOTE SHIELD REMOVAL

*AFT FLIGHT DECK

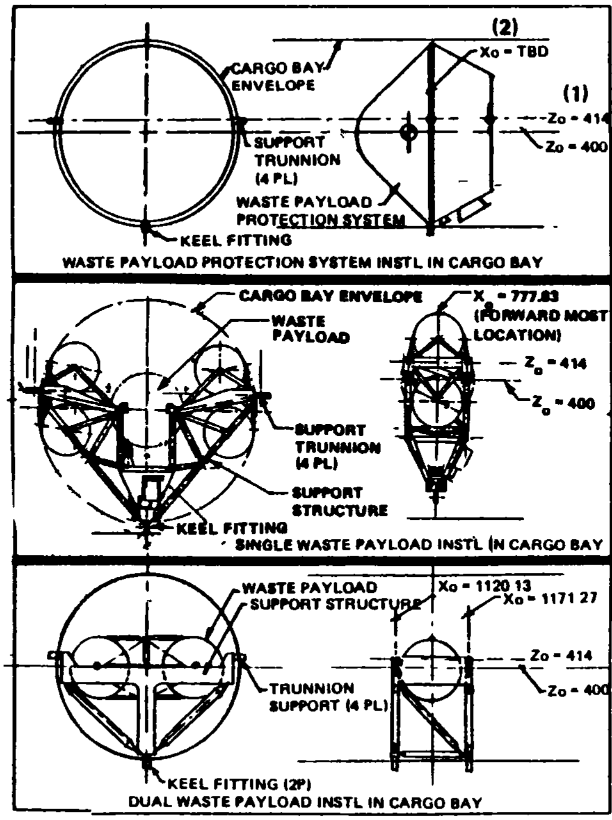
Figure 7.2-1: Flight Support System Functional Requirements

7.3 LAUNCH VEHICLE FLIGHT SUPPORT CONFIGURATIONS

Figure 7.3-1 illustrates the waste payload system options requiring flight support. The most common configuration, used on the largest number of options, uses the waste payload protection system to support the waste payload. This option has been studied in detail in previous studies (19) and these studies were used to provide the mass estimates used in Task 8. The length of this option is 3510 mm for all waste payload options considered.

The next most common configuration is the single waste payload supported in the cargo bay, used on total system options SL2, SL6 and SL10. The mass of this flight support system configuration ranged from about 2000 kg to about 4000 kg depending on the mass of the waste payload supported. Length in the shuttle bay ranged from 1.4m to 1.7m.

The last configuration studied was required to support a dual waste payload system mounted in the cargo bay. The mass of this system including coolant required for waste payload heat dissipation is 53,017 kg and it occupies a length of 2700 mm in the orbiter cargo bay. Orbiter cargo bay stations for the



CONFIGURATION	USED ON OPTIONS	MASS (KG)	LENGTH (M)
WASTE PAYLOAD SUPPORTED BY WASTE PAYLOAD PROTECTION SYSTEM (WPPS)	DL 1 (3) SL-1 SL-3 SL-4 SL-5 SL 7 SL-8 SL 9	CARRIED AS PART OF WPPS	3 51 (ALL OPTIONS)
SINGLE WASTE PAYLOAD IN CARGO BAY	SL 2 SL-6 SL-10	1927* 2707* 4386*	1 40 1 49 1 69
DUAL WASTE PAYLOAD IN CARGO BAY	DL-2	5317*	2 7M (APPROX)

(1) ORBITER WATERLINE (REF.) (2) ORBITER STATION (REF.) *INCLUDES COOLANT
(3) FOR OPTION DEFINITION SEE SECTION 5.7

Figure 7.3-1: Waste Payload System Options Requiring Flight Support

installation are tabulated for the single waste payload and dual waste payload system. Locations were determined as a function of the orbiter longeron fitting load capabilities.

7.3.1 Flight Support System for Single Waste Payload. Key features of the single waste payload flight support system are illustrated in Figure 7.3-2. The waste payload serves as an integral part of the structure. It is attached by four motor driven latches to a titanium space frame and two built-up frames which distribute the load from the waste payload to four longeron fittings and one keel fitting. The structure has been sized for orbiter flight and crash load as specified in the STS Payload Accommodation Documents (JSC 07700).

A built-up titanium sheet keel fitting reacts keel fitting kick loads and provides a mounting location for the STS interface avionics and the waste payload cooling and electrical umbilical. Four aluminum alloy coolant tanks carry sufficient water coolant for once through cooling on both nominal and contingency launch vehicle mission profiles.

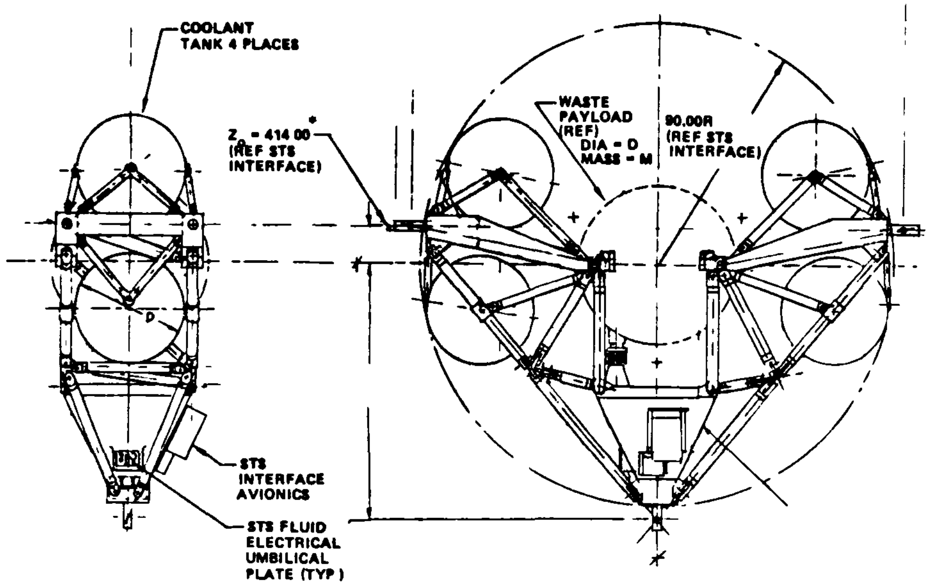


Figure 7.3-2: Flight Support System Configuration for Single Waste Payloads

Total mass including coolant ranges from 1927 kg to 4400 kg depending on the waste payload mass. A detailed group weight statement for the configuration is contained in Appendix I.

The single waste payload flight support system is designed to eliminate the requirement for separate waste payload transfer provisions. In operation the waste payload is attached to both the OTV waste payload support structure and the shuttle mounted flight support system. The three crosses flanking the waste payload in the illustration represent the position of the orbit transfer system mounted waste payload support system struts. Strut compliance is adjusted to assure primary load transfer through the flight support system rather than the orbit transfer vehicle. Prior to orbit transfer vehicle tilt up the four motor driven latches are opened allowing the orbit transfer system waste payload support system to lift the waste payload clear of the flight support system during OTV tilt up. For contingency recovery the procedure can be reversed.

7.3.2 <u>Flight Support System for Dual Waste Payload</u>. The dual waste payload system serves three primary functions in association with the interpayload support structure described in the preceeding section. It supports the dual waste payload in the STS cargo bay; incorporates an external docking ring which allows orbit transfer system docking prior to waste payload transfer from the orbiter to the orbit transfer system; and provides a tilt table and guide rails which interface with the waste payload guide rails to guide the waste payload during transfer to the orbit transfer system.

Key features are illustrated in Figure 7.3-3. Two built-up titanium tee frames braced by tubular titanium struts transfer loads from the dual waste payload to four longeron fittings and two keel fittings which interface with the space transportation system.

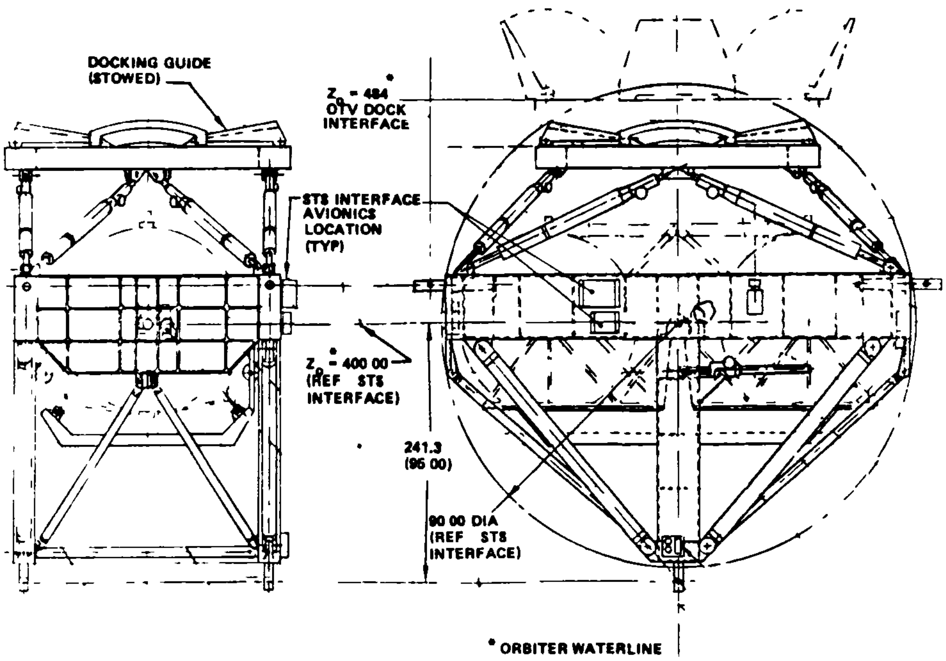


Figure 7.3-3: Flight Support System Configuration for Dual Waste Payloads

An extendable docking collar is stowed during launch and ascent and extended prior to orbit transfer system docking by support struts incorporating linear actuators. Support struts also incorporate impact attenuators to reduce the docking loads.

A tilt table driven by two linear actuators is incorporated to rotate the waste payload 90 deg prior to waste payload transfer. STS interface avionics and a TV camera to aid docking are mounted on the forward tee frame. Coolant is carried in tanks mounted beneath the cargo bay liner in the orbiter wing carry through structure.

Operation of the dual payload flight support system is illustrated schematically in Figure 7.3-4. In operation, the orbit transfer system docks to the

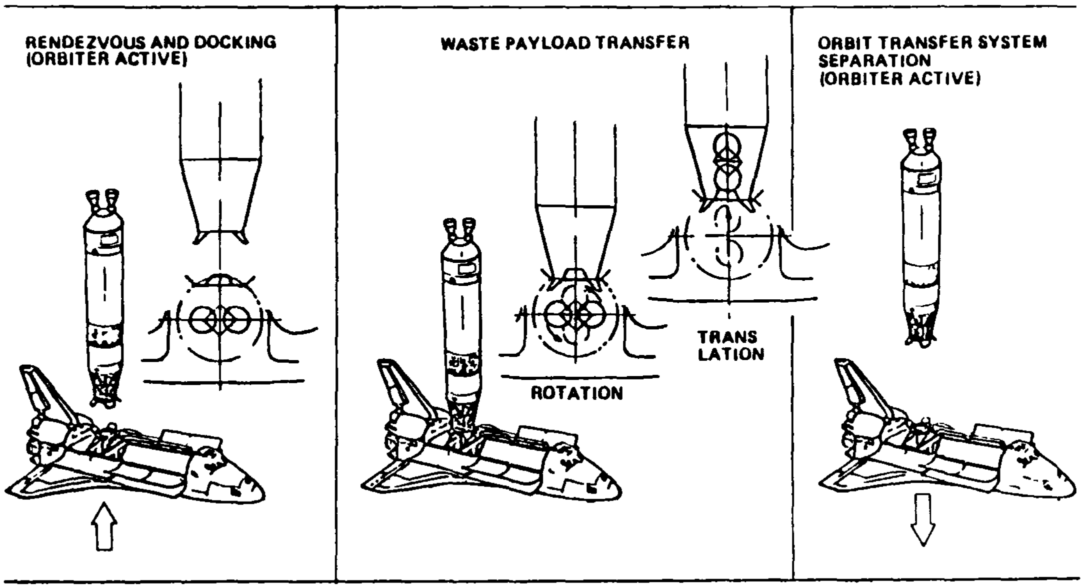


Figure 7.3-4: Dual Waste Payload Flight Support System Operation Sequence

extended docking collar. The STS orbiter performs the active role in this docking sequence. Waste payload transfer is accomplished by rotating the transfer cradle 90 deg using the linear actuators, allowing the waste payload to be translated through the center of the joined ring shaped docking collars to its final location in the orbit transfer system waste payload support structure. The orbiter then undocks and backs off and the orbit transfer system is powered up to initiate the transfer to the destination.

8.0 SYSTEMS INTEGRATION AND EVALUATION

8.1 INTRODUCTION

Primary objectives of the systems integration and evaluation task were to integrate the systems resulting from Tasks 2, 3, 5 and 6 into alternative total system concepts, to define these candidate concepts in terms of performance and risk, and to trade the total systems in the areas of risk and performance to select alternative systems of high merit for further definition.

Primary issues in systems integration included technical feasibility, total system performance and relative risk levels of different alternatives concepts. Technical feasibility and long term risk issues were settled in previous tasks (2, 3, 5, 6, 7) and were not treated in Task 8. Total system performance was defined as 1980 dollars per kilogram of waste form delivered to destination and was calculated for each alternative concept. Risk was evaluated on a relative basis only, with the risk of each alternative system compared on a qualitative basis with the risk level of the reference system.

Primary evaluation criteria for selection of systems with high merit were (1) relative risk to be less than the Marshall Space Flight Center reference concept and (2) relative cost less than or equal to the Marshall Space Flight Center reference concept.

The results of these studies were the selection of four systems as alternative systems possessing high merit. Two single launch solar electric options utilize high specific impulse to deliver 5405 kg of waste payload fully shielded to the 0.85 AU destination (Options SL-9, SL-10).

Two dual launch concepts use efficiencies of scale in launch vehicles and the increased specific impulse offered by long life cryogenic placement stages to delivery 10,150 kg of waste form fully shielded to the 0.85 AU destination.

No concept significantly less costly than the reference concept has been identified. All of the results obtained in this study indicate that increased performance of space systems translates to risk reductions far more readily than cost reduction for the space disposal mission.

8.2 SYSTEM INTEGRATION

The systems integration effort was accomplished in two steps. In the first, past studies were surveyed to ensure identification of all space system concepts. In the second phase, the "winners" of trades on launch vehicles and orbit transfer systems were arrayed against shield removal options and waste payload protection options identified in Task 2 to define a matrix showing reasonable alternative space system options for further analysis.

8.2.1 Identification of Options. Option identification began with a catalogue of space system concepts obtained from a survey of 57 references on space disposal of nuclear waste. Thirty-six distinct space system concepts were identified. From the initial screening, concepts using electromagnetic mass drivers and particle accelerators for disposal of waste products were rejected as out of the scope of the study. The remaining systems were characterized in the areas of (1) waste mixes, (2) waste payload system configuration, (3) space disposal destination, (4) launch vehicle usage in terms of the numbers of launch vehicles per mission and how they were manifested, (5) payload protection removal location, (6) number and reusability options used for upper stages, and (7) quantity of waste payload emplaced per mission. Results of this characterization are tabulated for each study investigated in Appendix C.

A second level of screening was then applied using the criteria of availability, risk and cost. System availability considerations led to rejection of systems using expendable launch systems due to be phased out before the 1995 IOC used in the study. System availability also caused rejection of SSTO concepts on the basis of initial operational capability past the period of interest and on insufficient configuration data to provide reliable costing. Systems evidencing risks greater than the qualitative risk level of the Marshal reference system were rejected. The primary rescue criterion evaluated was the availability of a rescue option. Systems with costs clearly greater than the cost of the Marshall reference were also rejected using vehicle reusability and the payload transported to destination for launch as the primary criterion.

Characteristics of the remaining systems were arranged in matrix form to define the entire range of reasonable total system options. The resulting matrix is illustrated in Figure 8.2-1. Primary option areas are identical with study tasks treating orbit transfer systems in Section 4, launch vehicles in Section 5, and payload protection in Section 6. These options represented the starting point for studies in the launch vehicle, orbit transfer system, and waste payload protection areas.

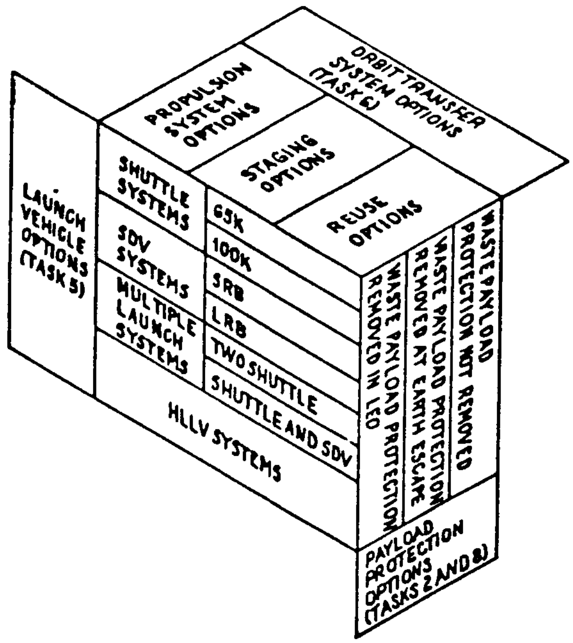


Figure 8.2-1: Identification of Options

8.2.2 Selection of Options for Evaluation. Results of studies in the areas noted were used to reduce the comprehensive matrix shown to a specific definition of options for evaluation. Specific options included were (1) the two launch vehicle options identified in Section 5, (2) orbit transfer system candidates defined for dual and single launch scenarios in Section 4, and (3) waste payload protection options including shield removal, shield removal location, and incorporation of an ejectable waste payload protection system defined in Section 6. Combining these options, using the appropriate constraints, allowed preparation of the matrix illustrated in Figure 8.2-2. Options are designated

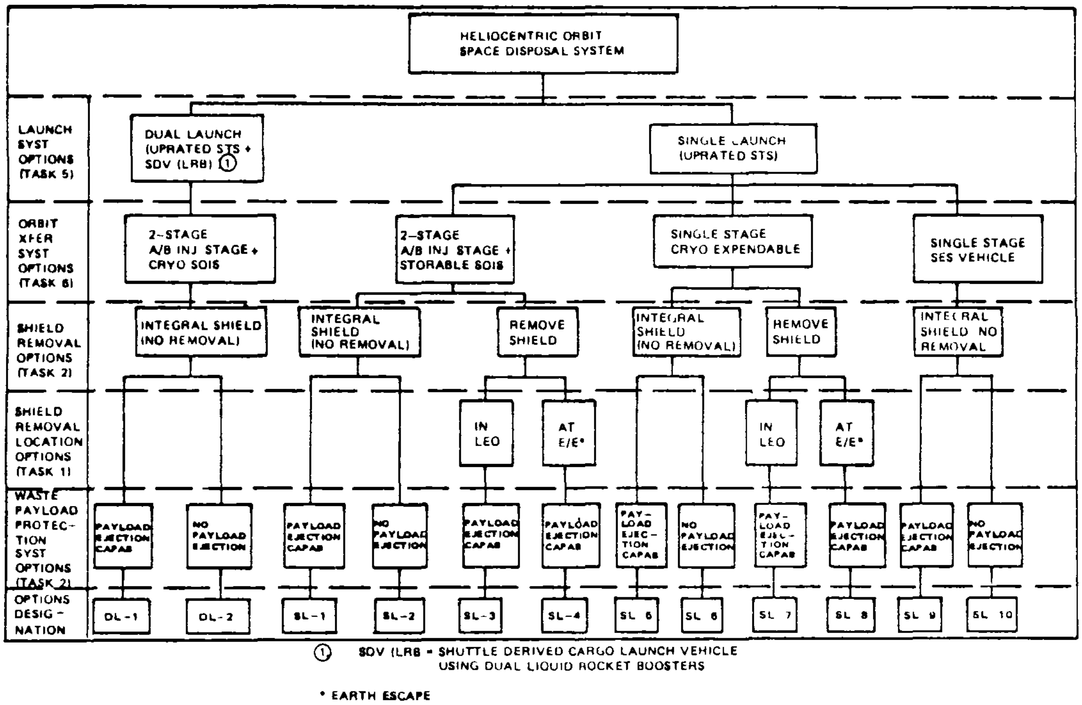


Figure 8.2-2: Selection of Options for Evaluation

by number on the bottom row. The description of each option can be obtained by following one branch of the trade tree reading from the top down. As an example, Option DL-1 is a dual launch option using a two stage orbit transfer system composed of an aerobraked reusable injection stage and a cryogenic SOIS. This option has an integral shield waste payload system and makes use of an ejectable waste payload protection system.

These options represent the complete range of reasonable alternative systems for space disposal, resulting from combining the highest performance orbit transfer options, the most cost effective launch vehicle options, and all viable waste payload protection options.

8.3 SYSTEM PERFORMANCE EVALUATION

8.3.1 Performance Evaluation Criteria.

The alternative space systems defined in Section 8.2 were evaluated for performance in terms of payload delivered per mission and cost. Performance was determined from the performance parametrics defined in Tasks 4, 5 and 6 and BAC performance and mass estimating codes. Cost was calculated on the basis of average cost per flight based on launch vehicle production and operations cost from past studies, orbit transfer system operations based on estimates obtained in the Phase A OTV study, and production cost for orbit transfer systems based on Phase A derived unit cost for orbit transfer system hardware. The output of the system performance evaluation was a figure of merit expressed in recurring dollars per kilogram of waste form delivered to destination.

8.3.2 Option Throw Weight Evaluation. For each option, the flight support system mass (from Task 3) and the waste payload parametrics (from Task 2), including the waste payload protective system, where applicable, were combined with the orbit transfer system airborne support equipment mass (from Task 6) and subtracted from the launch vehicle throw weight (from Task 5) to determine the initial orbit transfer system mass delivered to low orbit. This mass which includes interstages, orbit transfer vehicle flight support equipment, and avionics radiation shielding, if required, was then iterated using the orbit transfer system performance parametrics from Task 6 to determine the waste form mass delivered to the destination per flight.

8.3.3 Option Cost Evaluation. Costs for each option were estimated using FY 80 dollars on the basis of recurring cost per mission. A preliminary survey indicated that DDT&E costs were not a significant differentiator. Launch vehicle operations and procurement costs were obtained from Task 5; procurement costs were amortized over 100 missions. Development cost was not counted. Orbit transfer system average costs were calculated from data obtained during studies of the Phase A OTV and the solar electric propulsion system. OTV total first unit costs were estimated from the Phase A study results. Solar electric stage costs were scaled to the increased mass of the 270 kW solar electric stage. Operations costs for both ground and flight control operations were obtained from Phase A OTV study estimates. The cost per flight estimates generated exclude costs associated with the waste payload protective system, ground support equipment, facilities, program operation costs, and contingency.

8.3.4 Option Performance Evaluations. This section presents a summary performance assessment for each of the 12 options identified. Each assessment consists of a space system summary mass statement and a space system summary cost estimate. The waste form delivered per mission in kilograms, cost per mission in 1980 dollars, and the unit cost in 1980 dollars per kilogram for deployment are also tabulated. Mass values shown are estimated using the parametric weight estimating relationships derived in Tasks 4 and 6 and the airborne support equipment and waste payload flight support system masses generated in Task 3.

Production costs for orbit transfer systems were calculated by using the theoretical first unit costs and the appropriate amortization for the number of flights and production runs required. Launch vehicle costs were amortized over 100 flights.

The following sections present details of space system performance assessment and a summary description of each option.

Option DL-1. This is a dual launch option using the uprated space transportation system teamed with a liquid rocket booster version of the shuttle derived cargo launch vehicle. A two stage orbit transfer system uses an aerobraked injection stage and a cryogenic propellant solar orbit insertion stage. The waste payload is protected by a nonremovable integral shield and an ejectable waste payload projection system. The performance assessment for option DL-1 is presented in Figure 8.3-1. This option delivers two 4250 kg waste forms per mission for a total waste form delivered per mission of 8500 kg. Cost per mission is $85.25 million with operations, primarily launch costs, accounting for $51.65 million and production costs accounting for $33.6 million, dominated by the $23 million devoted to amortization of launch vehicles. This option delivers the waste mix to the destination for approximately $10,000 per kilogram. It shares with option SL-9 the distinction of being the lowest risk option considered and is the second lowest in unit delivery cost.

Option DL-2. Option DL-2 is similar to DL-1 except that no ejectable waste payload protection system is incorporated, with the hardened integral shield providing primary protection for the waste payload. The resulting decrease in tare weight allows the waste form delivered per mission to be increased to 10,150 kg, resulting in a decrease in unit cost from $10,000 to about $8,000 per kilogram. The performance assessment of option DL-2 is illustrated in Figure 8.3-2.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)	
TOTAL LAUNCH MASS	45,361 (LV-1)
WASTE PAYLOAD SYSTEM	43,628
WASTE PAYLOAD	2 @ 15,559
WASTE FORM	2 @ 4,250
SHIELDING	2 @ 11,309
WPPS	2 @ 6,255
TOTAL LAUNCH MASS	83,915 (LV-2)
ORBIT TRANSFER SYSTEM	81,647
INJECTION STAGE	67,177
PLACEMENT STAGE	13,762
PAYLOAD ADAPTER	454
INTERSTAGE	254
AIRBORNE SUPPORT EQUIP.	
STAGE ASE (1)	2,268
WASTE PAYLOAD (2)	1,733
FLIGHT SUPPORT	

SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL COST (PER MISSION)	85.24
OPERATIONS	51.65
LAUNCH OPS	46.20
LV-1	27.50
LV-2	18.70
OTS OPS	5.45
FLT OPS	1.25
GROUND OPS	4.20
PRODUCTION	33.59
INJ. STAGE	1.08
PLACEMENT STAGE	6.43
WASTE PAYLOAD SYST.	1.00
FLIGHT SUPPORT	1.78
LV-1	15.10
LV-2	8.20

(2) LV-1 ONLY
(1) LV-2 ONLY

WASTE FORM DELIVERED	8,500 KG
COST PER MISSION	85.24 M
UNIT COST. $/KG	10,028

Figure 8.3-1 Space System Performance Assessment: Option DL-1

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)	
TOTAL LAUNCH MASS	39,101 (LV-1)
WASTE PAYLOAD SYSTEM	33,784
WASTE PAYLOAD	2 @ 16,892
WASTE FORM	2 @ 5,075
SHIELDING	2 @ 11,817
TOTAL LAUNCH MASS	83,915 (LV-2)
ORBIT TRANSFER SYSTEM	81,647
INJECTION STAGE	67,177
PLACEMENT STAGE	13,762
PAYLOAD ADAPTER	454
INTERSTAGE	254
AIRBORNE SUPPORT EQUIP.	
STAGE ASE (1)	2,268
WASTE PAYLOAD (2)	5,317
FLIGHT SUPPORT	

SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL COST (PER MISSION)	85.24
OPERATIONS	51.65
LAUNCH OPS	46.20
LV-1	27.50
LV-2	18.70
OTS OPS	5.45
FLT OPS	1.25
GROUND OPS	4.20
PRODUCTION	33.59
INJ. STAGE	1.08
PLACEMENT STAGE	6.43
WASTE PAYLOAD SYST.	1.00
FLIGHT SUPPORT	1.78
LV-1	15.10
LV-2	8.20

(1) LV-2 ONLY
(2) LV-1 ONLY

WASTE FORM DELIVERED	10,150 KG
COST PER MISSION	85.24 M
UNIT COST. $/KG	8,398

Figure 8.3-2· Space System Performance Assessment. Option DL-2

Option SL-1. This is the first of the single launch options. It uses a two stage orbit transfer system consisting of an aerobraked injection stage and a storable propellant SOIS. The waste payload is protected by an integral shield and this option incorporates an ejectable waste payload protection system. The summary performance assessment for option SL-1 is illustrated in Figure 8.3-3. This system is capable of delivering approximately 1600 kg of waste form per launch at a cost per mission of approximately $54 million. This is divided into a $34 million operations cost dominated by launch operations and approximately $20 million in production costs dominated by the amortization cost of the launch vehicle. This option is the lowest performance option evaluated with a unit cost for deployment of $33,000 per kilogram.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	45,354	TOTAL COST (PER MISSION)	53.77
WASTE PAYLOAD SYSTEM	11,097	OPERATIONS	33.95
WASTE PAYLOAD	7510	LAUNCH OPS	28.00
WASTE FORM	1,599	LV-1	28.00
SHIELDING	5,911	LV-2	
WPPS	3,587	OTS OPS	5.95
ORBIT TRANSFER SYSTEM	29,722	FLT OPS	1.25
INJECTION STAGE	22,634	GROUND OPS	4.70
PLACEMENT STAGE	6,862	PRODUCTION	19.82
PAYLOAD ADAPTER	226	INJ. STAGE	0.61
INTERSTAGE		PLACEMENT STAGE	3.79
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.61
STAGE ASE	2,494	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	14.29
FLIGHT SUPPORT		LV-2	--

WASTE FORM DELIVERED	1,599 KG
COST PER MISSION	53.77 M
UNIT COST: $/KG	33,627

Figure 8.3-3: Space System Performance Assessment: Option SL-1

Option SL-2. This option is similar to SL-1 except that the ejectable waste payload protection system is not used. This allows an increase in the waste form delivered per mission to 1800 kg, decreasing the unit cost in dollars per kilogram for delivery to $30,133. The performance assessment for SL-2 is illustrated in Figure 8.3-4.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	45,176	TOTAL COST (PER MISSION)	54.24
WASTE PAYLOAD SYSTEM	8,391	OPERATIONS	33.85
WASTE PAYLOAD	8,391	LAUNCH OPS	28.00
WASTE FORM	1,800	LV-1	28.00
SHIELDING	6,591	LV-2	
		OTS OPS	5.85
ORBIT TRANSFER SYSTEM	32,250	FLT OPS	1.25
INJECTION STAGE	24,812	GROUND OPS	4.60
PLACEMENT STAGE	7,212	PRODUCTION	20.39
PAYLOAD ADAPTER	226	INJ. STAGE	0.62
INTERSTAGE		PLACEMENT STAGE	3.56
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.50
STAGE ASE	2,494	FLIGHT SUPPORT	0.61
WASTE PAYLOAD	2,041	LV-1	15.10
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	1,800 KG
COST PER MISSION	54.24 M
UNIT COST: $/KG	30,133

Figure 8.3-4: Space System Performance Assessment: Option SL-2

Option SL-3. This option shares the two stage orbit transfer system used for Option SL-1 and SL-2 but removes the radiation shield in low Earth orbit prior to deployment of the waste payload from low orbit to the destination. Protection for the waste payload is provided by an ejectable waste payload protection system. This option is similar to the MSFC reference system except for the use of an aerobraked return for the injection stage and use of the steel/graphite composite radiation shield. A summary performance assessment for Option SL-3 is presented in Figure 8.3-5. Removal of the shield in low orbit increases the waste payload delivered per mission to over 4000 kg at approximately the same cost per flight. This allows a reduction in the unit cost for deployment to $13,400 per kilogram making this the fourth highest performance option evaluated. This unit cost was used as a standard for comparison of other concepts.

Option SL-4. This option is similar to option SL-3 except that the radiation shield used for protection is removed from the waste payload at Earth escape rather than at low Earth orbit. This results in a decrease in the waste payload deployed per mission from 4000 kg to approximately 2000 kg, just about doubling the unit cost for deployment, which rises to $23,400 per kilogram.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	47,625	TOTAL COST (PER MISSION)	54.60M
WASTE PAYLOAD SYSTEM	20,729	OPERATIONS	33.85
WASTE PAYLOAD	14,700	LAUNCH OPS	28.0
WASTE FORM	4,082	LV-1	28.0
SHIELDING	10,618	LV-2	
WPPS	6,029	OTS OPS	5.85
ORBIT TRANSFER SYSTEM	22,361	FLT OPS	1.25
INJECTION STAGE	17,826	GROUND OPS	4.60
PLACEMENT STAGE	4,309	PRODUCTION	20.75
PAYLOAD ADAPTER	226	INJ. STAGE	0.68
INTERSTAGE		PLACEMENT STAGE	4.32
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.13
STAGE ASE	2,494	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	15.10
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	4,082
COST PER MISSION	54.7 M
UNIT COST: $/KG	13,376

Figure 8.3-5: Space System Performance Assessment: Option SL-3

'he summary performance assessment for option SL-4 is presented in Figure 8.3-6.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	45,357	TOTAL COST (PER MISSION)	54.14
WASTE PAYLOAD SYSTEM	14,066	OPERATIONS	33.95
WASTE PAYLOAD	9.943	LAUNCH OPS	28.0
WASTE FORM	1,978	LV-1	28.0
SHIELDING	7,965	LV-2	
WPPS	4,123	OTS OPS	5.95
ORBIT TRANSFER SYSTEM	26,756	FLT OPS	1.25
INJECTION STAGE	22,557	GROUND OPS	4.70
PLACEMENT STAGE	3,973	PRODUCTION	20.19
PAYLOAD ADAPTER	226	INJ. STAGE	0.65
INTERSTAGE		PLACEMENT STAGE	3.60
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.70
STAGE ASE	2,494	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	14.72
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	1,978 KG
COST PER MISSION	54.14M
UNIT COST: $/KG	27,371

Figure 8.3-6. Space System Performance Assessment: Option SL-4

Option SL-5. This option takes advantage of the most effective orbit transfer system identified for single launch options. The orbit transfer system is a single stage cryogenic vehicle which is used in an expendable mode. It carries the waste payload all the way to the destination orbit at 0.85 AU.

Option SL-5 uses an integral shield for waste payload protection and also incorporates an ejectable waste payload projection system. Figure 8.3-7 presents the summary performance assessment for option SL-5. The increased performance

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)	
TOTAL LAUNCH MASS	47,818
WASTE PAYLOAD SYSTEM	13,060
WASTE PAYLOAD	9,118
WASTE FORM	2,100
SHIELDING	7,018
WPPS	3,942
ORBIT TRANSFER SYSTEM	27,764
INJECTION STAGE	27,538
PLACEMENT STAGE	
PAYLOAD ADAPTER	226
INTERSTAGE	
AIRBORNE SUPPORT EQUIP.	6,994
STAGE ASE	2,494
WASTE PAYLOAD	4,500
FLIGHT SUPPORT	

SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL COST (PER MISSION)	57.81
OPERATIONS	29.96
LAUNCH OPS	27.5
LV-1	27.5
LV-2	
OTS OPS	2.46
FLT OPS	1.25
GROUND OPS	1.21
PRODUCTION	27.85
INJ. STAGE	12.21
PLACEMENT STAGE	0.60
WASTE PAYLOAD SYST.	
FLIGHT SUPPORT	0.52
LV-1	14.52
LV-2	

WASTE FORM DELIVERED	2,100 KG
COST PER MISSION	57.91 M
UNIT COST: $/KG	27,529

Figure 8.3-7: Space System Performance Assessment: Option SL-5

of the cryogenic expendable stage increases waste form delivered to 2100 kg compared to 1600 kg for the most closely equivalent two stage single launch system. The cost per mission is only slightly increased at $58 million with the increased cost of the LLOTV stage at $12.2 million partially compensated for by the deletion of the orbit transfer vehicle ground refurbishment cost. Unit cost for this option is $27,580 per kilogram.

Option SL-6. This is the most efficient single launch option which preserves the low risk feature of carrying the shield all the way to the destination. Increased payload is provided by the deletion of the ejectable waste payload protection system with the hardened integral shield now providing primary waste payload protection. The resulting performance increase is shown in Figure 8.3-8. Waste form delivered per mission is increased to 2500 kg and the cost per mission remains approximately the same, decreasing the unit cost to approximately $22,700 per kilogram.

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)	
TOTAL LAUNCH MASS	45,353
WASTE PAYLOAD SYSTEM	10,147
WASTE PAYLOAD	10,147
WASTE FORM	2,500
SHIELDING	7,647
ORBIT TRANSFER SYSTEM	30,671
INJECTION STAGE	30,445
PLACEMENT STAGE	
PAYLOAD ADAPTER	226
INTERSTAGE	
AIRBORNE SUPPORT EQUIP.	4,535
STAGE ASE	2,494
WASTE PAYLOAD	2,041
FLIGHT SUPPORT	

SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL COST (PER MISSION)	56.73
OPERATIONS	29.96
LAUNCH OPS	27.5
LV-1	27.5
LV-2	
OTS OPS	2.46
FLT OPS	1.25
GROUND OPS	1.21
PRODUCTION	26.77
INJ. STAGE	11.06
PLACEMENT STAGE	
WASTE PAYLOAD SYST.	0.50
FLIGHT SUPPORT	0.61
LV-1	14.60
LV-2	

WASTE FORM DELIVERED	2,500 KG
COST PER MISSION	56.73 M
UNIT COST: $/KG	22,692

Figure 8.3-8: Space System Performance Assessment: Option SL-6

Option SL-7. This option combines the cryogenic expendable orbit transfer system with shield removal at low Earth orbit to achieve the best performance of any single launch option. The waste payload shield is removed in low Earth orbit. Protection during ascent is provided by an ejectable waste payload protection system. The performance assessment illustrated in Figure 8.3-9 shows the efficiency of this concept which delivers almost 5000 kg of waste form per mission at a cost of just over $58 million. Unit cost for this option is approximately $11,600 per kilogram delivered.

Option SL-8. This option is similar to SL-7 but the shield is removed at Earth escape instead of in low Earth orbit. The severe penalty this exacts is illustrated in Figure 8.3-10 which shows the waste form delivered per mission reduced to 2100 kg with the cost per mission essentially unchanged at $58 million. This more than doubles the unit cost per deployment to $27,600 per

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	46,946	TOTAL COST (PER MISSION)	58.24
WASTE PAYLOAD SYSTEM	23,133	OPERATIONS	30.06
WASTE PAYLOAD	17,500	LAUNCH OPS	27.50
WASTE FORM	4,990	LV-1	27.50
SHIELDING	12,510	LV-2	
WPPS	5,633	OTS OPS	2.56
ORBIT TRANSFER SYSTEM	19,278	FLT OPS	1.25
INJECTION STAGE	19,052	GROUND OPS	1.31
PLACEMENT STAGE		PRODUCTION	28.18
PAYLOAD ADAPTER	226	INJ. STAGE	13.40
INTERSTAGE		PLACEMENT STAGE	
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.13
STAGE ASE	2,494	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	14.13
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	4,990 KG
COST PER MISSION	58.24 M
UNIT COST: $/KG	11,671

Figure 8.3-9: Space System Performance Assessment: Option SL-7

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	45,358	TOTAL COST (PER MISSION)	58.01
WASTE PAYLOAD SYSTEM	14,447	OPERATIONS	30.06
WASTE PAYLOAD	10,256	LAUNCH OPS	27.50
WASTE FORM	2,105	LV-1	27.50
SHIELDING	8,151	LV-2	
WPPS	4,191	OTS OPS	2.56
ORBIT TRANSFER SYSTEM	26,376	FLT OPS	1.25
INJECTION STAGE	26,150	GROUND OPS	1.31
PLACEMENT STAGE		PRODUCTION	27.95
PAYLOAD ADAPTER	226	INJ. STAGE	12.21
INTERSTAGE		PLACEMENT STAGE	
AIRBORNE SUPPORT EQUIP.	4,535	WASTE PAYLOAD SYST.	0.70
STAGE ASE	2,494	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	14.52
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	2,105 KG
COST PER MISSION	58.01 M
UNIT COST. $/KG	27,558

Figure 8.3-10: Space System Performance Assessment: Option SL-8

kilogram, illustrating the significant penalty of carrying the massive radiation shield through the delta-V required to reach Earth escape.

Option SL-9. This option is the first considered that uses an electric propulsion orbit transfer system. A waste payload protection system is also incorporated to provide a second level of protection beyond the integral shield. Figure 8.3-11 illustrates the performance assessment for this option and points

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)		SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL LAUNCH MASS	45,132	TOTAL COST (PER MISSION)	78.16 M
WASTE PAYLOAD SYSTEM	24,267	OPERATIONS	32.14
WASTE PAYLOAD	17,500	LAUNCH OPS	27.5
WASTE FORM	5,449	LV-1	27.5
SHIELDING	12,051	LV-2	
WPPS	6,767	OTS OPS	4.64
ORBIT TRANSFER SYSTEM	16,556	FLT OPS	1.25
INJECTION STAGE	16,330	GROUND OPS	3.39
PLACEMENT STAGE		PRODUCTION	46.02
PAYLOAD ADAPTER	226	INJ. STAGE	30.4
INTERSTAGE		PLACEMENT STAGE	
AIRBORNE SUPPORT EQUIP.	4,309	WASTE PAYLOAD SYST.	0.50
STAGE ASE	2,268	FLIGHT SUPPORT	0.52
WASTE PAYLOAD	2,041	LV-1	14.60
FLIGHT SUPPORT		LV-2	

WASTE FORM DELIVERED	5,449 KG
COST PER MISSION	78.16 M
UNIT COST: $/KG	14,344

Figure 8.3-11 Space System Performance Assessment: Option SL-9

out some of the differences between the electric space systems and the chemical propellant space systems. Waste form delivered per mission is almost 5500 kg. This is the largest sized fully shielded waste form compatible with passive rejection of the thermal load. Cost per mission for the electric vehicle is approximately $78 million; for the first time, the production cost of $46 million outweighs the operations cost of $32 million.

Operation costs, as in the other options, are dominated by the space shuttle launch costs. In the production cost category, for the first time, the $30 million cost of the solar-electric propulsion injection stage outweighs the $14 million required for launch vehicle amortization. The high performance of the solar electric stage largely outweighs the relatively high cost per mission resulting in a unit cost for deployment of $14,300 per kilogram.

Option SL-10. This option is similar to SL-9 except for deletion of the ejectable waste payload protection system. Figure 8.3-12 shows the system summary performance assessment. Because the payload is limited by thermal dissipation considerations, the reduced mass cannot be readily translated to increase payloads and since the cost per mission remains the same, the unit cost is also

SPACE SYSTEM SUMMARY MASS STATEMENT (KG)	
TOTAL LAUNCH MASS	40,710
WASTE PAYLOAD SYSTEM	17,500
WASTE PAYLOAD	17,500
WASTE FORM	5,449
SHIELDING	12,051
ORBIT TRANSFER SYSTEM	16,556
INJECTION STAGE	16,330
PLACEMENT STAGE	
PAYLOAD ADAPTER	226
INTERSTAGE	
AIRBORNE SUPPORT EQUIP.	6,654
STAGE ASE	2,268
WASTE PAYLOAD	4,386
FLIGHT SUPPORT	

SPACE SYSTEM SUMMARY COST ESTIMATE (M $)	
TOTAL COST (PER MISSION)	78.16 M
OPERATIONS	32.14
LAUNCH OPS	27.50
LV-1	27.50
LV-2	
OTS OPS	4.64
FLT OPS	1.25
GROUND OPS	3.39
PRODUCTION	46.0
INJ. STAGE	30.4
PLACEMENT STAGE	
WASTE PAYLOAD SYST.	0.50
FLIGHT SUPPORT	0.52
LV-1	14.60
LV-2	

WASTE FORM DELIVERED	5,449 KG
COST PER MISSION	78.16 M
UNIT COST: $/KG	14,344

Figure 8.3-12: Space System Performance Assessment: Option SL-10

identical to that of SL-9 at $14,300 per kilogram. This points out the fact that the relative insensitivity of the electric vehicle options allows incorporation of an ejectable waste payload protection system at essentially no penalty in cost per flight.

Figure 8.3-13 presents a comparison of the figures of merit for 12 options considered. Costs are normalized to the cost of the reference system, identified as system SL-3. The dotted lines above and below the cost shown for the reference system represent cost increases and decreases of 50%, respectively. To allow for the uncertainty in cost and performance estimates, only

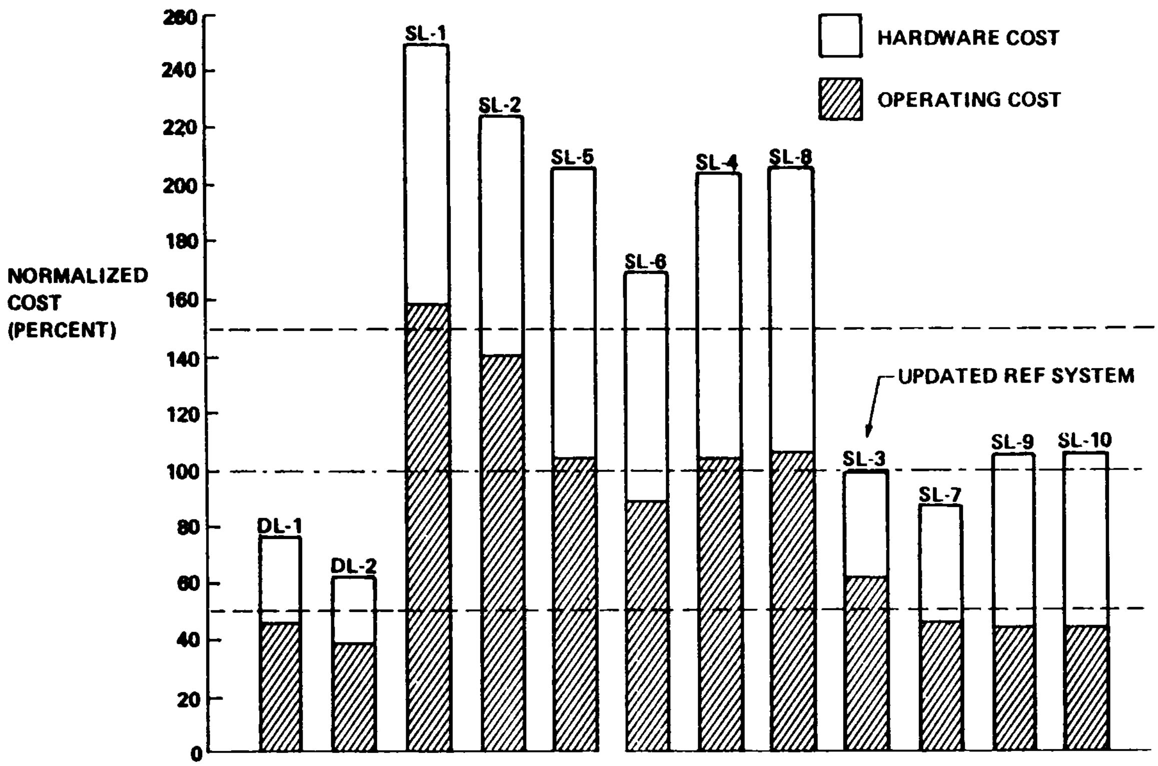


Figure 8.3-13: Comparison of Total System Performance

systems within the band were considered essentially equivalent in cost to each other and to the reference system. Systems falling outside the band were considered to be significanty more costly than the reference. It is apparent that only five options fall within this category, not counting the updated reference system. The two dual-launched systems (DL-1, DL-2) which take advantage of economies of scale and the higher efficiency of the cryogenic placement stages are the lowest cost options; the two electric vehicles (SL-9, SL-10), which utilize the high specific impulse of electric propulsion can for all practical purposes be considered equivalent in cost to the updated reference system. Both of these systems carry the waste payload shielding all the way to the destination. This represents a significant decrease in risk compared to the reference system. The single-stage cryogenic propellant option (SL-7), unlike the previously mentioned options, removes all waste payload protection in low Earth orbit. This option is equivalent in risk to the reference concept.

8.4 SYSTEM RISK EVALUATION

Risk evaluations were conducted to rank the alternative systems considered with respect to the risk of the updated reference system (option SL-3).

8.4.1 Risk Evaluation Criteria. Relative risk for the alternative systems was evaluated by comparing the risk reduction provisions incorporated in each option. Risk reduction provisions considered included: (1) whether or not the payload had an integral hardened shield or whether the shield was removable, implying a reduction in hardness; (2) whether or not an ejectable waste payload protection system was incorporated for coping with launch vehicle failures; (3) whether the waste payload was shipped to orbit in a dry payload bay containing no propellants or whether it was packaged in a payload bay, also containing an orbit transfer vehicle fueled with substantial quantities of highly energetic hypergolic or cryogenic propellants; and (4) whether or not the payload was protected from contingency reentry. Reentry protection for the hardened waste payloads is provided by the 50 mm graphite shield which surrounds the steel primary shield and container. Payloads removing the radiation shield at Earth escape are protected from contingency reentry by fundamental laws of orbital mechanics. Payloads with radiation shields and protection removed in low orbit, however, remain vulnerable to liberation of substantial quantities of waste form during an unprotected contingency reentry (14).

8.4.2 Option Risk Evaluation. Figure 8.4-1 presents the qualitative risk ranking for the 12 options considered. Four differentiable risk levels were

			DEFINING CRITERIA (RISK AMELIORATION)			
	OPTION	RISK LEVEL	INTEGRAL SHIELD (HARD PAYLOAD)	WASTE PAYLOAD EJECTION CAPABILITY	"DRY" PAYLOAD BAY? (NO PROPELLANTS)	PROTECTED DURING CONTINGENCY RE-ENTRY
LOWEST RISK	DL-1, SL-9	1	YES	YES	YES	YES
	SL-1, SL-5	2	YES	YES	NO	YES
	DL-2, SL-10	2	YES	NO	YES	YES
	SL-2, SL-6	3	YES	NO	NO	YES
	SL-4, SL-8	3	NO	YES	NO	YES
HIGHEST RISK	SL-3, SL-7	4	NO	YES	NO	NO

Figure 8.4-1 Qualitative Risk Ranking

identified based on the number of risk reduction provisions incorporated in the option. Risk level one, shared by options DL-1 and SL-9, incorporates all provisions for risk reduction. Risk level two, shared by options SL-1, SL-5, DL-2, and SL-10, indicates an option incorporating three of the four risk reduction provisions. Options designated as risk level three incorporate two of the four risk reduction provisions, and options SL-3 and SL-7, at risk level four, represent the highest risk systems considered. All options except SL-7 are lower in qualitative risk than the updated reference system (SL-3).

8.5 SYSTEM SCREENING

A final screening was conducted to combine the performance and risk evaluations to select alternative concepts possessing high merit.

8.5.1 <u>Total System Evaluation Criteria</u>. Five criteria are identified for total system evaluation:

1) Technical feasibility
2) Risk
3) Cost
4) Environmental impact
5) Long term risk

Three of these criteria are not differentiators. Technical feasibility was established in Tasks 2, 5, and 6 for all of the options considered. Environmental impact is proportional to the number of launches and as such is measured by the system performance. Fewer launches equal less environmental impact. Long term risk was screened in Task 7; the destination selected, 0.85 AU heliocentric orbit, had the lowest long term risk of any destination identified.

The key differentiators are relative risk and performance. Risk criteria was evaluated by relative ranking. Systems having a risk equal to the reference system were rejected. Systems possessing a risk less than the reference system were judged acceptable for risk. Performance was also evaluated relative to the reference system. Systems whose costs per mission were more than 150% of the reference system cost per mission were rejected. Systems possessing costs per mission judged less than or equal to that of the reference system were accepted.

8.5.2 Space Systems Screening for Performance and Risk. Figure 8.5-1 presents the results of the screening for cost and risk conducted on the 12 candidate concepts. A 1 in the columns marked "risk score" and "cost score" constitutes

CANDIDATE SYSTEM DESIGNATION	SUMMARY DESCRIPTION	WASTE FORM MASS DELIVERED PER FLT(KG)	RISK SCORE	COST SCORE	DISPOSITION AND COMMENTS
DL-1	DUAL LAUNCH A/B INJ + CRYO SOIS, WPPS + INTEGRAL SHIELD	8500	1	1	ACCEPT TIES SL-9 FOR LOWEST RISK
DL-2	DUAL LAUNCH A/B INJ + CRYO SOIS, INTEGRAL SHIELD ONLY	10,150	1	1	ACCEPT HIGHEST PERFORMANCE SYSTEM
SL-1	SINGLE LAUNCH A/B INJ + STORABLE SOIS, WPPS + INTEGRAL SHIELD	1599	1	X	REJECT LOWEST PERFORMANCE SYSTEM
SL-2	SINGLE LAUNCH A/B INJ + STORABLE SOIS, INTEGRAL SHIELD ONLY	1800	1	X	REJECT PERFORMANCE < REF
SL-3	SINGLE LAUNCH A/B INJ + STORABLE SOIS, WPPS + REMOVABLE SHIELD (LEO)	4082	X	1	UPDATED REFERENCE SYSTEM
SL-4	SINGLE LAUNCH A/B INJ + STORABLE SOIS, WPPS + REMOVE SHIELD (E/E)	1977	1	X	REJECT PERFORMANCE < REF
SL-5	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV, WPPS + INTEGRAL SHIELD	2099	1	X	REJECT PERFORMANCE < REF
SL-6	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV, INTEGRAL SHIELD ONLY,	2500	1	X	REJECT PERFORMANCE < REF
SL-7	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV, WPPS + REMOVABLE SHIELD (LEO)	4989	X	1	REJECT HIGHEST PERFORMANCE SINGLE LAUNCH SYSTEM BUT RISK = REF SYSTEM
SL-8	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV, WPPS + REMOVABLE SHIELD (E/E)	2099	1	X	REJECT PERFORMANCE < REF
SL-9	SINGLE LAUNCH, SINGLE STAGE EXPENDABLE SES WPPS + INTEGRAL SHIELD	5449	1	1	ACCEPT TIES DL-1 FOR LOWEST RISK
SL-10	SINGLE LAUNCH, SINGLE STAGE EXPENDABLE SES INTEGRAL SHIELD	5449	1	1	ACCEPT PERFORMANCE + RISK ACCEPTABLE

Figure 8.5-1: Space System Screening

acceptance. An X constitutes rejection. Based on these criteria, four systems were found acceptable. Option DL-1 is significantly less costly than the reference system and shares with option SL-9 the distinction of being the lowest risk concept identified. Option DL-2 is significantly lower in risk than the reference system and is the highest performance system considered. The other two systems accepted use electrical propulsion for the orbit transfer system. Option SL-9 is essentially equal in cost to the reference system but the risk is significantly lower. Option SL-10 also possesses identical costs to the reference with the same substantial risk benefits.

8.6 DEFINITION OF SELECTED ALTERNATE SPACE SYSTEM MISSION SCENARIOS

Composite operation flows and schematic diagrams of the mission operations were generated for the selected chemical propellant and electrical propulsion space systems. For simplicity, the concepts using the integral shield without the waste payload protection system have been illustrated.

8.6.1 Chemical Propellant Systems. Figure 8.6-1 illustrates the composite operation flow derived from the generic operations flow for option DL-2. Figure 8.6-2 provides a schematic illustration of key mission events for the chemical propellant, dual launch option. Key events include:

1) Launch of the cargo launch vehicle which places the two stage orbit transfer system in low Earth orbit (LEO).

2) Launch of the waste payload to LEO in the uprated space shuttle.

3) Rendezvous between the orbit transfer system and orbiter in LEO.

4) Transfer of the waste payload to the orbit transfer system from the flight support system (FSS) which supports it in the orbiter cargo bay. Subsequent to waste payload transfer, the orbiter waits in LEO for recovery of the first stage of the orbit transfer system.

5) Injection of the expendable SOIS into heliocentric transfer orbit by the recoverable first stage.

6) After a 165 day coast, the SOIS injects itself and the waste payload into the destination heliocentric orbit at 0.85 AU.

7) Recovery of the injection stage for reuse following a retro burn and aerobraking maneuver which inserts it into LEO.

8.6.2 Electric Propulsion Systems. Figure 8.6-3 illustrates the space system composite operations flow derived for option SL-10. Key events in the mission sequence are illustrated schematically in Figure 8.6-4. The mission sequence is significantly simpler than the dual launch option. Launch vehicle ascent operations are followed immediately by deployment and checkout of the orbit transfer system while still attached to the shuttle. At the completion of checkout, waste payload transfer is accomplished by simply unlatching the waste payload from the flight support system after which the orbiter backs away from the orbit transfer system and payload and proceeds immediately to recovery and landing. Following separation, the solar electric stage begins its injection spi-

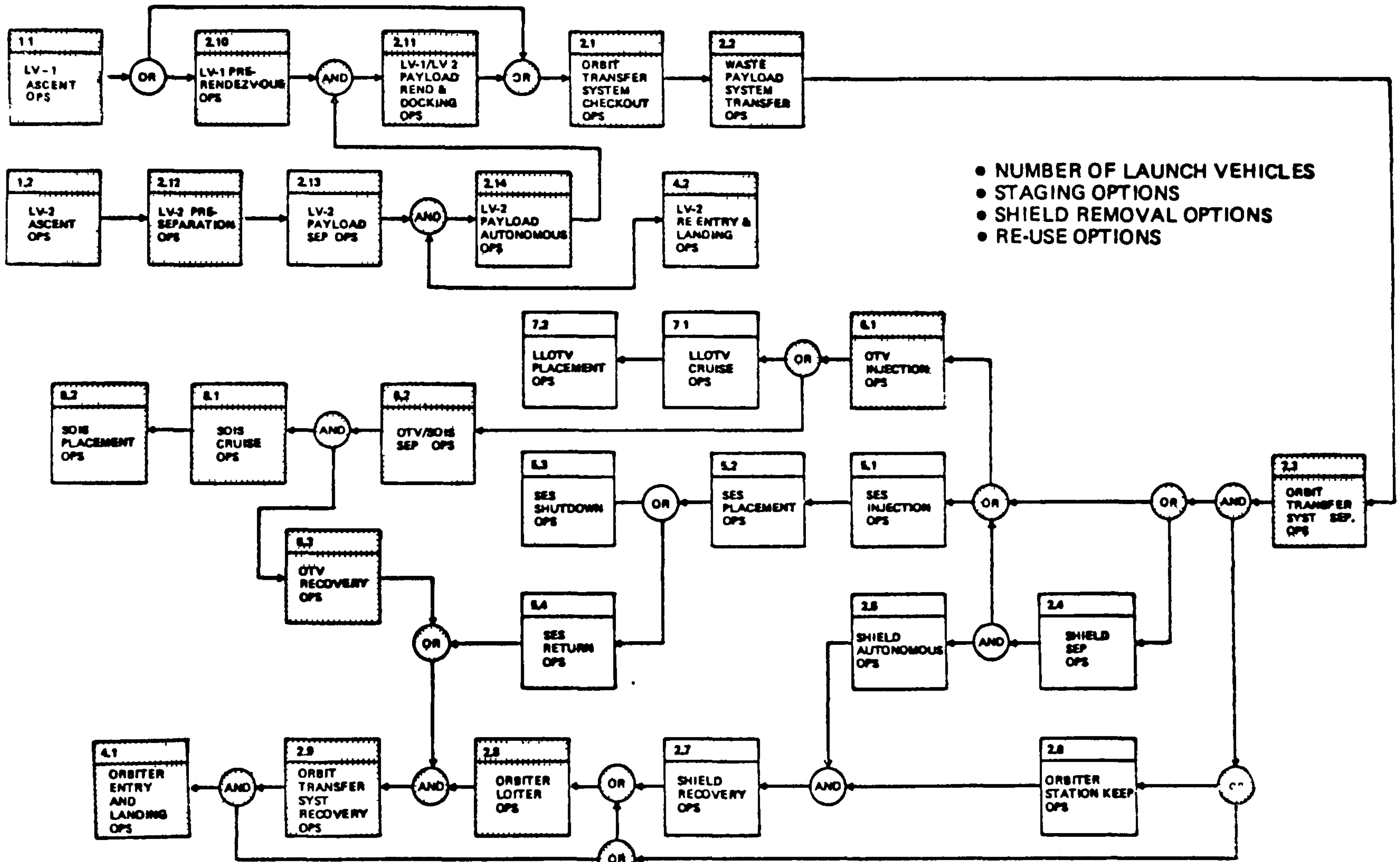


Figure 8.6-1: Space System Composite Operations Flow for Option DL-2

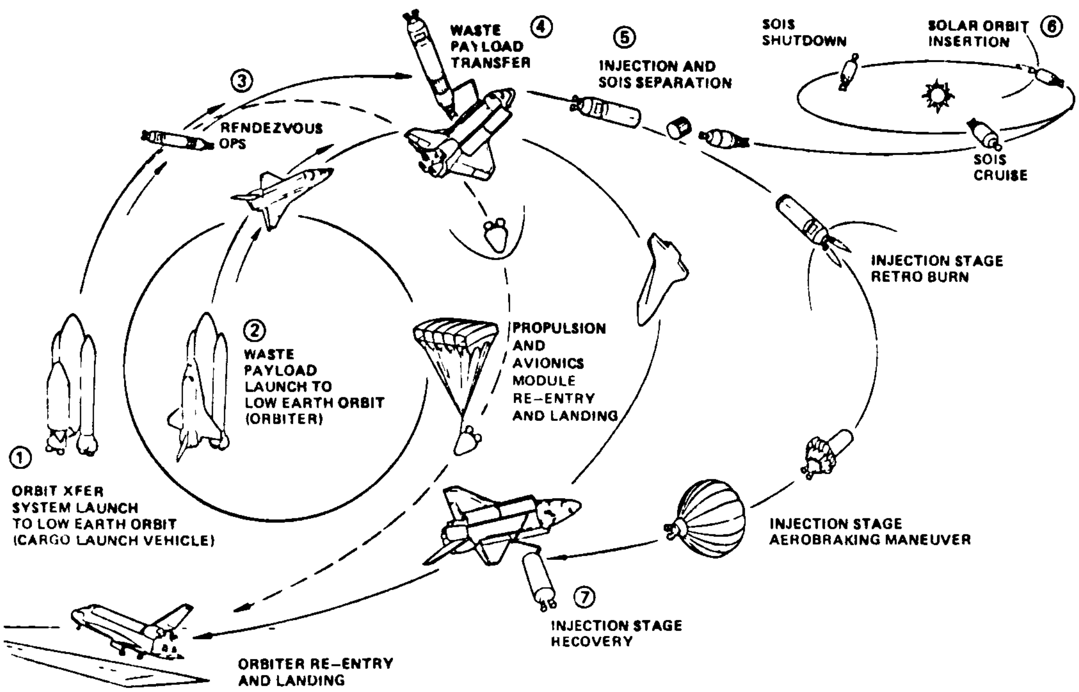


Figure 8.6-2: Orbital Operations for Option DL-2

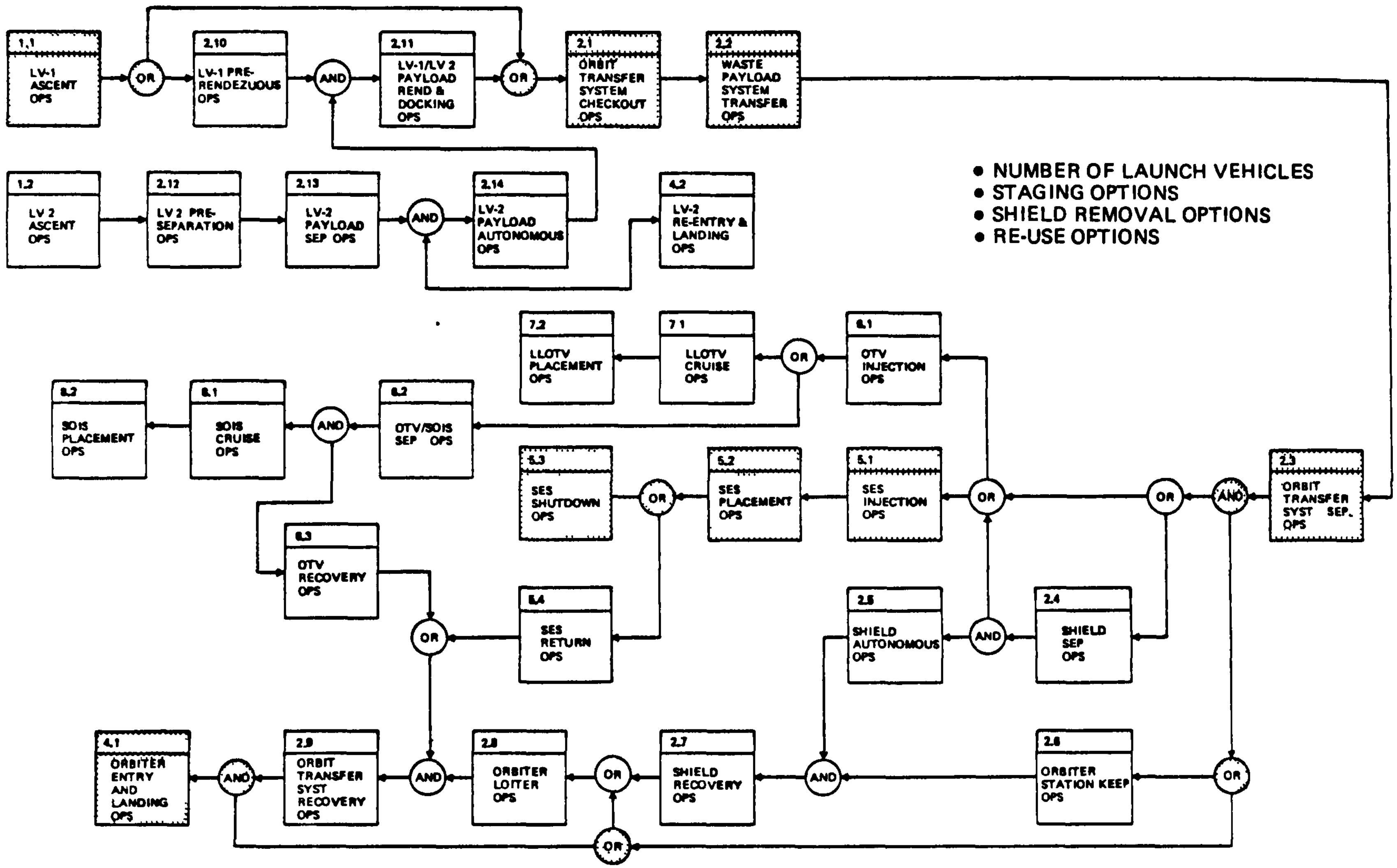


Figure 8.6-3: Space System Composite Operations Flow for Option SL-10

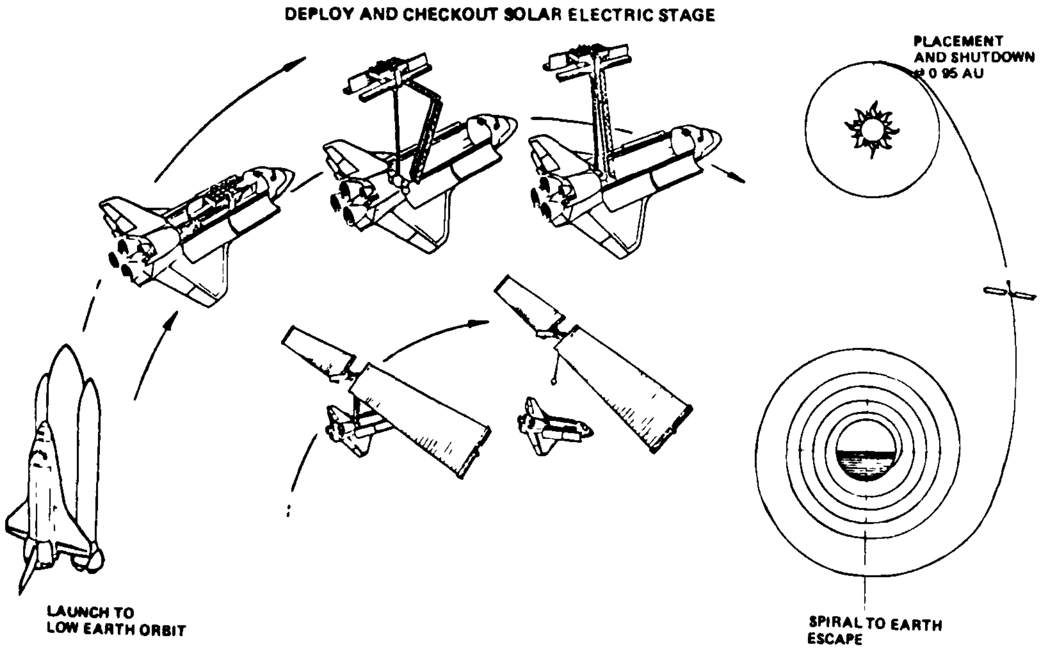


Figure 8.6-4: Orbital Operations for Option SL-10

ral. Completion of the injection spiral leaves it on a two-third turn transfer spiral to the 0.85 AU destination. Placement is accomplished by shutdown of the solar electric propulsion system when 0.85 AU is achieved, followed by verification of destination orbit parameters by ground control and the permanent shutdown of the solar electric stage, completing the deployment.

9.0 LAUNCH SITE SYSTEMS

9.1 INTRODUCTION

The primary objective of the launch site systems study was to define the impact of the selected space disposal options on facilities and operations at the Kennedy Space Center (KSC). In addition, an alternate site was to be selected and defined to allow an evaluation of the cost and benefits of accomplishing launches at a remote site. A review of the applicable references revealed that a great deal of work had been accomplished by the Marshall Space Flight Center in this area in the 1978 study (19). Accordingly, we confined our efforts to updating the KSC impact based on the Orbiter Fleet Size Study/KSC Launch Capability Analysis Study performed by KSC in 1980. Definition and evaluation of alternate sites was based on the Marshall Space Flight Center assessment in the reference cited.

Key findings are that the facilities impact of selected options is not sufficient to make it a primary differentiator between alternative space systems and that, while continued evaluation of an alternate site is warranted, this effort should be conducted as a part of the domestic and international affairs effort rather than as a part of the space system studies.

9.2 LAUNCH RATES

Launch rates for the space system options selected in Task 8 screening are illustrated in Figure 9.2-1. The single launch options, SL-9 and SL-10, require approximately 52 flights per year. The dual launch options require, respectively, 35 uprated shuttle launches and 35 shuttle derived cargo launch vehicle launches per year for option DL-1. Option DL-2 requires 30 launches of each vehicle per year. These numbers are approximately equivalent to 70 and 60 flights per year for the basic uprated space transportation system, respectively.

OPTION:	TOTAL MISSIONS	UPRATED STS MAX FLTS/YR.	SDCLV* MAX FLTS/YR.
SL-9, SL-10	540	52	--
DL-1	764	35	35
DL-2	580	30	30

● THIS IS EQUIVALENT TO 52, 70, AND 60 FLTS PER YEAR FOR THE UPRATED STS

* SHUTTLE DERIVED CARGO LAUNCH VEHICLE

Figure 9.2-1: Launch Rates for Selected Options

9.3 EVALUATION OF SPACE SYSTEMS SUPPORT FACILITIES

The impact of these launch rates on support facilities at the Kennedy Space Center is shown in Figure 9.3-1. Inspection of the tabulated requirements for

OPTION	SL-9, SL-10	DL-2	DL-1
EQUIV. STS FLIGHT RATE	52	60	70
EQUIV. ORBITER FLEETSIZE	2	4	5
KSC PAD	2	2	2
MLP	4	4	5
VAB INTEG. CELLS	2	3	3
OPF CHECKOUT BAYS	2	3	3
LRB BOOSTER PROCESSING	1	1	1
ET PROCESSING	4	5	5
ORBIT XFER SYSTEM PROCESSING*	1	1	1

THE SELECTED OPTIONS ARE APPROXIMATLEY EQUIVALENT.
FACILITY IMPACT IS NOT A DIFFERENIATOR.

* FROM OTV ØA STUDY. ASSUMES NEW DESIGN

Figure 9.3-1: Facilities Impact for Selected Options

pads, mobile launch platforms, verticle integration cells, and the other major facilities shows that the selected options are approximately equivalent in their demands. Accordingly, it was concluded that the facility impact is not a primary differentiator between alternate concepts and that further investigation in the context of this study was not warranted.

9.4 SPECIALIZED WASTE PAYLOAD PROCESSING FACILITIES

A comprehensive treatment of waste payload processing facilities at KSC is presented in Reference 19. A careful review of these results indicated that the information provided was sufficient for the purposes of this study. Further investigation was not pursued.

9.5 ALTERNATE LAUNCH SITE

A review of past studies of alternate launch sites, particularly reference 7, shows that the Kennedy Space Center is well located for the maximum overwater ground track desirable for safety in the launch of waste payloads. Feasibility of a remote launch site has, however, been established in past studies. The safety rational used as the basis of the selection of the remote site seems tenuous at best. Space disposal, if implemented, will be done on the basis of a system design which will preclude catastrophic accidents. On this basis, KSC is as suitable a launch site as any and it seems highly doubtful that a waste payload system unacceptable for safety reasons at KSC would be acceptable elsewhere on Earth.

The primary payoff of a remote launch site would appear to be political in the context of an international space disposal scenario. Until such a scenario becomes a part of the evaluation of the space option for nuclear waste isolation, it is suggested that investigation of the Kennedy Space Center as a prime launch site continue.

9.6 CONCLUSIONS

Primary conclusions reached in a brief survey of launch site systems include (1) the selected alternatives would almost double the existing space transportation system flight support facilities at the Kennedy Space Center (assuming no impact on the existing STS mission model), (2) facilities in addition to flight support facilities at KSC would be limited to a dedicated nuclear waste processing facility, and (3) continued evaluation of the alternate launch site, if pursued, should be conducted as part of the domestic and international affairs portion of the total program evaluation effort.

10.0 CONCLUSIONS

This section summarizes a few of the general conclusions reached as a result of this study.

1. The 0.85 AU heliocentric orbit space disposal destination offers the lowest verified risk of any destination studied; an alternate class of destinations in the geolunar system offers the potential for equivalent risk at significantly lower program cost (Section 2.6).

2. Orbit transfer systems for the space disposal mission benefit significantly from the use of aerobraking for injection stage recovery, the use of cryogenic propellants for the placement mission, and the use of solar electric propulsion.

3. Approximately 99% of rescue missions can be classed as nominal rescue missions, assuming reasonable reliabilities for orbit transfer systems. Implementation of the nominal rescue mission is straightforward using systems derived from the standard delivery vehicles.

4. Contingency rescue issues require further study.

5. A steel graphite composite radiation shield which doubles as primary containment offers a practical and effective approach to passive protection of the waste payload from ascent accidents and contingency reentry.

6. A waste payload protection system, which incorporates an ejection capability to protect the waste payload from ascent accident conditions, requires further study and quantification of launch vehicle reliability to determine benefits.

7. For the reference waste mix studied, the liquid rocket booster upgraded space shuttle is the most effective launch vehicle for single launch mission scenarios. For dual launch mission scenarios, the upgraded shuttle teamed with the liquid rocket booster version of the shuttle derived cargo

launch vehicle shows significant life cycle cost savings for a wide range of assumptions on DDT&E and recurring costs.

8. The Kennedy Space Center should be retained as the prime launch site for consideration in further studies. Evaluation of an alternate launch site should be considered as part of the domestic and international affairs effort.

9. System level cost and performance trades in Task 8 have defined four alternative space systems which deliver waste payloads to the the selected 0.85 AU heliocentric orbit destination at least as economically as the reference system without requiring removal of the protective radiation shield/container.

10. No concepts significantly less costly than the reference concept have been identified. The increased performance in space systems translates far more readily to risk reduction by carrying the fully shielded waste payload to the destination than it does to cost reduction.

11.0 RECOMMENDATIONS

Specific recommendations resulting from the space system study effort are summarized below:

1. Further analysis should be conducted of space disposal destinations in the geolunar system. Efforts should be aimed at defining the best geolunar destination and, when selected, validating its stability to the same level as the reference 0.85 AU heliocentric orbit destination.

2. The system safety design requirements for the reference concept should be updated to incorporate the STS crash conditions and the revised flight radiation shielding radiation limits set forth in this study.

3. An analysis of the reference integral shield waste payload system aimed at validating its ability to withstand terminal velocity impact should be conducted as the first part of a comprehensive payload effects analysis for this concept. This effort would provide preliminary verification of the technical viability of the waste payload system and, by implication, the entire space disposal system. It would also be the first step in a more extensive effort aimed at the validation and qualification of the waste payload system.

4. Thermal loading of any waste form considered should be limited to values which will preclude the possibility of post burial melt down.

5. A preliminary study of the contingency rescue mission in more detail than reported in Reference 14 is required to identify concepts and define areas for further study more specifically.

6. A system level study is required to determine the impact of alternative waste forms and mixes primarily in the area of launch rate and its effect on selection of launch vehicles and orbit transfer systems.

APPENDIX A

ACRONYMS AND ABBREVIATIONS

ACRONYMS AND ABBREVIATIONS

ABOTV	aerobraking orbital transfer vehicle
ACS	attitude control system
AMOTV	aeromaneuvering orbital transfer vehicle
APOTV	all-propulsive orbital transfer vehicle
ASE	airborne support equipment
ATP	authority to proceed
AU	astronomical unit
CDR	critical design review
DDT&E	design, development, test, and evaluation
DFI	development flight instrumentation
DOD	Department of Defense
DOE	Department of Energy
DRM	design reference mission
EMC	electromagnetic compatibility
EPS	electrical power system
ER	Earth radii
FFC	final flight certification
FOSR	flexible optical surface reflector
FSS	flight support system
FTV	facilities test vehicle
FY	fiscal year
GEO	geosynchronous Earth orbit
GEODSS	ground-based electro-optical deep space surveillance
GPS	global positioning system
HEO	high Earth orbit
HLLV	heavy lift launch vehicle

IMU	inertial measurement unit
IOC	initial operational capability
IUS	inertial upper stage
JSC	Johnson Space Center
k	thousand
KSC	Kennedy Space Center
kW	kilowatts
L/D	lift-to-drag ratio
LEO	low Earth orbit
LeRC	NASA Lewis Research Center
LLOTV	long-life OTV
M	million
MCF	mission cost factor
MECO	main engine cutoff
MGR	Mined Geological Repositor
MPS	main propulsion system
MQF	mission quantity factor
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NEP	nuclear electric propulsion
NSTL	National Space Technology Laboratory
OFI	operational flight instrumentation
ORNL	Oak Ridge National Laboratories
OTV	orbital transfer vehicle
PFC	preliminary flight certification
P/L	payload

PPU	power processing unit
PSMC	payload and sequential mass calculation
PTA	propulsion test article
QCF	qualitative cost factor
QTV	qualification test vehicle
RCS	reaction control system
REM	reaction engine modules
RF	radiofrequency
RSI	reusable surface insulation
S/C	spacecraft
SEPS	solar electric propulsion system
SES	solar electric stage
SIRTF	Shuttle Infrared Telescope Facility
SOIS	solar orbit insertion stage
SSTO	single stage to orbit
SSUS	spin stabilized upper stage
STA	structural test article
STDN	space tracking and data network
STS	space transportation system
TDRS	tracking and data relay satellite
TPS	thermal protection system
TVC	thrust vector control
WPPS	waste payload protection system

APPENDIX B

REFERENCES AND BIBLIOGRAPHY

REFERENCES

1. Friedlander, A. L. et al., "Analysis of Long-Term Safety Associated With Space Disposal of Hazardous Material," SAI-1-120-677-T11, NASA Contract NAS2-9272, Science Applications, Inc., Schaumburg, Illinois, Dec. 1977.

2. O'Leary, B. T., "Asteroid Prospecting and Retrieval," at the 4th Princeton/AIAA Conference on Space Manufacturing Facilities, AIAA Paper 79-1432, Princeton, N. J., May 1979.

3. "Feasibility of Space Disposal of Radioactive Nuclear Waste," NASA-TM-X-2912, NASA/Lewis Research Center, Cleveland, Ohio, May 1974.

4. Friedlander, A. L. and Davis, D. R., "Long-Term Risk Analysis Associated with Nuclear Waste Disposal in Space," Paper 79-175, AAS/AIAA Astrodynamics Specialists Conference, Provincetown, Mass., June 1979.

5. Friedlander, A. L. et al., "Aborted Space Disposal of Hazardous Material: The Long-Term Risk of Earth Reencounter," SAI-1-120-676-T8 (NASA CR 152199), NASA Contract NAS2-9272, Science Applications, Inc., Rolling Meadows, Ill., Feb. 1977.

6. Weissman, P. R. and Wetherill, G. W., "Periodic Trojan-Type Orbits in the Earth-Sun System," Astro. J., 79, pp 404-412, 1974.

7. Burns, R. E. et al., "Nuclear Waste Disposal in Space," NASA-TP-1225, NASA/Marshall Space Flight Center, Huntsville, Alabama, May 1978.

8. Brown, B. G. et al., "Nuclear Waste Disposal Mission Analysis," TR-223-1874, Northrop Services, Inc., Huntsville, Alabama, NASA Contract NAS8-32392, Nov. 1977.

9. Pardue, W. M., Rice, E. E., Miller, N. E., and Davis, D. K., "Preliminary Evaluation of the Space Disposal of Nuclear Waste," 8-32391(100), (NASA CR 150440), NASA Contract NAS8-32391, Battelle Memorial Institute, Columbus, Ohio, Aug. 1977.

10. Heppenheimer, T. A., "Colonies in Space," Stackpole Books, Ch. 8, pp 108-112, 1977.

11. Miller, R. H. and Smith, D. B. S., "Extraterrestrial Processing and Manufacturing of Large Space Systems," Final Report, Vols. I and II, NASA CR-161293, Sept. 1979.

12. Anonymous, "Lunar Resources Utilization for Space Construction," midterm presentation, Contract NAS9-15560, General Dynamics Convair Div., Oct. 1978.

13. Puthoff, R. L., "An 810 ft/sec Soil Impact Test of a 2-ft Diameter Model Nuclear Reactor Containment System," NASA TMX-68180, Dec. 1972.

14. Rice, E. E., Miller, N. E., Yates, K. R., Martin, W. E., and Friedlander, A. L., "Analysis of Nuclear Waste Disposal in Space - Phase II," Vols. I

and II (NASA CR 161418 and 161419), NASA Contract NAS8-32391, Battelle Columbus Laboratories, Columbus, Ohio, Mar. 1980.

15. "Satellite Surveillance System in Testing," Aviation Week & Space Technology, p. 123, Sept. 22, 1980.

16. Natenbruk, P., "Use of a Large Space Structure as an Orbital Depot for Hazardous Wastes," 30th Congress of the International Astronautical Federation (IAF), IAF Paper No. 79-209, Munich, Germany, Sept. 1979.

17. Kolenkiewcz, R. and Carpenter, L., "Stable Periodic Orbits About the Sun Perturbed Earth-Moon Triangular Points," AIAA, J., 6, pp 1301-1304, 1968.

18. Katz, J. I., "Numerical Orbits Near the Triangular Lunar Libration Points," Icarus, 25, pp 356-359, 1975.

19. Priest, C. C. et al., "Nuclear Waste Management (Space Option) - Mission and Design Analysis," NASA/Marshall Space Flight Center, Huntsville, Alabama, Dec. 1978.

20. "Concept Definition Document for Nuclear Waste Disposal in Space," NASA/Marshall Space Flight Center, Huntsville, Alabama, Jan. 1980.

21. McCarthy, J. F., Jr. et al., "Concepts for Space Disposal of Nuclear Waste," a student design project, Department of Aerospace Engineering, MIT, Oct. 1972.

22. "Space Systems for the 1980's; Vol. VI: Nuclear Waste Disposal," Aerospace Report No. ATR-72 (7259-01)-1, Vol. VI, The Aerospace Corporation, El Segundo, California, Sept. 1971.

23. "Feasibility of Space Disposal of Radioactive Nuclear Waste," NASA-TM-X-2911, NASA/Lewis Research Center, Cleveland, Ohio, Dec. 1973.

BIBLIOGRAPHY

1. Burns, R. E., "Nuclear Energy Waste - Space Transportation and Removal," NASA-TM-X-64973, NASA/Marshall Space Flight Center, Huntsville, Alabama, Dec. 1975.

2. Burns, R. E., "Radioactive Waste Disposal via Electric Propulsion," AIAA Paper 75-424, AIAA 11th Electric Propulsion Conference, New Orleans, Louisiana, Mar. 1975.

3. Burns, R. E., "The Disposal of Nuclear Waste in Space," AIAA Paper 28-280, AIAA 16th Aerospace Sciences Meeting, Huntsville, Alabama, Jan. 1978.

4. Burns, R. E. and DeField, J. G., "Disposal of Radioactive Iodine in Space," NASA-TP-1313, NASA/Marshall Space Flight Center, Huntsville, Alabama, July 1978.

5. "Comparative Assessment of Terrestrial Options for Nuclear Waste Disposal," JPL 5030-170, Jet Propulsion Laboratory, Pasadena, California, Aug. 1978.

6. "Draft Environmental Impact Statement for the Management of Commercially Generated Radioactive Waste," DOE/EIS-0046 D, U.S. Department of Energy, Washington, D.C., Apr. 1979.

7. Edgecombe, D. S., Rice, E. E., Miller, N. E., Yates, K. R., and Conlon, R. J., "Evaluation of the Space Disposal of Defense Nuclear Waste-Phase II," Vol. I, II, and III (NASA CR 161101, 161102, 161103), Battelle Columbus Laboratories, Columbus, Ohio, Jan. 1979.

8. Finnegan, P. M., Puthoff, R. L., Turnbow, J. W., "Preliminary Impact Speed and Angle Criteria for Design of a Nuclear Airplane Fission Product Containment Vessel," NASA-TM-X-2245, May 1971.

9. Friedlander, A. L., "Interplanetary Low-Thrust Transport Capabilities," Science Applications, Inc., Contract NASW-3035, May 1979.

10. Giacaglia, G, ed., "Periodic Orbits, Stability and Resonances," D. Reidel Pub., 1970.

11. Gold, T., remarks on the paper "The Tidal Loss of Satellite-Orbiting Objects and Its Implications for the Lunar Surface," by Mark Reed, Icarus, 24, pp 134-135, 1975.

12. Gregory, D. L., "A Comparison of Two Launch Vehicle Options for the In-Space Disposal of Nuclear Waste," Boeing document D180-25903-1, Mar. 1980.

13. Lipscentz, R. D., "Radioactive Waste: Policies, Technology, and Risk," a report of the Union of Concerned Scientists, pp 88-92, Ballinger Publishing Co., Cambridge, Mass., 1980.

14. MacKay, J. S., "An Evaluation of Some Special Techniques for Nuclear Waste Disposal in Space," NASA TM-X-62272, NASA/Ames Research Center, Moffett Field, California, Aug. 1973.

15. "Nuclear Waste Management Space Option - Historical Activity Summary," NASA/Marshall Space Flight Center, Huntsville, Alabama, June 1979.

16. "Nuclear Waste Management (Space Options)," MSFC in-house study midterm review, Nov. 1979.

17. "Orbital Propellant Handling and Storage Systems for Large Space Programs," Vol. II, Final Report, Contract NAS9-15305, General Dynamics Convair Div., Apr. 1978.

18. Peoples, J. A., "Thermal Control of High-Energy Nuclear Waste, Space Option," NASA-TM-78300, NASA/Marshall Space Flight Center, Huntsville, Alabama, May 1979.

19. Platt, A. M. and Schneider, K. J., "High-Level Radioactive Waste Management Alternatives," BNWL-1900, Vol. 4, Battelle Northwest Laboratories, Richland, Washington, May 1974.

20. Priest, C. C. et al., "Nuclear Waste Disposal in Space, MSFC 1979-1980 Conceptual Design Study," PD01-80-1, NASA/Marshall Space Flight Center, Huntsville, Alabama, Mar. 1980.

21. Puthoff, R. L. and Guinn, K. H., "Parametric Study of a Frangible Tube Energy Absorption System for Protection of a Nuclear Aircraft Reactor," NASA-TN-D-5730.

22. "Reference Mission Descriptions for the Space Option for Nuclear Waste Management," letter PD/01-79-52, dated Mar. 22, 1979, to H. P. Gierow (PS04) from C. R. Darwin (PD01), NASA/Marshall Space Flight Center, Huntsville, Alabama.

23. Reichert, R. G., "Transportation of Radioactive Waste Materials into the Sun," 23rd Congress of the International Astronautical Federation (IAF), Vienna, Austria, Oct. 1972.

24. Ruppe, H. O. et al., "A Nuclear Waste Deposit in Space - The Ultimate Solution for Low-Cost and Safe Disposal," 30th Congress of the International Astronautical Federation (IAF), IAF Paper No. 79-15, Munich, Germany, Sep. 1979.

25. Rochlin, G. I., Metlya, D., and Windham, P., "Space Disposal of Nuclear Wastes: Socio-Political Aspects," NASA-CR-153054, University of California, Berkeley, California, Institute of Governmental Studies, Dec. 1976.

26. Salkeld, R. and Beichel, R., "Post Shuttle Transport and Nuclear Waste Disposal in Space," 30th Congress of the International Astronautical Federation (IAF), IAF Paper No. 79-51, Munich, Germany, Sept. 1979.

27. "Space Augmentation of Military High-Level Waste Disposal," JPL 79-45, Jet Propulsion Laboratory, Pasadena, California, NASA Contract NAS7-100, May 1979.

28. "Space Operations Center - Shuttle Interaction Study - First Quarterly Review," Rockwell International Contract NAS9- 16153, Sept. 1980.

29. "Stability Requirements for Quasi-Periodic Motion in the Restricted Problem of Three Bodies," Astro. J., 79, pp 710-720.

30. "Study of Extraterrestrial Disposal of Radioactive Wastes," NASA-TM-X-71557, NASA/Lewis Research Center, Cleveland, Ohio, May 1974.

31. "Study of Extraterrestrial Disposal of Radioactive Wastes - Part II," NASA-TM-X-68147, NASA/Lewis Research Center, Cleveland, Ohio, Oct. 1972.

32. "Study of Extraterrestrial Disposal of Radioactive Wastes - Part III," NASA-TM-X-68216, NASA/Lewis Research Center, Cleveland, Ohio, Apr. 1973.

33. "Summary of the Study of Disposal of Nuclear Waste Into Space/" NASA-TM-X-68235, NASA/Lewis Research Center, Cleveland, Ohio, Apr. 1973.

34. "Summary Report of Space Transportation and Destination Considerations for Extraterrestrial Disposal of Radioactive Waste," NASA-TM-X-68211, NASA/Lewis Research Center, Cleveland, Ohio, Apr. 1973.

35. "Teleoperator Maneuvering System Orientation Briefing," Vought Corp., May 1980.

APPENDIX C

SUMMARY CHARACTERISTICS OF SURVEYED SPACE SYSTEMS

SUMMARY CHARACTERISTICS OF SURVEYED SPACE SYSTEMS

This appendix summarizes the characteristics of 36 distinct space systems for the disposal of nuclear waste that were identified in a review of the articles listed in Appendix B, References and Bibliography.

Chief characteristics of each system are listed on a single sheet, along with a recommended disposition and its rationale. Item 1 for each case references it to the article number in Appendix B from which it was derived. Numbers preceded by an R will be found in the reference section; those preceded by a B, in the bibliography.

Five systems are recommended for further consideration on the basis of potentially superior costs or risks with respect to the MSFC reference system. MSFC reference systems are identified for the STS, uprated STS, and HLLV. Additional systems will be added for consideration as they are generated.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 1 DATA SHEET

1. REFERENCES: B23
2. WASTE MIXES CONSIDERED: HLW, thermal output 1.3 kW/kg
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Integral with H_2 thruster
4. DESTINATION: Solar impact
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: Saturn V
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Saturn IV B/Centaur

   Reuse ____ Expend _X__
9. PLACEMENT STAGE: AMOS

   Reuse ____ Expend _X__
10. WASTE PAYLOAD PER MISSION (kg): 1000
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - Saturn V capability no longer exists
    - Waste mix unrealistic
    - Waste configuration unrealistic
    - Performance inferior to MSFC reference (higher cost)
    - Risk higher than MSFC reference (no rescue)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 2 DATA SHEET

1. REFERENCES: B23
2. WASTE MIXES CONSIDERED: HLW, thermal output 1.3 kW/kg
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Integral with H_2 thruster
4. DESTINATION: Solar impact
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Space tug

   Reuse _X_ Expend ___
9. PLACEMENT STAGE: AMOS

   Reuse ___ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 600
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - Waste mix unrealistic
    - Waste configuration unrealistic
    - Performance inferior to MSFC reference (higher LCC)
    - Risk higher than MSFC reference (no rescue)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 3 DATA SHEET

1. REFERENCES: B31, B32
2. WASTE MIXES CONSIDERED: All fission products (no separation) and actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Cylindrical, spherical end caps
4. DESTINATION: Earth escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Space tug

   Reuse ____ Expend _X__
9. PLACEMENT STAGE: N/A
10. WASTE PAYLOAD PER MISSION (kg): 7936 (includes shielding)
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - Unacceptable destination
    - Generally obsolete
    - Risk greater than MSFC reference
    - Cost equivalent to MSFC reference for normalized mission
13. COMMENTS: Superseded by later studies

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 4 DATA SHEET

1. REFERENCES: B31, B32
2. WASTE MIXES CONSIDERED: All fission products (no separation) and actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Cylindrical, spherical end caps
4. DESTINATION: Earth escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/payload, (2) STS/tug
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: Space tug

   Reuse ____ Expend _X__
9. PLACEMENT STAGE: N/A
10. WASTE PAYLOAD PER MISSION (kg): 14,059 (includes shielding)
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: See Case 3
13. COMMENTS: Superseded by later studies

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 5 DATA SHEET

1. REFERENCES: R3, B10
2. WASTE MIXES CONSIDERED: Actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical waste payload
4. DESTINATION: HEO or heliocentric circular at 0.9 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: Titan III E
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Centaur

   Reuse ____ Expend _X__
9. PLACEMENT STAGE: Spin stabilized solid

   Reuse ____ Expend _X__
10. WASTE PAYLOAD PER MISSION (kg): 3860
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Titan Centaur phaseout prior to disposal mission IOC
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 6 DATA SHEET

1. REFERENCES: R3, B10
2. WASTE MIXES CONSIDERED: Actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical afterbody contains spherical waste payload
4. DESTINATION: HEO or heliocentric circular at 0.9 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: Saturn V
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Centaur

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: Spin stablized solid

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 16,005
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Launch vehicle unavailable
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 7 DATA SHEET

1. REFERENCES: R3, B10
2. WASTE MIXES CONSIDERED: Actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical afterbody contains spherical waste payload
4. DESTINATION: HEO or heliocentric orbit at 0.9 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Tug
   Reuse _X_ Expend ___
9. PLACEMENT STAGE: Spin-stabilized solid
   Reuse ___ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 4170
11. DISPOSITION: Superseded by MSFC reference systems, Case 22
12. DISPOSITION RATIONALE: --
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 8 DATA SHEET

1. REFERENCES: R3, B10
2. WASTE MIXES CONSIDERED: Actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical afterbody contains spherical waste payload
4. DESTINATION: HEO
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Centaur (optimized)
9. PLACEMENT STAGE: N/A

   Reuse ____ Expend ____
10. WASTE PAYLOAD PER MISSION (kg): 8470
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Centaur capability nonexistent in time period of interest
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 9 DATA SHEET

1. REFERENCES: B14
2. WASTE MIXES CONSIDERED: Generic solid waste, 0.3 kW/km
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Integral with thermoelectric generator
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No _X_
8. INJECTION STAGE: Centaur

   Reuse ____ Expend _X__
9. PLACEMENT STAGE: RTG/ion

   Reuse ____ Expend _X__
10. WASTE PAYLOAD PER MISSION (kg): 4850
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:

    o Risk greater than MSFC reference (no rescue)

    o Cost greater than MSFC reference (no vehicle recovery)
13. COMMENTS: Similar to Case 11

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 10 DATA SHEET

1. REFERENCES: B14
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Not specified
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No ___ (Not specified)
8. INJECTION STAGE: Centaur

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: SEPS 30 kg/kW, 20 kW

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 1320 (est.)
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:

    o Risk greater than MSFC reference (no rescue)

    o Cost greater than MSFC reference (no vehicle recovery)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 11 DATA SHEET

1. REFERENCES: B1, B2
2. WASTE MIXES CONSIDERED: Actinides
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Integral with thermoeletric generator, removable shield, and thermal control to LEO; ejection provisions
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/tug, (2) STS/NEWSTAR
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: Tug

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: NEWSTAR

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 4140
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Risk greater than MSFC reference (no rescue)
    - o Cost greater than MSFC reference (no vehicle reuse, expend specialized vehicle)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 12 DATA SHEET

1. REFERENCES: R7, R8
2. WASTE MIXES CONSIDERED: Generic (mixes 3, 5, 5A)
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Waste payload packaged in conical entry vehicle; ejection provisions
4. DESTINATION: Lunar surface
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/OTV, (2) STS/waste payload
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: OTV

   Reuse ___ Expend _X_
9. PLACEMENT STAGE: N/A
10. WASTE PAYLOAD PER MISSION (kg): 4408
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Cost greater than MSFC reference (no vehicle recovery option)
    - o Risk approximately equal to MSFC reference
13. COMMENTS: Could require complex lunar surface operations (see R8)

CASE 13 DATA SHEET

1. REFERENCES: R7, R8
2. WASTE MIXES CONSIDERED: Generic (mixes 3, 5, 5A)
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Waste payload to LEO in shielded conical entry vehicle; ejection provisions
4. DESTINATION: 0.86 AU circular heliocentric orbit
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/OTV/SOIS, (2) Waste payload
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: OTV
   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Spin stabilized solid
   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 4408
11. DISPOSITION: Superseded by MSFC reference system (Case 22)
12. DISPOSITION RATIONALE:
    o Cost greater than MSFC reference (two STS launches for equivalent payload)
    o Risk identical to MSFC reference
13. COMMENTS: Remains viable option for 65K STS

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 14 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: HLLV
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: Three-stage, large, solid

   Reuse ____ Expend __X__
9. PLACEMENT STAGE: N/A
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Risk greater than MSFC baseline (no rescue)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 15 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/tug, (2) STS/tug/WPS
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: Tug

   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Tug

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:

    o Risk greater than MSFC reference (no rescue)

    o Cost greater than MSFC reference (two launches, tug expend)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 16 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: Tug

   Reuse ____ Expend __X__
9. PLACEMENT STAGE: 120 kW NEP

   Reuse ____ Expend __X__
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Risk greater than MSFC reference (no rescue)
    - o Cost greater than MSFC reference (no vehicle recovery, expend high-cost vehicle)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 17 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar system escape
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: 240 kW NEP

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: 240 kW NEP

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Risk greater than MSFC reference (no rescue)
13. COMMENTS: NEPS technology questionable for 1992-95 IOC

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 18 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit 0.86 AU, i = 20 deg
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: Tug

   Reuse ____ Expend __X__
9. PLACEMENT STAGE: SEPS

   Reuse ____ Expend __X__
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Superseded by Case 36
12. DISPOSITION RATIONALE: --
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 19 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar impact
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/tug,(2) STS/NEP/payload
7. WASTE PAYLOAD PROTECTION REMOVED: N/A
8. INJECTION STAGE: Tug

   Reuse ____ Expend __X__
9. PLACEMENT STAGE: NEPS

   Reuse ____ Expend __X__
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Risk greater than MSFC reference (no rescue)
    - o Cost greater than MSFC reference (expend high-cost vehicle)
13. COMMENTS: NEPS technology questionable for 1992-95 IOC

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 20 DATA SHEET

1. REFERENCES: R5
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar impact
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: 240 kW NEP

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: 240 kW NEP

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 3270
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Risk greater than MSFC reference (no rescue)
13. COMMENTS: NEPS technology questionable for 1992-95 IOC

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 21 DATA SHEET

1. REFERENCES: B7, B16
2. WASTE MIXES CONSIDERED: Defense HLW
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry body, spherical waste payload
4. DESTINATION: 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/OTV, (2) STS/waste payload
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO)
8. INJECTION STAGE: OTV

   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Storable SOIS

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 5500
11. DISPOSITION: MSFC reference for STS disposal of defense HLW
12. DISPOSITION RATIONALE:
13. COMMENTS: Also evaluated in Case 55 for (a) removal of waste payload system at Earth escape, and (b) solar escape destination.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 22 DATA SHEET

1. REFERENCES: R20
2. WASTE MIXES CONSIDERED: PW-4b
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical waste payload
4. DESTINATION: Solar orbit, 0.85 AU, i = 1 deg, circular
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: 100K STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO)
8. INJECTION STAGE: OTV

   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Storable SOIS

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 5000
11. DISPOSITION: MSFC reference system
12. DISPOSITION RATIONALE: --
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 23 DATA SHEET

1. REFERENCES: R20
2. WASTE MIXES CONSIDERED: PW-4b/modified PW-4b
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, spherical waste payload
4. DESTINATION: Solar orbit, 0.85 AU, i = 1 deg, circular
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: HLLV (231,000 kg payload)
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: OTV (3)
   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Storable SOIS (3)
   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 9500
11. DISPOSITION: MSFC advanced concept reference
12. DISPOSITION RATIONALE: --
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 24 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU, i = 1 deg, circular
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: IUS (3-stage)

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: Spin stabilized solid

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 1456
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Risk greater than MSFC reference (no rescue option)
    - o Cost much greater than MSFC reference (low payload, no vehicle recovery)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 25 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: Centaur

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: Storable SOIS

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 3158
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Cost greater than MSFC reference
    - o Risk greater than MSFC reference (no rescue option)
13. COMMENTS: Also evaluated for (a) solar orbit-WPS protection removed at Earth escape = 778 kg, and (b) solar system escape-WPS protection removed at LEO = 1424 kg, Earth escape = 517 kg.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 26 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit at 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes X No ___ (LEO, E/E)
8. INJECTION STAGE: IUS
   Reuse ____ Expend X
9. PLACEMENT STAGE: SEPS
   Reuse X Expend ____
10. WASTE PAYLOAD PER MISSION (kg): 1315
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Cost greater than MSFC reference (low payload/ launch)
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 27 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: NEPS

   Reuse _X_ Expend ____
9. PLACEMENT STAGE: NEPS

   Reuse _X_ Expend ____
10. WASTE PAYLOAD PER MISSION (kg): 10,260
11. DISPOSITION: Reject for detailed consideration. Assess as part of advanced prop. effort.
12. DISPOSITION RATIONALE: Insufficient characterization of NEP vehicle
13. COMMENTS:
    - o Assessment of DDT&E and operational constraints are problem areas.
    - o Also considered for (a) removal of protection at Earth escape and (b) solar system escape destination.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 28 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 2
6. BOOSTER CONFIGURATION: (1) STS/OTV, (2) STS/WPS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: 58K LLOTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: 58K LLOTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 10,773
11. DISPOSITION: Consider as STS launch vehicle option
12. DISPOSITION RATIONALE:

    o Potentially lower cost (no new vehicle development, larger payload)

    o Risk potentially equivalent to MSFC reference
13. COMMENTS: Also characterized for (a) removal of protection at Earth escape and (b) solar system escape definition.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 29 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: OTV

   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Storable SOIS

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 2403
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Cost greater than MSFC reference (low payload/launch)
13. COMMENTS: Also characterized for (a) removal of protection at Earth escape and (b) solar system escape destination.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 30 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: LLOTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: LLOTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 4327
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: Cost greater than MSFC reference (low payload/ launch)
13. COMMENTS: Also characterized for (a) removal of protection at Earth escape and (b) solar system escape destination.

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 31 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: SDV
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: LLOTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: LLOTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 16,241
11. DISPOSITION: Consider as alternate space system
12. DISPOSITION RATIONALE: Cost potentially lower than MSFC reference
13. COMMENTS:
    - o Risk issues need investigation
    - o Also characterized for (a) removal of WPS protection at Earth escape and (b) solar system escape destination

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 32 DATA SHEET

1. REFERENCES: B16
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 10
6. BOOSTER CONFIGURATION: HLLV
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (E/E)
8. INJECTION STAGE: 1207K COTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: 1207K COTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 51,981
11. DISPOSITION: Consider as alternate space system
12. DISPOSITION RATIONALE:
    - o Cost much less than MSFC reference
    - o Risk less than MSFC reference
13. COMMENTS: Restricted to SPS scenario

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 33 DATA SHEET

1. REFERENCES: B22
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.85 AU, i = 1 deg
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: 100K STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: 48K LLOTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: 48K LLOTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 9145
11. DISPOSITION: Consider as alternate space program
12. DISPOSITION RATIONALE:
    - o Cost potentially less than MSFC reference (higher payload/launch, no SOIS development)
    - o Risk equal to MSFC reference
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 34 DATA SHEET

1. REFERENCES: B26
2. WASTE MIXES CONSIDERED: Mix 2
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Conical entry vehicle, hemispherical containment vessel
4. DESTINATION: Solar orbit, 0.85 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 6
6. BOOSTER CONFIGURATION: (All) RASV*
7. WASTE PAYLOAD PROTECTION REMOVED: Yes _X_ No ___ (LEO, E/E)
8. INJECTION STAGE: OTV

   Reuse _X_ Expend ___
9. PLACEMENT STAGE: Storable SOIS

   Reuse ___ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 6000
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE:
    - o Cost greater than MSFC reference
    - o SSTO out of scope as launch vehicle
13. COMMENTS: * Reusable aerospace vehicle, BAC SSTO concept

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 35 DATA SHEET

1. REFERENCES: B26
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: 9.7-ft diameter, approx. 35 ft long
4. DESTINATION: Solar orbit, 0.86 AU
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: SSTO
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No ___ (Unspecified)
8. INJECTION STAGE: OTV

   Reuse ____ Expend _X_
9. PLACEMENT STAGE: OTV

   Reuse ____ Expend _X_
10. WASTE PAYLOAD PER MISSION (kg): 4173
11. DISPOSITION: Reject
12. DISPOSITION RATIONALE: SSTO out of scope as launch vehicle, comparative costing difficult to impossible
13. COMMENTS: None

SPACE DISPOSAL FOR NUCLEAR WASTE

CASE 36 DATA SHEET

1. REFERENCES: Current study
2. WASTE MIXES CONSIDERED: Generic
3. WASTE PAYLOAD SYSTEM CONFIGURATION: Generic
4. DESTINATION: Solar orbit, 0.85 AU, i = 1 deq
5. NUMBER OF BOOSTER LAUNCHES PER MISSION: 1
6. BOOSTER CONFIGURATION: 100K STS
7. WASTE PAYLOAD PROTECTION REMOVED: Yes ___ No ___ (Unspecified)
8. INJECTION STAGE: Aerobraked OTV
   Reuse _X_ Expend ____
9. PLACEMENT STAGE: Uprated SEPS space base
   Reuse _X_ Expend ____
10. WASTE PAYLOAD PER MISSION (kg): TBS est. greater then 10,000
11. DISPOSITION: Consider as alternate space system
12. DISPOSITION RATIONALE: Cost estimate less than MSFC reference (payload greater than MSFC reference, SOIS reusable)
13. COMMENTS: Establish performance, fleet size, rescue options

APPENDIX D

REFERENCE SYSTEM DESCRIPTION

(The material in this appendix is taken from the Battelle-Columbus Laboratories Volume II, Technical Report on Analysis of Nuclear Waste Disposal in Space--Phase III, to National Aeronautics and Space Administration Marshall Space Flight Center (Contract No. NAS8-32391), DPD No. 580, DR No. 4, March 31, 1980.)

2.3 Overall Reference Mission

The overall reference mission, described in this section and developed during the course of this study, represents the concept for which most of the analyses in this report and in the NASA/MSFC documentation were conducted. Because of the many possible variations within the space disposal option, one point of reference is necessary. The major aspects of the reference mission are illustrated in Figure 2-3. This mission profile has been divided into seven major activities. The first two are expected to be the responsibility of the Department of Energy (DOE) and the last five are expected to be NASA's. These are:

(1) Nuclear Waste Processing and Payload Fabrication (DOE)
(2) Nuclear Waste Ground Transport (DOE)
(3) Payload Preparation at Launch Site (NASA)
(4) Prelaunch Activities (NASA)
(5) Uprated Space Shuttle Operations (NASA)
(6) Upper Stage Operations (NASA)
(7) Payload Monitoring (NASA).

Considerations of rescue and recovery systems are discussed in Sections 2.4 and 2.6. Also, see Section 6.2 for a detailed discussion of rescue mission technology. More complete definitions for individual system elements are discussed in Section 2.4. The following paragraphs provide the reader with a general overview of the reference mision.

2.3.1 Nuclear Waste Processing and Payload Fabrication (DOE)

Typically, spent fuel rods from domestic power plants would be transported to the waste processing and payload fabrication sites via conventional shipping casks. Using the Purex process, high-level waste containing fission products and actinides, including 0.1 percent plutonium and 0.1 percent uranium, would be processed from these spent fuel rods (see Section 3.2.1). The high-level waste would be formed into a cermet matrix by a calcination and hydrogen reduction process (see Section 3.1.2). The waste form would then be fabricated into a 5000 kg spherical payload (see Section 3.5). Within a remote shielded cell, the waste payload is loaded into a container, the container is then closed and sealed, inspected, decontaminated, and packaged into a flight-weight gamma radiation shield assembly (see Section 3.4.2). During these operations and subsequent interim storage at the processing site, the waste payload is cooled by an auxiliary cooling system.

2.3.2 Nuclear Waste Ground Transport (DOE)

The shielded waste container would then be loaded into a ground transportation shipping cask (see Figure 2-4). This cask, which provides additional shielding, thermal, and impact protection for the waste container

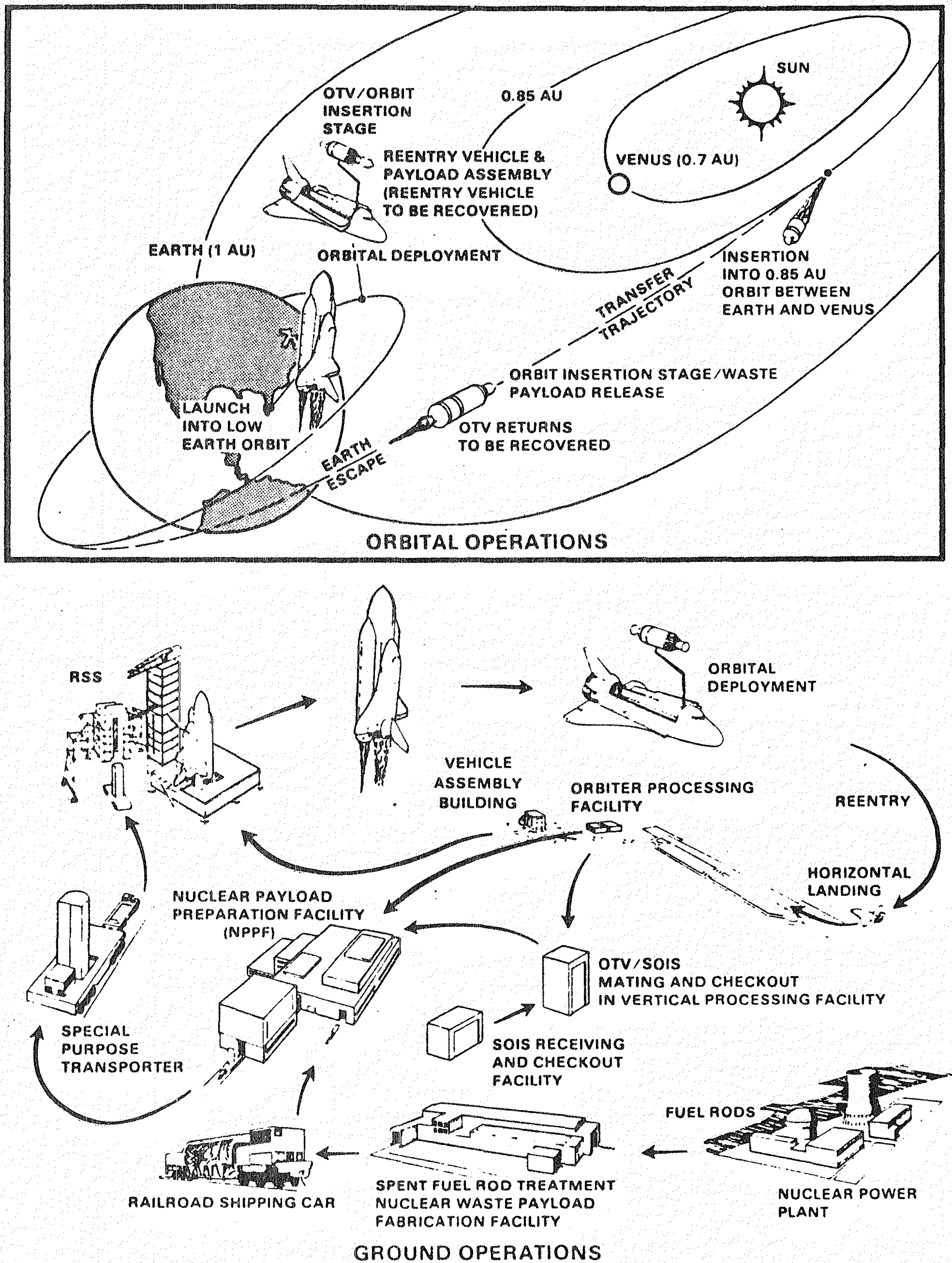


FIGURE 2-3. GROUND AND SPACE OPERATIONS FOR REFERENCE SPACE DISPOSAL MISSION

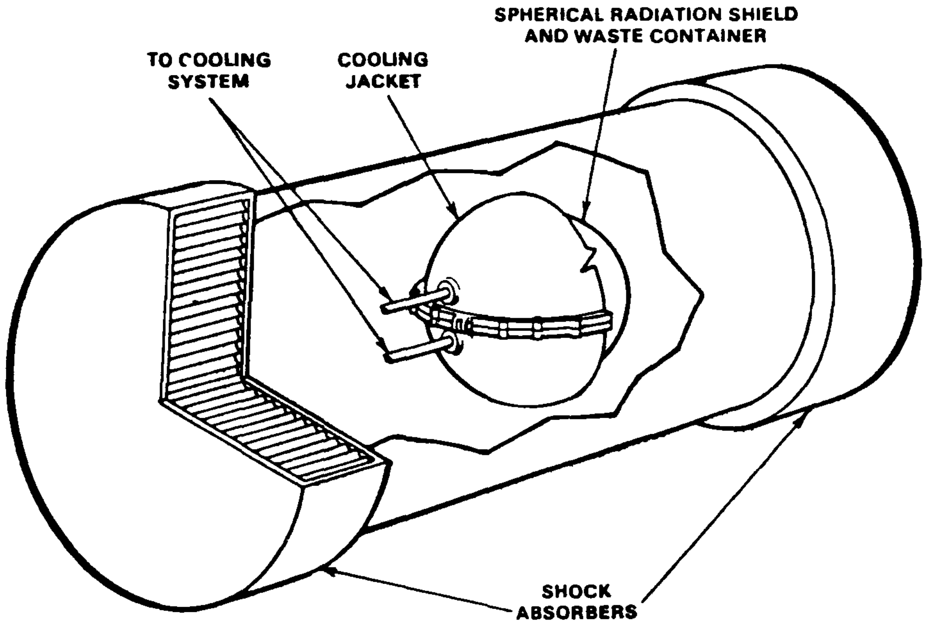


FIGURE 2-4. REFERENCE CONCEPT FOR NUCLEAR WASTE PAYLOAD SHIPPING CASK FOR TERRESTRIAL TRANSPORT

to comply with the Nuclear Regulatory Commission/Department of Transportation regulations, is then loaded onto a specially designed rail car for transporting the waste container from the waste payload fabrication site to the Kennedy Space Center (KSC), Florida launch site. Once the cask reaches the launch site, it is offloaded into the Nuclear Payload Preparation Facility (NPPF).

2.3.3 Payload Preparation at Launch Site (NASA)

The NPPF is expected to provide interim storage capability for up to three shielded waste containers, which affords efficient preparation for launches plus capacity for unplanned delays. During storage, additional radiation shielding, thermal control, monitoring and inspection of the waste container would be provided.

2.3.4 Prelaunch Activities (NASA)

In preparation for launch of the nuclear waste into space, the integrated Space Shuttle waste payload is prelaunch checked in the NPPF. The integrated Shuttle payload consists of: the waste form; the container; the radiation shield; the reentry vehicle (RV), which protects and structurally supports the waste in the Orbiter cargo bay (see Figure 2-5); the Solar Orbit Insertion Stage (SOIS), which circularizes the waste payload into the solar orbit disposal destination, and the Orbit Transfer Vehicle (OTV), which provides escape from low Earth orbit and insertion into the heliocentric transfer trajectory. Integration and checkout in the NPPF is typical of future ground flow planning at Kennedy Space Center and parallel to the current use of the Vertical Processing Facility (VPF) by the Inertial Upper Stage (IUS). Prelaunch checkout in the NPPF includes verification of the payload and the payload to Orbiter interface systems. The Orbiter interface would be simulated by standardized equipment. Typically, storable propellant loading would occur in the NPPF to minimize the hazard of propellant loading while the payload is in the Shuttle cargo bay on the launch pad.

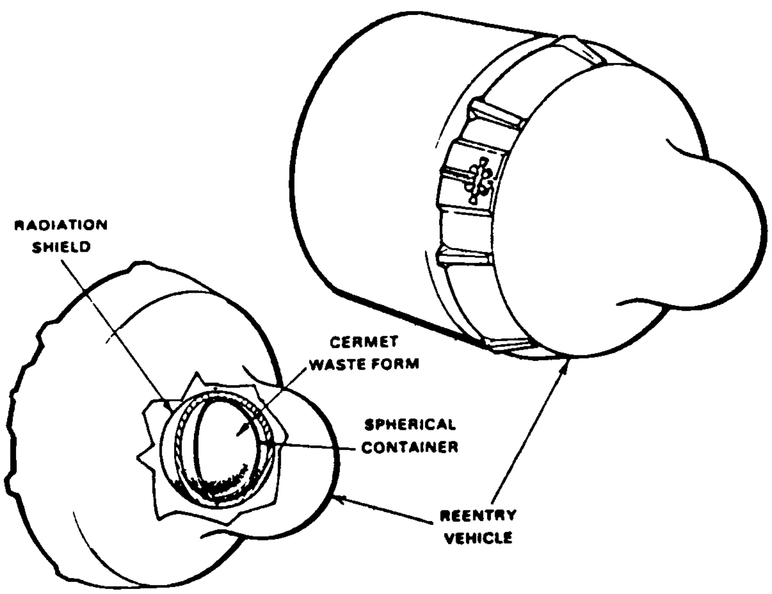


FIGURE 2-5. REFERENCE CONCEPT OF A LOADED REENTRY VEHICLE

Transfer of the payload to the launch pad's Rotating Service Structure (RSS), is accomplished by a special purpose transporter which maintains the Shuttle payload in the proper position for installation in the Orbiter cargo bay (see Figure 2-3). The payload is transferred from the NPPF to the pad after the Shuttle vehicle installation at the launch pad has been completed. The payload is then positioned by the RSS and installed in the Orbiter cargo bay. After payload installation, propellant loading of the OTV, and final systems checkout the decision to launch is made.

2.3.5 Uprated Space Shuttle Operations (NASA)

One Uprated Space Shuttle vehicle would be readied for launch for a given disposal mission. The to-be-constructed Pad C at KSC Launch Complex 39 would be used for this mission. Pad A or B could be used to launch the Shuttle vehicle that carries the rescue OTV, should it be required.

The Uprated Space Shuttle (45,400 kg payload to low Earth orbit), that is to perform the disposal mission, is launched at a 108 degree south azimuth to a 300 km (160 n.mi.) circular orbit inclined 38 degrees to the equator. A small degree of yaw steering is required such that early land overflight of various populated land masses (West Indies and South Africa) is avoided. Once on orbit, the loaded reentry vehicle (RV) in the Shuttle Orbiter cargo bay is remotely translated aft a short distance and structurally latched to the SOIS. Using the OTV payload bay rotation structure, the OTV, SOIS, and loaded RV are deployed from the Orbiter bay. Actual separation from the rotation structure is accomplished by a spring powered deployment system. After the OTV, SOIS, and loaded RV configuration has been stabilized in a fixed attitude, the Orbiter will move to a safe distance away to limit the radiation dose to the crew from the unshielded payload. At this time, the waste payload would be mechanically transferred by remote control to the SOIS payload adapter, and the OTV/SOIS/waste payload is oriented for the Earth escape propulsive burn. The reentry vehicle would remain in orbit and be recovered and returned to KSC by the Shuttle Orbiter.

The traffic model for the reference space disposal concept is given below in Table 1.

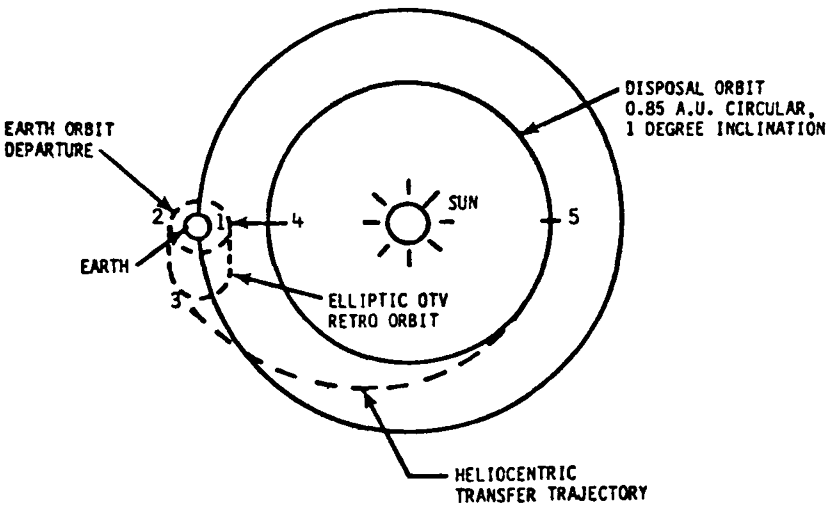
TABLE 1. PROJECTED UPRATED SPACE SHUTTLE TRAFFIC MODEL FOR COMMERCIAL HIGH-LEVEL NUCLEAR WASTE DISPOSAL MISSIONS (1992-2003)

	Year												
	92	93	94	95	96	97	98	99	00	01	02	03	Total
Uprated Space Shuttle Flights	10	20	50	50	50	50	50	60	60	60	60	60	580

2.3.6 Upper Stage Operations (NASA)

After the OTV/SOIS/waste payload system has passed final systems checkouts, the OTV propulsive burn would place the SOIS and its attached waste payload on the proper Earth escape trajectory. Control of the propulsive burn from low Earth orbit would be from the aft deck payload control station on the Orbiter, with backup provided by a ground control station. After the burn is complete, the SOIS/waste payload is then released. In about 160 days the payload and the storable liquid propellant SOIS would travel to its perihelion at 0.85 A.U. about the Sun. [One astronomical unit (A.U.) is equal to the average distance from the Earth to the Sun.] The SOIS would then place the payload in its final space disposal destination by reducing the aphelion from 1.0 to 0.85 A.U. To aid in obtaining the desired orbital lifetimes, this orbit would be inclined to the ecliptic plane by 1 degree. The recovery burns of the OTV would use the remaining OTV propellant to rendezvous with the

Shuttle Orbiter for its subsequent recovery, refurbishment, and reuse on a later mission. The reference OTV/SOIS mission profile is shown in Figure 2-6.



1-2 Uprated Space Shuttle (45,400 kg payload) ascent from Earth to a 300 km circular orbit with a 38° inclination

2-3 Prime OTV burn of approximately 10 min for escape from low-Earth orbit on elliptic solar orbit transfer trajectory with perigee of 0.85 A.U. and 1° inclination to the ecliptic. The ΔV for this maneuver is 3350 m/sec.

3 OTV separation from the SOIS/nuclear waste payload and retro burn to an elliptic Earth orbit. The ΔV for this maneuver is 640 m/sec. The OTV lifetime for return to the Orbiter is approximately 50 hours. The apogee for this orbit is 61,000 km.

4 OTV circularization into the 300 km, 38° inclination recovery orbit. The ΔV is 2770 m/sec.

5 SOIS and payload circularization into 0.85 A.U., 1° inclination to the ecliptic, solar orbit. The ΔV is 1160 m/sec.

FIGURE 2-6. REFERENCE OTV/SOIS MISSION PROFILE

2.3.7 Payload Monitoring (NASA)

The Earth escape trajectory of the SOIS/waste payload would be monitored by ground-based radar systems and telemetry from the SOIS and OTV. The final disposal orbit achieved would be monitored by NASA's Deep Space Network. Once the proper disposal orbit has been verified, no additional monitoring is necessary. However, monitoring could be re-established in the future, if required.

2.4 Reference System Element Definitions

The definitions for reference mission system elements are described below. These definitions have been used in almost all cases for the work documented in this report. Twelve major system elements identified for this definition document are:

(1) Waste Source
(2) Waste Mix
(3) Waste Form
(4) Waste Processing and Fabrication Facilities
(5) Payload Container, Shielding, and Reentry Vehicle
(6) Ground Transport Vehicles and Casks
(7) Launch Site Facilities
(8) Uprated Space Shuttle Vehicle
(9) Upper Stages
(10) Payload Ejection System
(11) Docking System
(12) Space Destination.

Definitions for the reference mission system elements follow.

2.4.1 Waste Source

The primary waste source would be nuclear waste generated by the operation of commercial nuclear power plants (see Section 3.2.1). Table 2-1 provides the most realistic projections of waste generation (assuming 200 GWe by the year 2000) found in the literature.[2-14] By assuming that the waste must be at least 10 years old before it can be disposed of in space, and that reprocessing capacities should be able to process the waste according to the proposed schedule, the annual amount of waste available for disposal is given. Projections of the mass available for space disposal are also given as a function of year. The mass of waste available annually for space disposal, in cermet form, would increase to 310 metric tons (MT) by the year 2000.

2.4.2 Waste Mix

Waste generated using the Purex process (fission products, actinides including approximately 0.1 percent Pu, and 0.1 percent U) is considered to be the reference waste mix composition. The specific reference waste used for the 1979-1980 Space Option study activity was defined as the Battelle Northwest Laboratory PW-4b waste mix (see Section 3.2.1 for details).[2-15] The elemental definition of this waste is given in Table 2-2, isotopic definition is given in Section 3.2.1.

TABLE 2-1. PROJECTED NUCLEAR POWER GENERATION REPROCESSING CAPACITY AND COMMERCIAL HIGH-LEVEL WASTE AVAILABLE FOR SPACE DISPOSAL

Year	Cumulative Power, GWe	Cumulative Waste, MTHM	Annual Nuclear Waste Available for Disposal, MTHM/yr	Annual High-Level Purex Waste in Cermet Form Available for Space Disposal, MT/yr
1979	61.9	5890(c)	0	0
1980	74.8	7690	0	0
1981	87.3	9790	0	0
1982	101.1	12,220	0	0
1983	115.4	14,990	0	0
1984	131.4	18,140	0	0
1985	144.3	21,600	0	0
1986	157.1	25,370	0	0
1987	164.9	29,330	0	0
1988	174.0	33,510	0	0
1989	180.9	37,850	5890(c)	410
1990	186.5	42,330	1800	125
1991	188.9	46,860	2100	146
1992	190.1	51,420	2430	169
1993	192.5	56,040	2770	193
1994	194.0	60,700	3150	219
1995	195.0	65,380	3460	241
1996	196.0	70,080	3500	244
1997	197.0	74,810	3960	275
1998	198.0	79,560	4180	290
1999	199.0	84,340	4340	301
2000	200.0	89,140	4480	310

(a) From: Yates, K. R., and Park, U. Y., "Projections of Commercial Nuclear Capacity and Spent-Fuel Accumulation in the United States", Fuel Reprocessing, pp. 350-352 (June 1979).
(b) MTHM is metric tons heavy metal.
(c) Includes 4400 MTHM PW-4b existing as of 1978.
(d) Assumes 40.8 kg/MT waste for space disposal and a cermet waste form loading of 58.7 percent.
(e) Computed by multiplying 5890 MTHM by 0.0408 MT/MTHM and dividing by 0.587.

TABLE 2-2. REFERENCE WASTE MIX COMPOSITION FOR SPACE DISPOSAL (BNWL PW-4b)

	Constituent	Amount, kg/MTHM		Constituent	Amount, kg/MTHM
Inerts	Na_2O	--	Fission Products (Cont'd.)	TeO_2	0.725
	Fe_2O_3	1.511		Cs_2O	2.880
	Cr_2O_3	0.345		BaO	1.567
	NiO	0.141		La_2O_3	1.480
	P_2O_5	0.672		CeO_2	3.323
	Gd_2O_3	--		Pr_6O_{11}	1.482
				Nd_2O_3	4.522
Fission Products	Rb_2O	0.354		Pm_2O_3	0.123
	SrO	1.059		Sm_2O_3	0.924
	Y_2O_3	0.598		Eu_2O_3	0.200
	ZrO_2	4.944		Gd_2O_3	0.137
	MoO_3	5.176			
	Tc_2O_7	1.291	Actinides	U_3O_8	1.169
	RuO_2	2.972		NpO_2	0.865
	Rh_2O_3	0.480		PuO_2	0.010
	PdO	1.483		Am_2O_3	0.181
	Ag_2O	0.088		Cm_2O_3	0.040
	CdO	0.097			
				TOTAL	40.8

Source: Reference 2-15.

2.4.3 Waste Form

The reference waste form for space disposal is the ORNL iron/nickel-based cermet. It has been chosen over may other waste forms (see Section 3.1.1). A cermet is a dispersion of ceramic particles in a continuous metallic phase. The reference cermet is formed by a process involving dissolution and precipitation from molten urea followed by calcination and hydrogen reduction to produce a continuous metallic phase (see Section 3.1.2). Nonhydrogen reducible oxides form the ceramic portion of the ceramic/metal matrix waste form. This waste form has been shown to have superior properties as compared to other potential waste forms for space disposal. The iron/nickel-based cermet has high waste loading (58.7 percent), a relatively high thermal conductivity (14 Watts/m-C at 300 C), a high density (6.7 g/cc), a good specific heat (0.583 kJ/kg-C), and a high structural integrity.

2.4.4 Waste Processing and Payload Fabrication Facilities

The waste processing and payload fabrication facilities are assumed to be collocated. The reference waste mix requires a waste processing facility utilizing the Purex process. After separation and generation of the aqueous waste stream, approximately 5 years of storage would occur before further processing. The waste would then be put into its final cermet waste form.

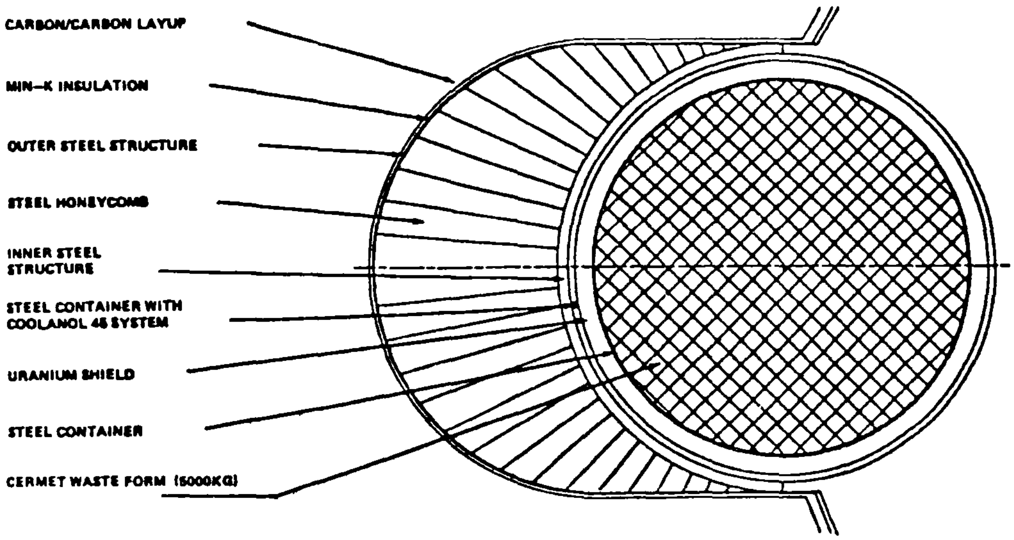
The waste payload fabrication facilities would provide a series of interconnected, shielded cells for loading the waste form into containers, closing, sealing, inspecting, decontaminating containers, and ultimate insertion into the flight-weight radiation shield assembly. Each cell would have provisions to connect the waste container to an auxiliary cooling system. Each facility would provide interim storage for a number of shielded waste packages and equipment/systems for cask handling and rail car loading.

2.4.5 Payload Container, Shielding, and Reentry Vehicle Systems

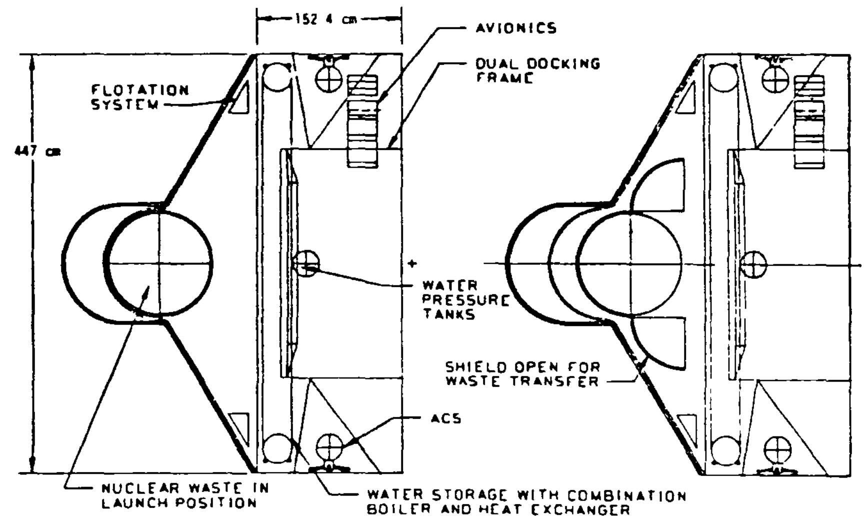
The primary containment for the radioactive waste is a stainless steel spherical container (1.27 cm thick -- see Figure 2-5). This container must provide high integrity containment for the waste during the various defined mechanical and thermal loads to which the total payload is subjected in anticipated normal and accident conditions. These loads would be mitigated in varying degrees by the waste form itself, the gamma radiation shield assembly, by the shipping cask which provides additional gamma radiation shielding for ground transportation, and by the reentry vehicle (RV) system during the prelaunch and boost phase of the disposal mission. The container would be designed to dissipate the heat generated within the waste form by passive cooling to the space environment during the orbital operations. During any normal operation, the maximum temperature of the waste form should not exceed the normal limiting temperature of 1200 C. During launch and on orbit while the waste is in the RV, the temperature is to be controlled with assistance of various auxiliary cooling systems located on the Shuttle Orbiter and the RV. If accidents occur, the temperature of the waste and container material may exceed the normal limit but must not exceed that which would cause loss of containment.

The container would be housed in a flight-weight radiation shield assembly for the period prior to leaving the waste fabrication facility until attached to the OTV in Earth orbit. This flight-weight shielding would be designed to limit the radiation level to 2 rem/hr at 1 meter from its surface (see Section 2.5.2.4). Additional shielding would need to be provided by temporary shielding at the NPPF and RSS, and possibly a shadow shield in the Shuttle Orbiter for the crew (see Section 2.5.1.1). The spherical radiation shield would be of depleted uranium sandwiched between two layers of stainless steel.

The radiation shielded waste container would be enclosed in a protective reentry and impact shield prior to launch and during the boost phase (see Figure 2-7). This system would be designed to minimize the probability of



Reentry Vehicle Nose Cross Section



Reentry Vehicle Configuration

FIGURE 2-7. REFERENCE REENTRY VEHICLE CONFIGURATION AND NOSE CROSS SECTION containment breach as a result of accidents or malfunctions which could occur during the prelaunch, launch, suborbital, orbital, or unplanned reentry phases of the mission. The protection shield would consist of an outer layer of carbon/carbon thermal protection and MIN-K insulation and a 61 cm thick steel honeycomb impact structure (at the nose point). The thermal protection is completely around the payload, whereas, the impact structure covers only the nose of the RV. The waste payload/reentry vehicle/SOIS/OTV configuration, as positioned in the Shuttle Orbiter, is shown in Figure 2-8.

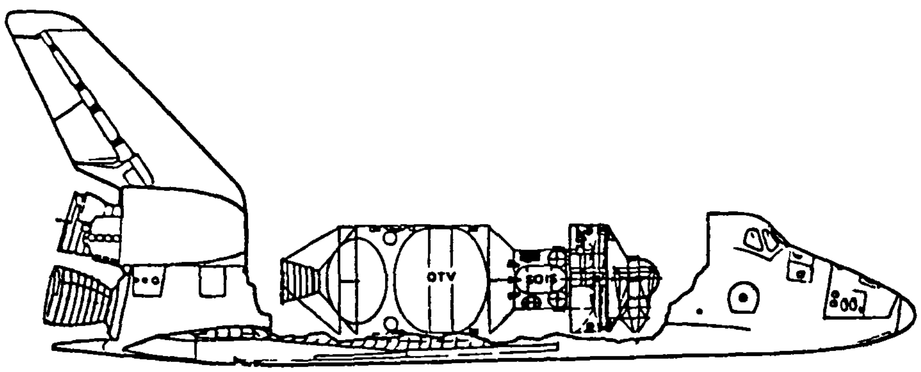


FIGURE 2-8. LOCATION OF PAYLOAD/RV/SOIS/OTV CONFIGURATION IN SHUTTLE ORBITER CARGO BAY

2.4.6 Ground Transport Vehicles and Casks

For transport from the waste fabrication facility to the launch site, the waste containers and associated flight-weight shielding would be housed in a shipping cask affording additional shielding, thermal and impact protection to meet the Nuclear Regulatory Commission/Department of Transportation regulations.(2-16) The cask (see Figure 2-4) is expected to be licensed by the Nuclear Regulatory Commission. The cask would be transported from the payload fabrication facilities to the KSC launch site on a specially designed rail car that adequately supports and distributes the weight of the cask and provides acceptable tie downs. In addition, the rail car would carry an auxiliary cooling system to reliably cool the waste package.

2.4.7 Launch Site Facilities

The reference launch site for launching nuclear waste payloads during the early phase of the program (early-1990's) would be Launch Complex 39 at Kennedy Space Center Florida. New facility construction and equipment expected during this period is noted below.

- A secure, sealed, environmentally controlled, Nuclear Payload Preparation Facility (NPPF) to store, cool, monitor, assemble, and checkout the waste payload systems from the time the shielded nuclear waste container arrives at KSC until the time the loaded payload reentry vehicle is moved to the launch pad.

- A dedicated, special-purpose transporter to move the nuclear waste payload from the NPPF to the Rotating Service Structure (RSS) at the launch pad. This includes construction of a roadway or tracks for the transporter to use.

The other currently planned Shuttle and upper stage launch facilities may or may not be adequate to support the additional Shuttle launches required by a high launch rate (60 per year) nuclear waste disposal program. Further analysis of the nuclear waste disposal traffic model coupled with the Space Shuttle traffic model and current turnaround time lines is needed. It is expected that additional facilities would likely be needed at the higher launch rates. Facilities envisioned are:

- A dedicated Space Shuttle launch pad (Pad C) for launching nuclear waste payloads. The waste payload would be installed in the Shuttle Orbiter at the pad. A specially designed RSS is required for this mission.

- A third Mobile Launch Platform (MLP) for transporting built-up Shuttles from the Vehicle Assembly Building (VAB) to the launch pads is required.

- A third firing room in the Launch Control Center (LCC) would have to be activated to handle the increased number of Space Shuttle flights dedicated to the nuclear waste disposal program. This firing room would be used exclusively for the waste disposal missions.

2.4.8 Uprated Space Shuttle Vehicle

During the early years of a space disposal program, the Uprated Space Shuttle (45,400 kg payload to low Earth orbit--see Figure 2-9) would represent an ideal vehicle to carry out the boost phase of the space transport. The National Aeronautics and Space Administration is now managing the development of the Space Shuttle (to be operational at Kennedy Space Center in 1981), a new class of space booster that is a highly reliable, reusable, low-cost vehicle that can transport payloads to low Earth orbit and back. It is anticipated that the Space Shuttle vehicle expected to fly space missions will be uprated around 1990. This uprating involves the use of a higher performance and environmentally cleaner Liquid Rocket Booster (LRB) as a replacement for the Solid Rocket Booster (SRB).

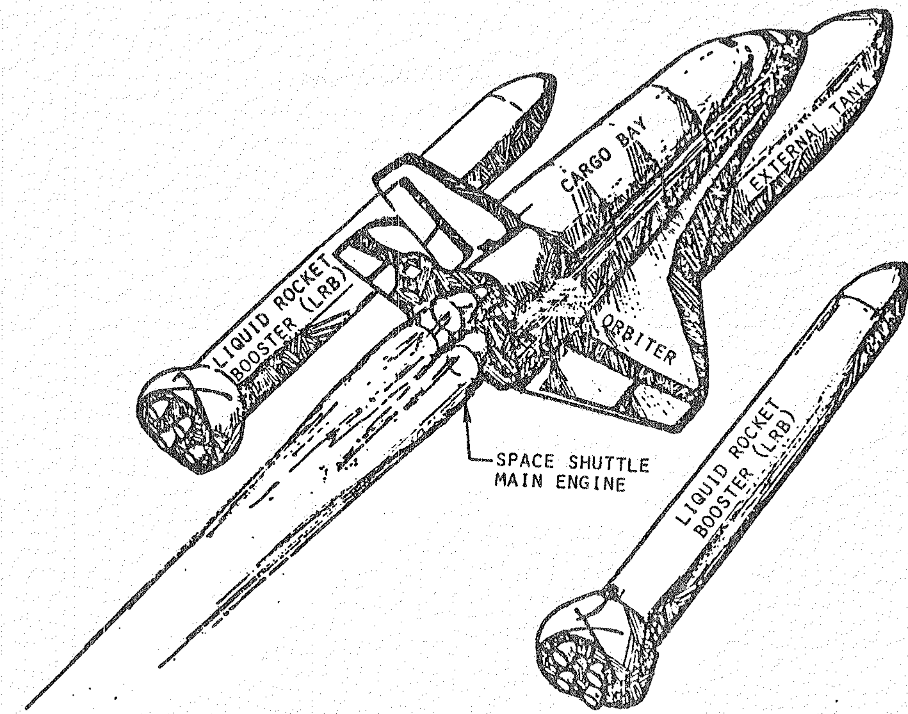


FIGURE 2-9. UPRATED SPACE SHUTTLE VEHICLE

The Uprated Space Shuttle consists of a piloted reusable orbiting vehicle (the Orbiter) mounted on an expendable External Tank (ET) containing hydrogen/oxygen propellants and two recoverable and reusable Liquid Rocket Boosters (LRB's). The propellants for the LRB's are RP-1 (kerosene) and liquid oxygen (LOX), having an oxidizer to fuel ratio of 2.9. The Orbiter will have three main hydrogen/oxygen liquid rocket engines and a cargo bay 18.29 m long and 4.57 m in diameter. At launch, both the LRB's and the Orbiter's three liquid rocket engines would burn simultaneously. After about 140 seconds and after the Space Shuttle vehicle attains an altitude of about 45 km (28 miles), the LRB's would be separated and subsequently recovered from the Atlantic Ocean. The ET is jettisoned before the Orbiter goes into orbit. The Orbital Maneuvering System (OMS) is then propels the Orbiter into the desired Earth orbit. The Orbiter with its crew and payload (weighing up to 45,400 kg) would remain in orbit to carry out its mission, normally from 1 to 7 days, but, when required, as long as 30 days. When the mission is completed, the Orbiter would be deorbited and piloted back to the launch site for an unpowered landing on a runway. The Orbiter and LRB's would subsequently be refurbished and reflown on other space missions. References 2-17, 2-18, and 2-19 provide additional information about the standard Space Shuttle and its capabilities. Reference 2-20 provides data on LRB's for the Uprated Space Shuttle. Table 2-3 provides a reference mass summary for the Uprated Space Shuttle Vehicle.

Small changes to the Space Shuttle system may be required to provide a safer and more reliable launch vehicle. These modifications have not yet been identified.

TABLE 2-3. MASS SUMMARY FOR UPRATED SPACE SHUTTLE VEHICLE

Vehicle Component/Element	Mass, kg	Weight, lb
Orbiter		
Dry (Less Engines)	63,875	140,821
Engines	9,063	19,980
Personnel and Equipment	1,197	2,640
Residuals and Reserves	4,212	9,285
Total Inert	78,347	172,726
OMS/RCS Propellants	12,322	27,166
Total at Liftoff	90,669	199,892
External Tank (ET)		
Dry	32,757	72,217
Residuals and Reserves	4,276	9,428
Total Inert	37,034	81,645
Usable Propellants (LOX/LH_2)	711,196	1,567,918
Total at Liftoff	748,230	1,649,563
Liquid Rocket Boosters (Both)		
Dry	126,269	278,376
Residuals	4,853	10,700
Total Inert	131,122	289,076
Usable Propellants (LOX/RP-1)	1,080,480	2,382,050
Total at Liftoff	1,211,602	2,671,126
Payload	45,360	100,000
Total Vehicle at Liftoff	2,095,861	4,620,581

Source: Reference 2-20.

2.4.9 Upper Stages

Two different upper stages have been defined for use for the nuclear waste disposal mission: (1) an Orbit Transfer Vehicle (OTV), and (2) a storable propellant Solar Orbit Insertion Stage (SOIS). The OTV is a completely reusable and recoverable stage; the SOIS is expendable. Orbital rescue capability would be performed by the OTV and SOIS systems.

The OTV is defined as a reusable LOX/LH_2 chemical propulsion stage similar to the cryogenic OTV defined in the past few years for possible development and use with the Space Shuttle. This vehicle would have separate propellant tanks, an oxidizer/fuel (O/F) mixture ratio of 6 and a delivered specific impulse (I_{sp}) of 470 seconds. It would also have an advanced, redundant, avionics and attitude control system. Figure 2-10 is a pictoral of the OTV.

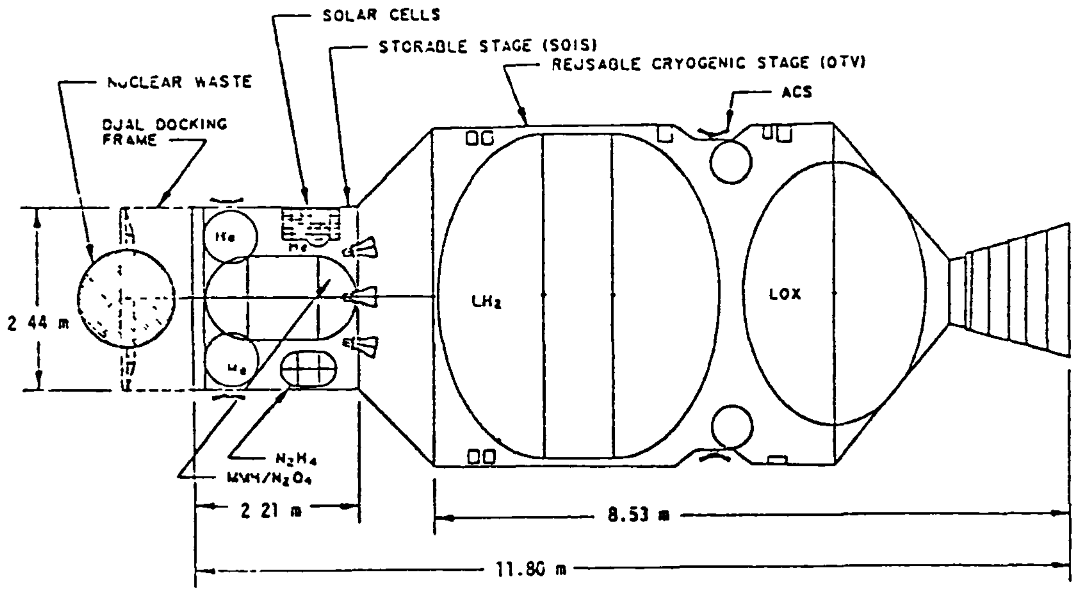


FIGURE 2-10. REFERENCE OTV/SOIS/WASTE PAYLOAD CONFIGURATION

The storable-propellant, pressure-fed SOIS would be sized to provide a specific impulse of 289 sec. The NASA/MSFC scaling relation for the SOIS is given below:

$$M_{B.O.} = 810 + 0.0522\ M_P$$

where $M_{B.O.}$ and M_P are the SOIS stage burnout mass (kg) and propellant loading mass (kg), respectively. The propellant loading includes a 15 percent flight reserve. This stage would have three off-the-shelf (Space Shuttle-Reaction Control System) pressure-fed engines at a thrust level of 3870 N (870 lb) each, monomethylhydrazine/nitrogen tetroxide (MMH/NTO) propellants, a guidance and control system, and a payload docking adapter system compatible with the docking system. The stage would be designed to adequately withstand the adverse nuclear radiation, and space environments experienced while coasting 160 days before firing.

The rescue vehicle would be a Shuttle launched OTV/SOIS system. It would include appropriate provisions for targeting and docking with the nuclear waste container attached to an OTV/SOIS, the nuclear waste container attached to an SOIS only, a payload reentry vehicle assembly, or an unshielded, separated waste container. It would be reusable or expendable depending upon the rescue mission. This vehicle would be required to have an on-orbit stay time of at least 300 hours with little reduction expected in reliability or performance. The rescue vehicle may be returned to Earth by the Shuttle Orbiter at the end of the cycle for refurbishment, if recoverable. Definition of other rescue requirements are expected as studies progress.

2.4.10 Payload Ejection System

A payload ejection system would be incorporated into the pallet which supports the reentry vehicle (see Figure 2-8). This system would employ 16 small solid propellant rocket motors which would be ignited to eject the loaded reentry vehicle from the Orbiter cargo bay in the event of a serious on-pad or ascent failure. The reentry vehicle would be designed to withstand the expected sea or ground impact environment.

2.4.11 Payload Docking and Transfer System

The payload docking and transfer system would be launched into orbit attached to the reentry vehicle. The docking/transfer system would be used to transfer the waste payload container from the reentry vehicle to the SOIS payload adapter. The payload adapter would be designed to jettison the nuclear payload during the very low probability occurrence of a grossly inaccurate OTV propulsive burn. This action would prevent possible reentry and allow subsequent recovery by a Shuttle or OTV rescue vehicle.

2.4.12 Space Destination

The reference space destination for the nuclear waste disposal mission is an orbital region between the orbits of the Earth and Venus. The nominal circular orbit is defined as 0.85 ± 0.01 A.U. The orbital inclination about the Sun is specified as 1 degree from the ecliptic plane.

APPENDIX E

SYSTEM SAFETY DESIGN REQUIREMENTS FOR THE REFERENCE SYSTEM

(The material in this appendix is taken from the Battelle-Columbus Laboratories Volume II, Technical Report on Analysis of Nuclear Waste Disposal in Space--Phase III, to the National Aeronautics and Space Administration Marshall Space Flight Center (Contract No. NAS8-32391), DPD No. 580, DR No. 4, March 31, 1980.)

2.5 System Safety Design Requirements for Reference Concept

This section defines system safety design requirements for the reference nuclear waste disposal in space mission (also see Section 3.3 and Appendix C). These requirements provide the guidelines against which nuclear waste payloads may be considered acceptable from a radiological safety point of view. These requirements should be used for future studies, and modified as changes in the concept occur.

The general safety design objectives for the nuclear waste payload and/or its associated system components are: (1) to contain the solid radioactive waste materials and (2) to limit the exposure of humans and the environment to the radioactive waste materials. For normal operations, complete containment and minimum exposure are required; for potential accident situations, the degree of containment and degree of interaction shall result in an acceptable risk to humans and the environment.

The following subsections describe the general and specific system design requirements for the nuclear waste disposal in space mission.

2.5.1 General System Safety Design Requirements

The general system safety design requirements for the nuclear waste disposal in space mission involve considering of the following:

(1) Radiation Exposure
(2) Containment
(3) Accident Environments
(4) Criticality
(5) Postaccident Recovery
(6) Monitoring Systems.

The following paragraphs define the requirements that should be followed for the reference system concept design activity.

2.5.1.1 Radiation Exposure

Radiation exposure limits for normal operations for the public and ground crews will be those contained in ERDA-MC-0524 and shown in Table 2-4. Radiation exposure limits for Space Shuttle crew members during normal operations will be those contained in the Space Shuttle Flight and Ground Specification, JSC 07700, Volume X, Revision A, Chapter 7.4 and shown in Table 2-5.

The normal radiation exposure limits for the current terrestrial transportation of nuclear waste materials would also apply to ground

TABLE 2-4. NORMAL OPERATIONS EXPOSURE LIMITS FOR INDIVIDUALS IN CONTROLLED AND UNCONTROLLED AREAS

INDIVIDUALS IN CONTROLLED AREAS:

Type of Exposure	Exposure Period	Dose Equivalent (Dose or Dose Commitment[a], rem)
Whole body, head and trunk, gonads, lens of the eye[b], red bone marrow, active blood forming organs.	Year Calendar Quarter	5[c] 3
Unlimited areas of the skin (except hands and forearms). Other organs, tissues, and organ systems (except bone).	Year Calendar Quarter	15 5
Bone.	Year Calendar Quarter	30 10
Forearms.[d]	Year Calendar Quarter	30 10
Hands[d] and feet.	Year Calendar Quarter	75 25

INDIVIDUALS IN UNCONTROLLED AREAS:

Annual Dose Equivalent or Dose Commitment (rem)[e]

Type of Exposure	Based on dose to individuals at points of maximum probable exposure	Based on an average dose to a suitable sample of exposed population
Whole body, gonads, or bone marrow	0.5	0.17
Other organs	1.5	0.5

(a) To meet the above dose commitment standards, operations must be conducted in such a manner that it would be unlikely that an individual would assimilate in a critical organ, by inhalation, ingestion, or absorption, a quantity of a radionuclide(s) that would commit the individual to an organ dose which exceeds the limits specified in the above table.
(b) A beta exposure below an average energy of 700 Kev will not penetrate the lens of the eye, therefore, the applicable limit for these energies would be that for the skin (15 rem/year).
(c) In special cases with the approval of the Director, Division of Operational Safety, a worker may exceed 5 rem/year provided his average exposure per year since age 18 will not exceed 5 rem per year.
(d) All reasonable efforts shall be made to keep exposures of forearms and hands to the general limit for the skin.
(e) In keeping with ERDA policy on lowest practicable exposure, exposures to the public shall be limited to as small a fraction of the respective annual dose limits as is practicable.

Source: See Section 4.5, Reference 4-7.

TABLE 2-5. RADIATION EXPOSURE LIMITS FOR SPACE SHUTTLE FLIGHT CREWS[a]

Constraints, rem	Bone Marrow, 5 cm	Skin, 0.1 mm	Eye, 3 mm	Testes[c]
1 year average daily rate	0.2	0.6	0.3	0.1
30-day maximum	25	75	37	13
Quarterly maximum[b]	35	105	52	18
Yearly maximum	75	225	112	38
Career limit	400	1200	600	200

Notes:

(a) These exposure limits and exposure rate constraints apply to all sources of radiation exposure. In making trade-offs between man-made and natural sources of radiation, adequate allowance must be made for the contingency of unexpected exposure. These data are from Space Shuttle Flight and Ground Specification, JSC 07700, Volume X, Revision A, Chapter 7.4.
(b) May be allowed for two consecutive quarters followed by six months of restriction from further exposure to maintain yearly limit.
(c) These dose and dose rate limits are applicable only where the possibility of oligospermia and temporary. infertility are to be avoided. For most manned space flights, the allowable exposure accumulation to the Germinal Epithelium (3 cm) will be the subject of a risk/gain decision for particular program, mission, and individuals concerned.

transportation of nuclear waste payloads. The radiation limits (49CFR 173.393) are given as:

- 1 m from external container surface...1000 mrem/hour
- External surface of transport vehicle...200 mrem/hour
- 2m from external surface of transport vehicle...10mrem/hour
- Normally occupied position of transport vehicle...2mrem/hour.

For accident conditions of terrestrial transport, dose rates are limited to 1000 mrem/hour at 1 meter from the external surface of the waste package. For launch/reentry accidents, higher dose limits are expected because of the anticipated lower probability for these accidents.

2.5.1.2 Containment

The containment requirements (also see Section 3.3) are different for the various portions of the disposal mission. For the current reference mission, four different types of containment configurations are used: (1) shipping cask/auxiliary shielding/flight-weight radiation shield/container/waste, (2) auxiliary shielding/flight-weight radiation shield/container/waste, (3) reentry shield/impact shield/flight-weight radiation shield/container/waste, and (4) container/waste. For all normal operations, the systems will be designed such that no release of radioactive material occurs. Configuration (1) must survive probable shipping accidents without major release. Configuration (2) must survive probable handling accidents without major containment breach. Configuration (3) must survive all handling, on-pad, booster ascent to orbit, and reentry accidents without major containment breach. Configuration (4) must be designed to survive the two postulated reentry conditions (see Section 2.5.1.3.4) with only minimal release possible. The accident environments for which the designs of these generic configurations must survive are given below.

2.5.1.3 Accident Environments

The accident environments that need to be considered in the design of containment and other auxiliary systems are as follows:

- Shipping accident
- Ground handling accident in NPPF
- On-pad or near-pad Uprated Space Shuttle vehicle failure
- Reentry accidents.

2.5.1.3.1 Shipping Accident Environments (for Configuration 1)

DOT and NRC regulations, as defined in 49 CFR 170 to 179 and 10 CFR 71, will be assumed for the ground shipment of nuclear waste payloads from the waste payload fabrication facility to the launch site. The following sequential test environments for shipping cask accidents are given below. Initial conditions are to be assumed the same as the normal condition.

- A 9-m drop in worst orientation onto an unyielding surface

- A 1-m drop in the worst orientation onto the end of 15-cm-diameter, 20-cm-high bar (mild steel)

- A 30-min. ground fire at 800 C followed by 3 hours of no artificial cooling; with a cask emissivity of 0.9 and cask absorbtivity of 0.8

- An 8-hour emersion in 0.9 m of water.

At the end of this test, surface radiation of the shipping cask should not exceed 1 rem/hour at 1 m from the surface, the contents must remain subcritical, and only minute radioactive material releases are allowed (see 10 CFR 71).

2.5.1.3.2 Handling in NPPF (for Configuration 2)

The payload systems, auxiliary support equipment and facilities must be designed to minimize the occupational radiation exposure to workers (see Table 2-4). Care must also be taken to insure that if certain subsystem failures occur during handing in the NPPF, radiation exposure and contamination is kept to as low as reasonably achievable. The handling area in the NPPF will be designed to be a total containment vessel.

2.5.1.3.3 On- or Near-Pad or Ascent Booster Accident (for Configuration 3)

The payload package must be designed to withstand the following nominal accident environments (developed in Section 4.2) in sequence without a major breach of primary containment. Initial conditions are assumed to be the normal condition.

- A blast side-on overpressure of 250 N/cm^2, a reflected overpressure of 1700 N/cm^2 and side-on and reflected impulses of 2.0 and 15.0 $N\text{-}s/cm^2$, respectively, in worst orientation. (Based upon a 10 percent yield of the ET propellants--see Table 4-9, Section 4.2.3)
- A potential edge-on penetration of 1 per m^2 of impacting fragments, assumed to be discs 100 cm in diameter and 0.56 cm thick, having a mass of 12 kg, and moving at 500 meters per second. The worst orientation is assumed. (Based upon data in Section 4.3.4)
- A heat flux of 3500 kW/m^2 for 15 seconds from a liquid propellant fireball. (Based upon results described in Section 4.2.1.3)
- A 60-min. ground fire at 1100 C followed by 2 hours of no artificial cooling. (Based upon results described in Section 4.2.2.3)
- An impact in the worst orientation onto an unyielding surface at 10 percent higher than the predicted terminal velocity.
- An impact in the worst orientation into 25 C water at a velocity 10 percent higher than the predicted terminal velocity.

2.5.1.3.4 Reentry Accidents (for Configuration 4)

The payload container and waste must be able to withstand reentry into the Earth's atmosphere and impact onto the Earth's surface without the dispersion of significant quantities of radioactive material. The reentry environments are defined as follows:

- A decaying reentry trajectory to provide maximum heating energy
- A reentry trajectory which provides the maximum heating flux
- An impact on an unyielding surface and in the ocean at a velocity 10 percent higher than the predicted terminal velocity.

The response of the container and waste to the reentry environments must be calculated after the specific reentry conditions have been determined by analysis.

2.5.1.4 Criticality

The radioactive waste package shall be subcritical (K-effective $\leq$ 0.95) for normal operations or any possible credible accident during processing, fabrication, handling, storage, or transport to the space destination.

2.5.1.5 Postaccident Recovery

Postaccident recovery teams will be made part of the operational disposal system. They will be responsible for all accident recovery operations, including accidents involving processing, payload fabrication and railroad shipment, payload preparation at the launch site, the launch and possible reentry.

2.5.1.6 Monitoring Systems

Monitoring systems will be developed for the overall system such that overall mission safety can be assured. Examples of such systems include devices for measuring radiation, temperature and, possibly, pressure in the waste package, and instruments to provide data for tracking the payload after it is placed into its solar orbit disposal region.

2.5.2 Specific System Safety Design Requirements

The following paragraphs define specific design requirements established for the elements of the reference disposal concept (see Sections 2.3, 2.4, 3.3 and Appendix C). As the reference concept changes, these requirements are also expected to change.

2.5.2.1 Waste Form

For normal conditions, the cermet fabrication temperature of 1200 C shall not be exceeded. For accident conditions, the cermet decomposition temperature of 1450 C shall not be exceeded.

2.5.2.2 Waste Processing and Payload Fabrication Facilities

The design and operation of these facilities will follow current proposed regulations, as specified for reprocessing plants.

2.5.2.3 Payload Primary Container

For normal conditions, the primary stainless steel container shall not exceed the creep limit temperature of 427 C. No chemical and physical interaction will occur between the cermet waste form and the container. For accident conditions, the primary container must not exceed the melt temperature of 1450 C.

2.5.2.4 Flight Radiation Shielding

Radiation shielding for flight systems will be designed to limit radiation to no more than 2 rem per hour at 1 meter from the shield surface under normal conditions. Auxiliary shielding will be designed such that radiation exposure limits (see Tables 2-4 and 2-5) for ground personnel and flight crews are not exceeded during handling or flight operations.

For normal conditions, the temperature limit for the depleted uranium/stainless steel flight radiation shield is 427 C. For accident conditions, the radiation shield must not exceed the uranium melt temperature of 1130 C.

APPENDIX F

ORBIT TRANSFER VEHICLE RETURN TRAJECTORY TRADES

A circular Heliocentric orbit of .85 AU radius has been selected for space disposal of nuclear waste. The objective of this task was to characterize the mission profile used by a chemical OTV in transferring the waste from LEO at an altitude of 160 n. mile to the destination orbit. For purposes of analysis the mission can be separated into Geocentric and Heliocentric segments.

Heliocentric Mission Segment

Figure 1 illustrates the Heliocentric mission segment. A two burn Hohmann transfer is used to transit from the Earth's orbit at 1 AU to the disposal orbit at .85 AU.

$$r_a = 1\,AU = 149.5 \times 10^6\ km$$

$$r_p = .85\,AU = 127.08 \times 10^6\ km$$

$$a = \frac{r_a + r_p}{2} = 138.28 \times 10^6\ km$$

$$\mu_s = 1.32495 \times 10^{11}\ km^3/s^2 \quad \text{FOR HELIOCENTRIC ORBITS}$$

for elliptical orbits: $$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \qquad (1)$$

$$V_a = 28.539\ km/s \qquad V_p = 33.573\ km/s$$

for circular orbits $$V_c = \sqrt{\frac{\mu}{r_c}}$$

at 1 AU $V_c = 29.771\ km/s$

at .85 AU $V_c = 32.290\ km/s$

The required velocity changes for the transfer are:

$$\Delta V_a = 29.771\ km/s - 28.539\ km/s = 1.232\ km/s$$

$$\Delta V_p = 33.573\ km/s - 32.290\ km/s = 1.283\ km/s$$

A circular Heliocentric orbit of .85 AU radius has been selected for space disposal of nuclear waste. The objective of this task was to characterize the mission profile used by a chemical OTV in transferring the waste from LEO at an altitude of 160 n. mile to the destination orbit. For purposes of analysis the mission can be separated into Geocentric and Heliocentric segments.

<u>Heliocentric Mission Segment</u>

Figure 1 illustrates the Heliocentric mission segment. A two burn Ho~~t~~hmann transfer is used to transit from the Earth's orbit at 1 AU to the disposal orbit at .85 AU.

$$r_a = 1\,AU = 149.5 \times 10^6\ km$$
$$r_p = .85\,AU = 127.08 \times 10^6\ km$$

$$a = \frac{r_a + r_p}{2} = 138.28 \times 10^6\ km$$

$\mu_s = 1.32495 \times 10^{11}\ km^3/s^2$ FOR HELIOCENTRIC ORBITS

for elliptical orbits: $$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \qquad (1)$$

$V_a = 28.539\ km/s$ $\qquad V_p = 33.573\ km/s$

for circular orbits $V_c = \sqrt{\frac{\mu}{r_c}}$

at 1 AU $\quad V_c = 29.771\ km/s$

at .85 AU $\quad V_c = 32.290\ km/s$

The required velocity changes for the transfer are:

$$\Delta V_a = 29.771\ km/s - 28.539\ km/s = 1.232\ km/s$$
$$\Delta V_p = 33.573\ km/s - 32.290\ km/s = 1.283\ km/s$$

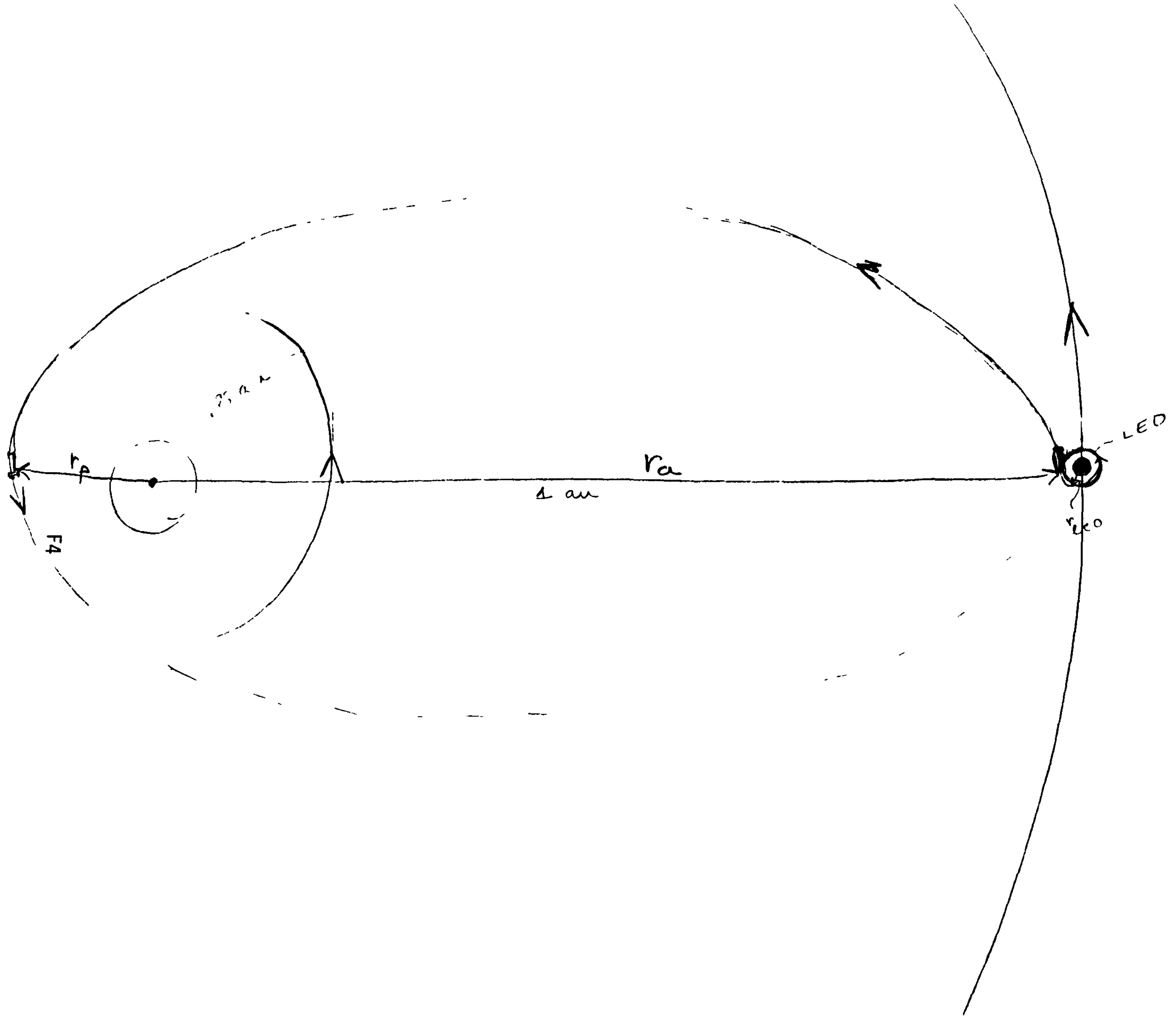


FIG. 1 Heliocentric Mission Segment

The period of the elliptical transfer orbit is defined by:

$$\tau = \frac{2\pi}{\sqrt{\mu}} a^{3/2} \qquad (2)$$

$$\tau = 28.068 \times 10^6 \, s = 7797 \, hr$$

The transit time from 1 AU to .85 AU is 3898 hours.

Geocentric Mission Segment

The objective of the Geocentric segment of the mission is to achieve the desired apohelion velocity for the transfer from 1 AU to .85 AU. This requires escaping the earth's gravitational field with an excess velocity, in a direction opposite to earth's heliocentric motion, that will reduce the heliocentric velocity to the transfer orbit apohelion velocity. Earth escape is achieved via a hyperbolic orbit. The required hyperbolic velocity at perigee is given by:

$$V_h = \sqrt{\frac{2\mu}{r} + V_\infty^2} \qquad (3)$$

where $\mu = 398601.2 \ km^3/s^2$ FOR GEOCENTRIC ORBITS,

$$V_\infty = \Delta V_a = V_c - V_a = 1.232 \ km/s$$

$$r = r_{LEO} = 6672 \ km$$

then $V_h = 11.000 \ km/s$

At LEO $V_c = 7.730 \ km/s$, the change in velocity is:

$$\Delta V_h = 11.000 - 7.730 = 3.270 \ km/s$$

The geometry of the escape orbit is shown in Figure 2. Its characteristics can be determined from the following relationships.

$$e = \frac{r_p^2 V_p^2}{\mu} - 1 \qquad (4)$$

$$a = \frac{r_p}{e - 1} \qquad (5)$$

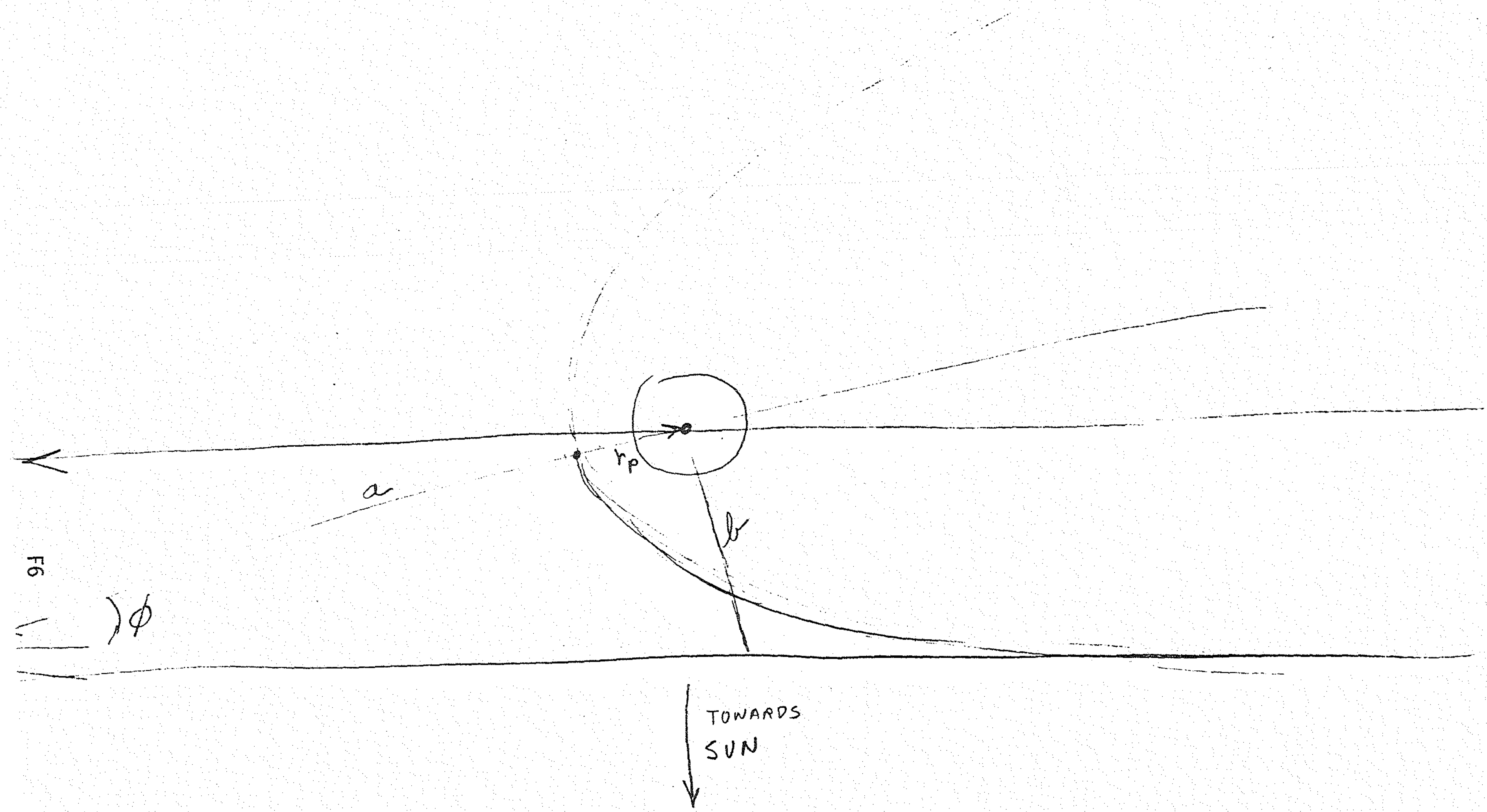


Figure 2. Escape Orbit Geometry

$$V = \sqrt{\mu\left(\frac{2}{r} + \frac{1}{a}\right)} \qquad (6) \qquad \text{(for hyperbolic orbits)}$$

$$V_\infty = \frac{\mu/r}{V} \tan\phi \qquad (7)$$

The orbit parameters are:

$$e = 1.0253$$

$$a = 263468 \text{ km}$$

$$\phi = 12.78^\circ$$

$$b = 59665 \text{ km}$$

Figure 3 depicts the orientation of the escape orbit with respect to the Heliocentric transfer orbit. The right ascension of the low earth orbit must be selected such that intersection of the LEO plane and the plane of the Heliocentric transfer orbit is perpindicular to the radius vector from the sun. It's value varies directly with the time of year and inclination of the orbit. For any inclination and launch site location there is a daily launch opportunity that allows the hyperbolic injection to be achieved without any ΔV penalty for plane change.

In the reference mode of operation for the nuclear waste mission the OTV boosts a Solar Orbit Insertion Stage (SOIS) carrying the waste payload into the hyperbolic orbit. The OTV then retro fires into an elliptical orbit returning it to LEO. At LEO a further reduction in velocity circularizes the orbit. The braking maneuver can be accomplished by propulsion or by aerobraking. Several issues are involved in characterizing the return orbit. There is a discrete amount of time required to separate the OTV from the SOIS and reorient it for the retro maneuver. This time has an impact on the magnitude and direction of the velocity change. The magnitude of the retro ΔV determines the semi-major axis, "a", of the return elliptical orbit and as a result its period. In addition the total ΔV required for the return to LEO is affected by the magnitude of the retro ΔV, for cases in which the retro

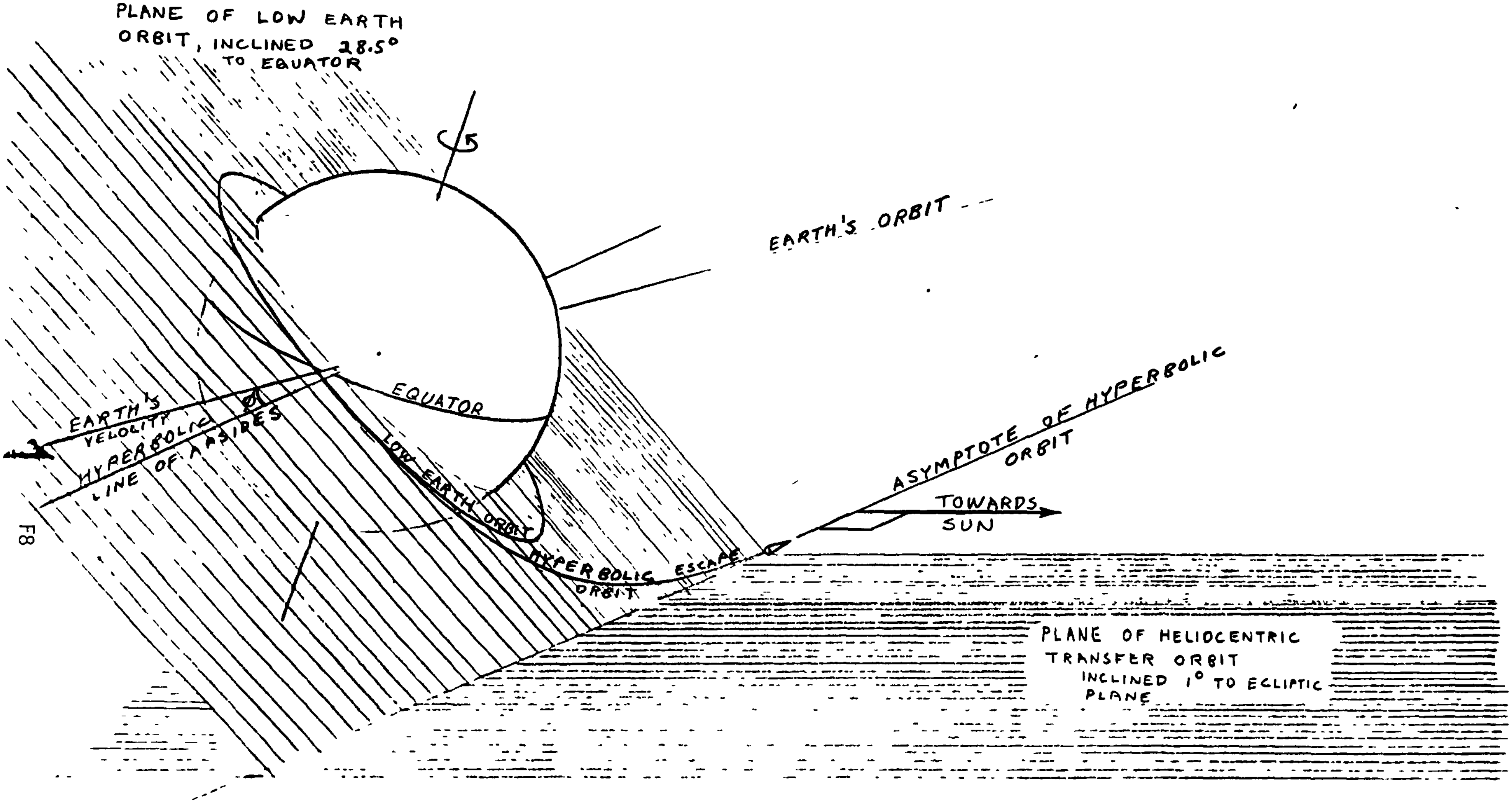


FIGURE 3. NUCLEAR WASTE EARTH ESCAPE ORBIT

maneuver occurs after perigee of the escape orbit. Since the actual time required to stage the SOIS and re-orient the OTV prior to the retro maneuver was not known, a parametric analysis was made of the impact of staging time. For a hyperbolic orbit the time from perigee is given by:

$$t_h = \frac{a^{3/2}}{\sqrt{\mu}} \left[\frac{e\sqrt{e^2-1}\,\sin\theta}{1 + e\cos\theta} - \ln\left(\frac{\sqrt{e+1} + \sqrt{e-1}\,\tan\theta/2}{\sqrt{e+1} - \sqrt{e-1}\,\tan\theta/2} \right) \right] \quad (8)$$

Where θ is the angle from perigee to the current orbital position.

The radius of a hyperbolic orbit can be expressed as a function of θ by:

$$r = \frac{r_p(1+e)}{1 + e\cos\theta}$$

The velocity can be determined from the ~~equation~~ relationship shown previously for hyperbolic orbits, equation (6).

Table 1 presents values of radius, velocity and time from perigee as a function of the angle from perigee.

(degrees)	Radius (km)	Velocity (km/s)	Time from Perigee (sec)
9	6714	10.97	100
19	6861	10.85	200
27	7062	10.70	300
35	7344	10.49	400
43	7722	10.23	500
49	8078	10.01	600
55	8509	9.76	700
61	9026	9.48	800
66	9536	9.23	900
70	10005	9.01	1000
74	10535	8.79	1100
78	11138	8.55	1200

Two different methods of performing the braking maneuver out of hyperbolic orbit were analyzed.

(1) Tangential Δ V, in which the braking velocity vector is applied tangentially to the orbital path.

(2) Co-apsidal Δ V, in which the braking velocity vector is oriented such that there is no shift in the orbit line of apsides.

Two equations are required in addition to equation (6) to fully describe motion along an orbit.

$$\tan\theta = \frac{(rv^2/\mu)\sin\beta\cos\beta}{(rv^2/\mu)\cos^2\beta - 1} \qquad (9)$$

$$e^2 = \left(\frac{rv^2}{\mu} - 1\right)^2 \cos^2\beta + \sin^2\beta \qquad (10)$$

The physical relationship of the variables is shown in Figure 4. For the case of a tangential ΔV applied after perigee there is a rotation in the orbit line of apsides along with the change in semi-major axis, "a", as shown in Figure 5. Equation (9) can be differentiated holding β constant and simplified.

$$d\theta = -\frac{2\sin\theta}{e}\frac{dV}{V} \qquad (11)$$

Integrating gives the relationship for the change in θ due to a tangential ΔV.

$$\frac{\tan\theta_2/2}{\tan\theta_1/2} = \left(\frac{V_2}{V_1}\right)^{-2/e} \qquad (12)$$

where $V_2 = V_1 - \Delta V$ and V_1 is the velocity prior to retro. The value of β can be determined in equation (10) from the original orbit values of e, r and v. Inserting the new velocity V_2 gives the eccentricity of the new orbit, e_2. The new value of semi-major axis, "a_2", can be determined from V_2 and r using the following expression:

$$\frac{a}{r} = \frac{1}{2-(rv^2/\mu)} \qquad (13)$$

The new apogee and perigee radii can then be calculated using equation (5) and the velocites ~~from~~ using equation (1). A computer code was developed to run through this sequence of calculations and determine the total ΔV to return to circular LEO for a range of times from perigee and braking ΔV's. The results are plotted in Figure 6. It indicates that total return ΔV is sensitive to the size of the initial retro ΔV, extremely so for the longer times from perigee. For a recoverable OTV it is important to complete staging and re-orientation as soon as possible after injection into the escape orbit. This may be a problem for SOIS that must be spun up and oriented prior to separation.

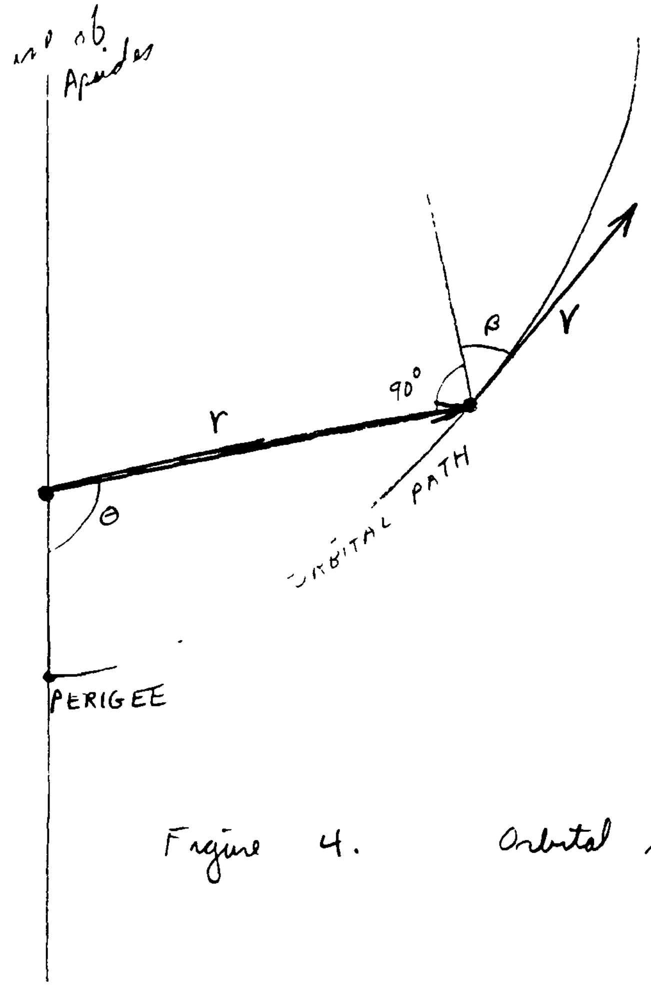


Figure 4. Orbital variables

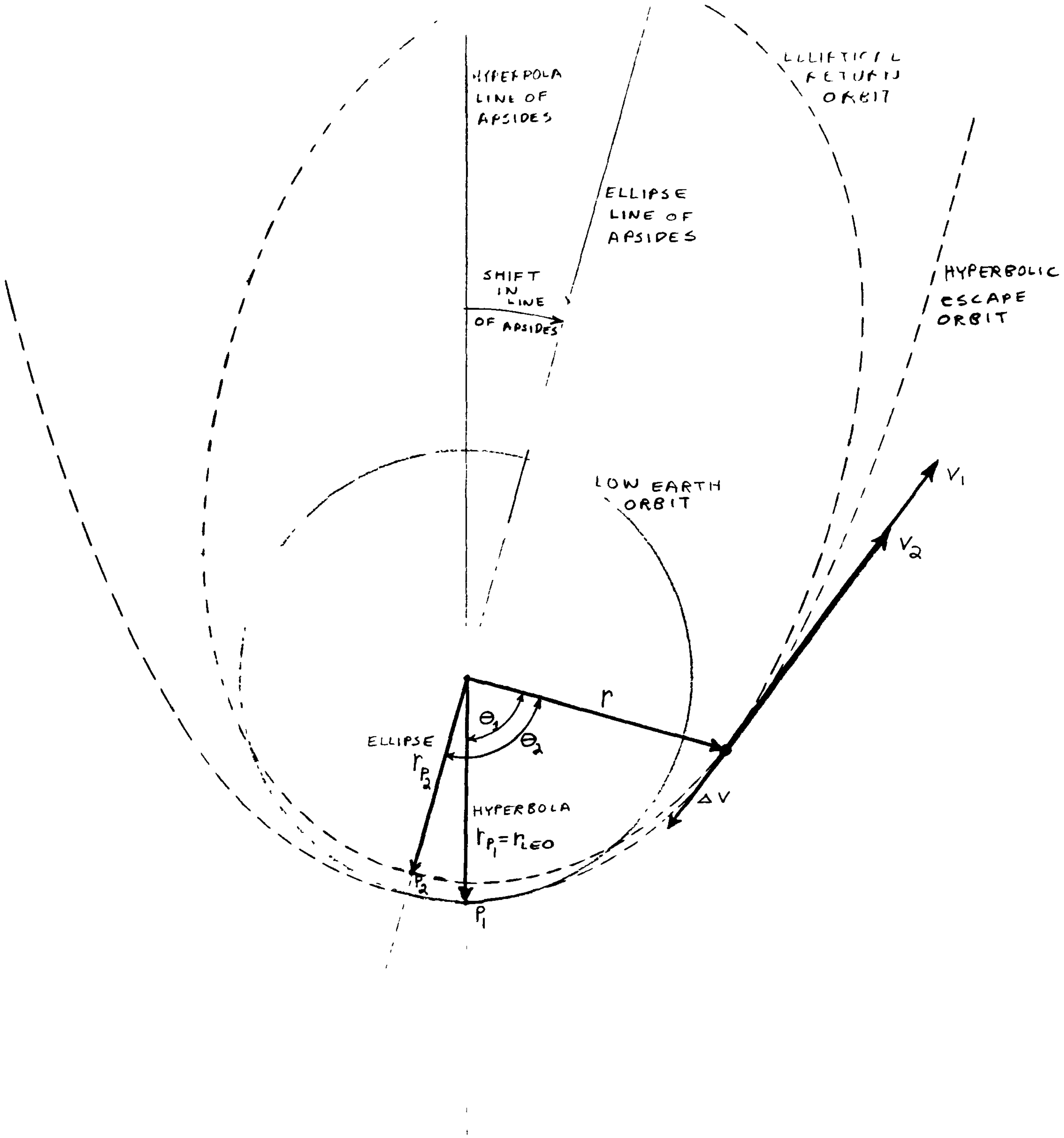


FIGURE 5. TANGENTIAL ΔV BRAKING MANEVER

A more useful parameter against which total return ΔV can be evaluated is period of the return orbit. This can be determined from the semi-major axis using equation (2). Figure 7 is a plot of total return ΔV as a function of return orbit period and time from perigee. For short staging times there is only a slight increase in ΔV to achieve orbit periods consistent with proposed OTV mission times of about 60 hours.

The co-apsidal retro maneuver requires that $\theta_2 = \theta_1$. Figure 8 illustrates the co-apsidal retro maneuver. Re-arranging equation (9):

$$\frac{\mu}{r V^2} = \cos^2\beta - \frac{\sin\beta\cos\beta}{\tan\theta} \qquad (14)$$

Since r and θ are constant for the maneuver, β_2 can be determined for any V_2. In this case V_2 and V_1 do not have the same direction. The required ΔV to change from V_1 to V_2 is given by:

$$\Delta V = \sqrt{V_1^2 + V_2^2 - 2V_1V_2\cos(\beta_1 - \beta_2)}$$

The remainder of the necessary orbit variables can be calculated in the same manner as described previously for the tangential ΔV case. A computer code similar to the Tangential ΔV code was developed to run through these calculations for various staging times and initial retro ΔV's. Figure 9 is a plot of total return ΔV as a function of orbit period and staging time. The relationship is similar to the Tangential ΔV case, but with greater ΔV required, as shown in the comparison of the two cases in Figure 10. The increase in total ΔV is due primarily to the cosine loss incurred by applying the retro ΔV at some angle ($\beta_2-\beta_1$) to the original velocity.

Based on these results the tangential retro ΔV was selected for the nuclear waste mission. Because no plane change is involved in either the escape or retro maneuvers the shift in line of apsides caused by Tangential ΔV is not a problem. An intermediate orbit can be used to phase back into LEO.

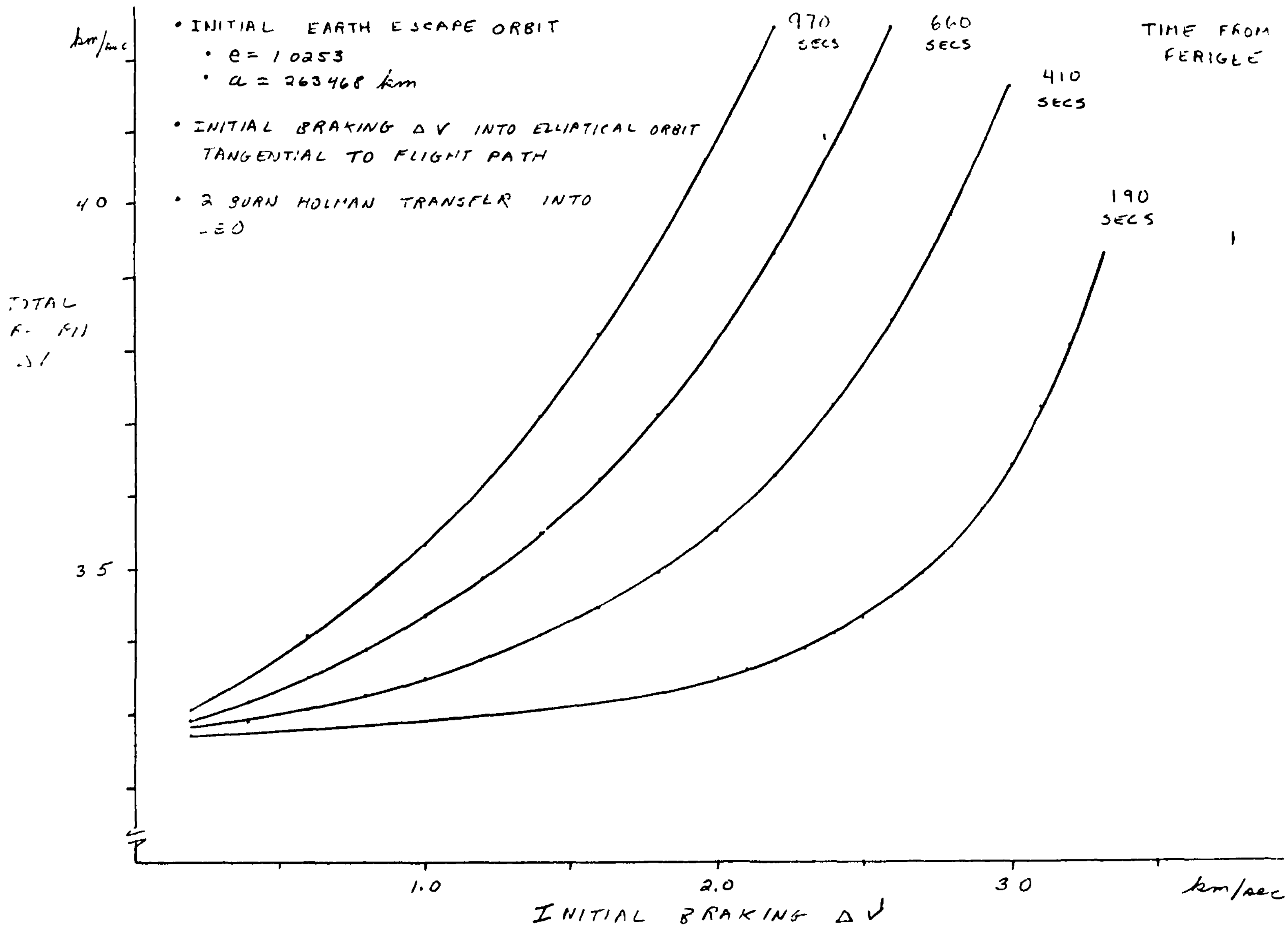


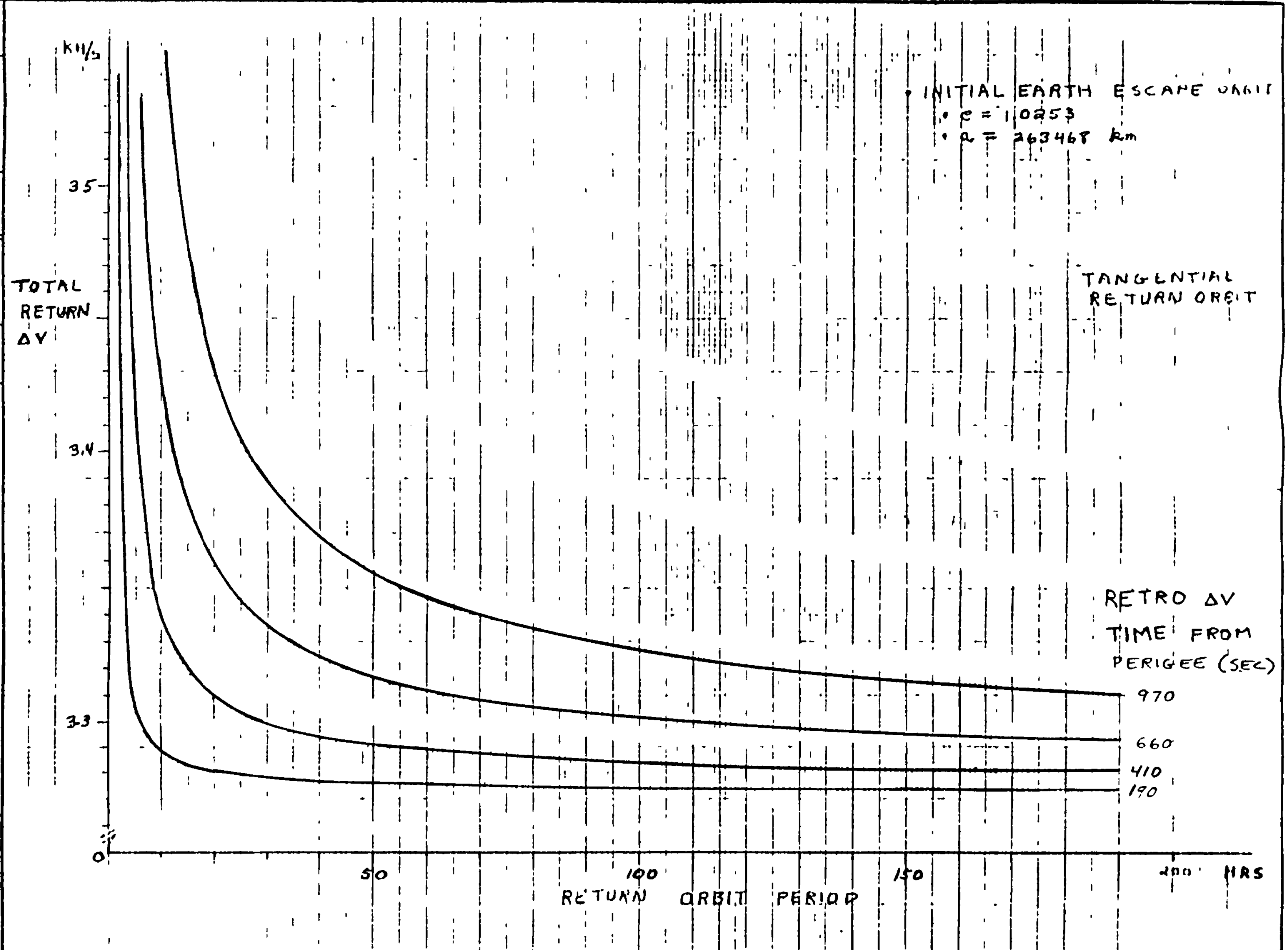
Figure 6. Tangential Retro ΔV, Total Return ΔV vs Initial Braking ΔV & ~~Bo Stg~~ Staging Time

ENGR			REVISED	DATE
CHECK				
APR				
APR				

F14 BOEING

6a

FIGURE 7. Tangential Retro ΔV, Total Return ΔV vs Orbit Period & Staging Time



CALC CHECK APR APR REVISED DATE

F15 THE BOEING COMPANY

PAGE

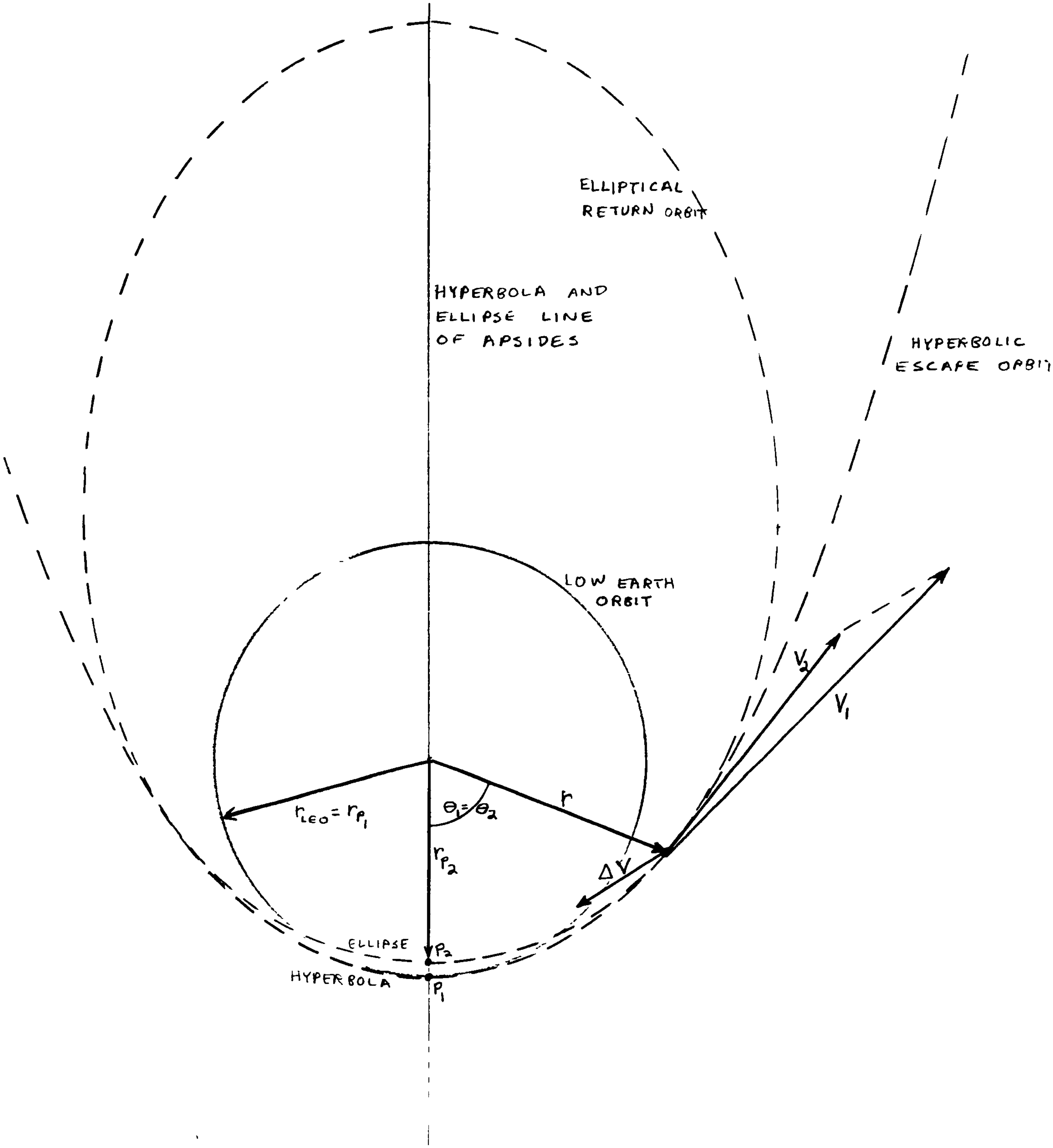
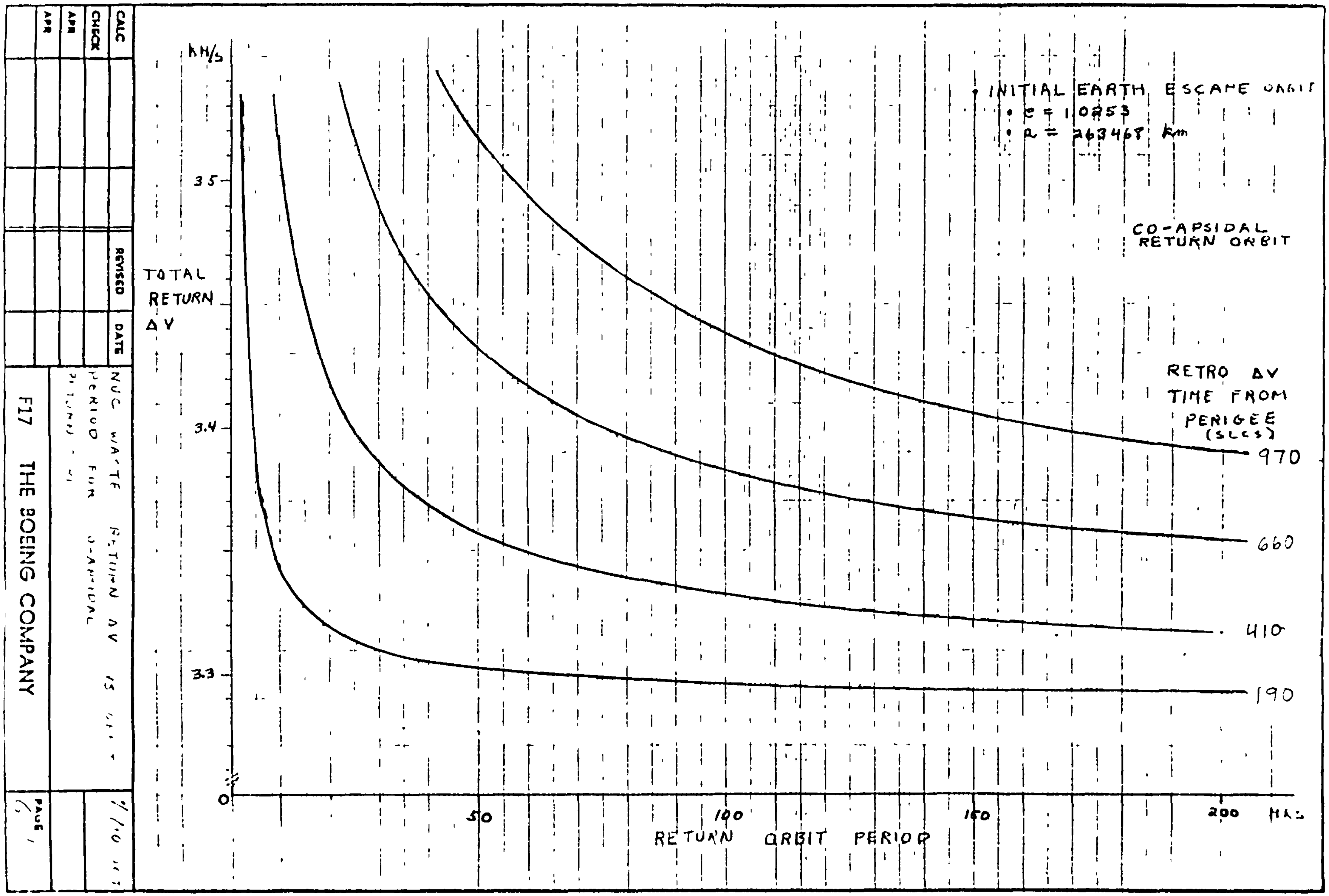


FIGURE 2 CO-APSIDAL ΔV BRAKING MANEUVER

1, C

FIGURE 9 Co-Apsidal Retro ΔV - Total Return Δ.
vs Orbit Period & Staging Time



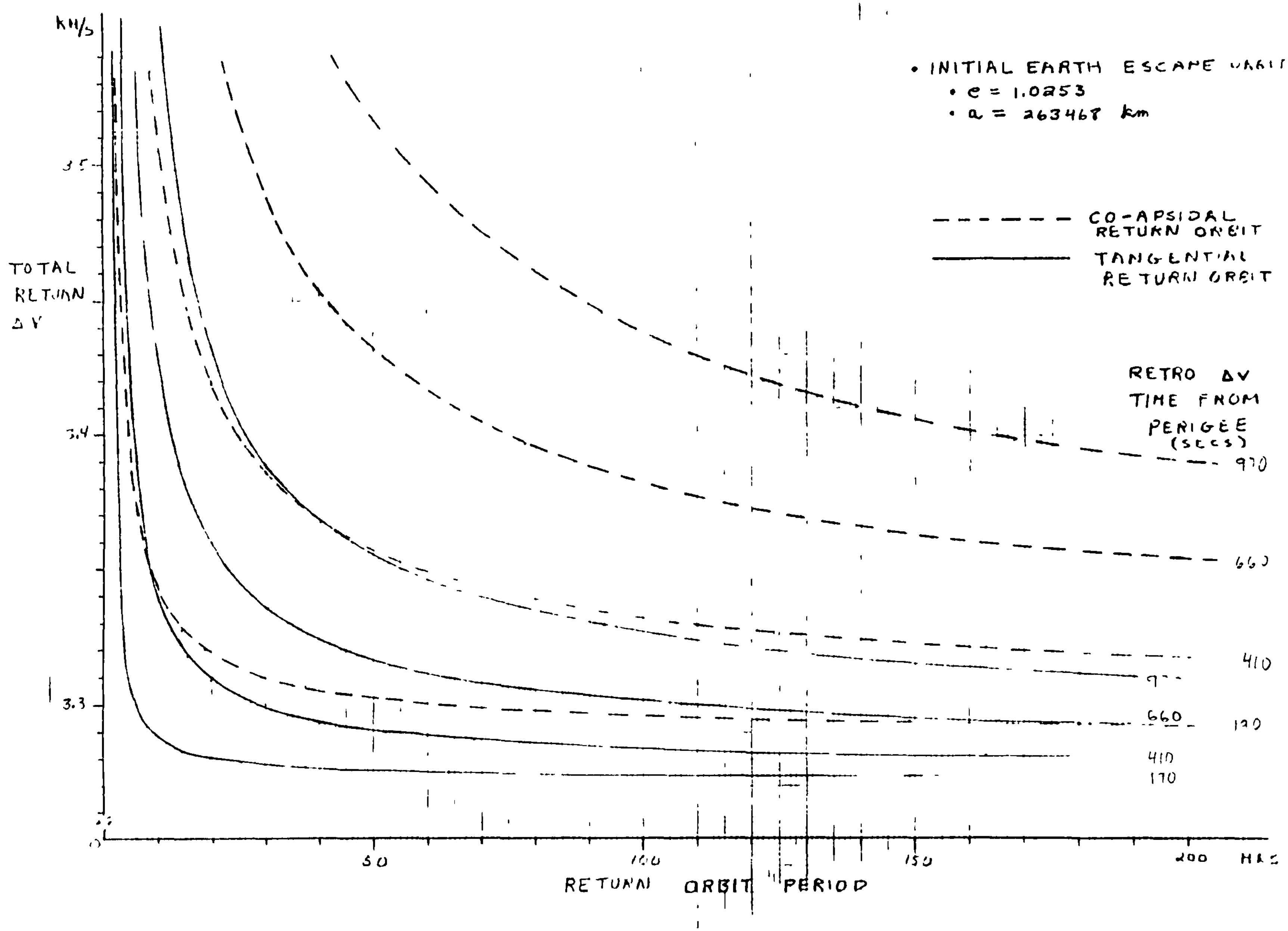


Figure 10 Comparison of Total Return ΔV for Tangential and Co-Apsidal Retro ΔV's.

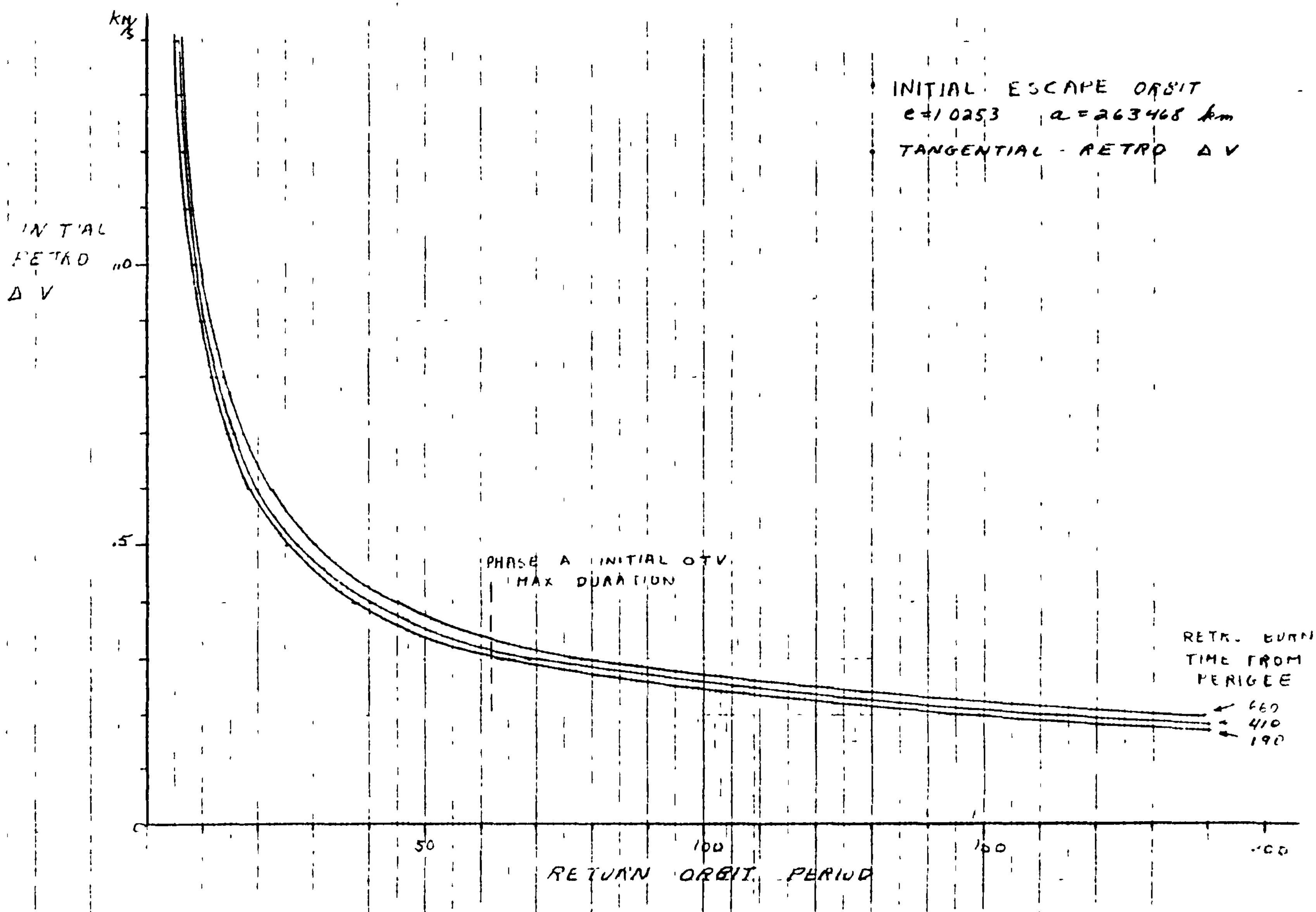
			REVISED	DATE		
CALC					[illegible] RETURN ΔV VS ORBIT PERIOD FOR CO-APSIDAL & TANGENTIAL RETURN ORBITS	7/11/86 JCJ
CHECK						
APR						
APR					F18 THE BOEING COMPANY	PAGE 6e

Aerobraking technology was assumed to be already developed in the time frame of interest in the Nuclear Waste study. Since an aerobraking OTV has much better performance than an all propulsive OTV, this mode was selected for the reference mission. The impact of staging time on total return Δ V is lessened when an aerobraked OTV is used. Aerobraking system weights are essentially independant of the magnitude of velocity reduction in the aerobraking maneuver for the range of velocities of interest here. As a result the initial retro Δ V essentially becomes the total return Δ V. (The Δ V at apogee of the ellipse is $\approx$10 m/s). The plot of initial retro Δ V as a function of return orbit period in Figure 11 shows the effect of staging time reduced to 10-50 m/sec. It also indicates that the Phase A OTV mission duration is appropriate for this mission.

The mission profile shown in Figure 12 was developed for an aerobraked OTV with an initial retro Δ V of .4 km/sec and a staging time of 410 seconds after perigee. The mission timeline and Δ V's are listed in Table 2 and a mission sequential mass statement for a LO2/LH2 OTV and a storable SOIS is shown in Figure 13.

A series of parametric performance curves for different OTV/SOIS combinations were developed to support the Nuclear Waste study. Figure 14 shows the performance capability as a function of OTV propellant capacity. It indicates the excellent performance capability of a reuseable OTV for earth escape missions. The method developed here and the computer codes are ideally suited to general analysis of earth escape or planetary missions involving reusable OTV's.

Figure 11. Initial Retro ΔV vs Return Orbit Period and Time from Perigee



CALC			REVISED	DATE	NUC WASTE INITIAL RETRO ΔV
CHECK					
APR					
APR					

F20 THE BOEING COMPANY

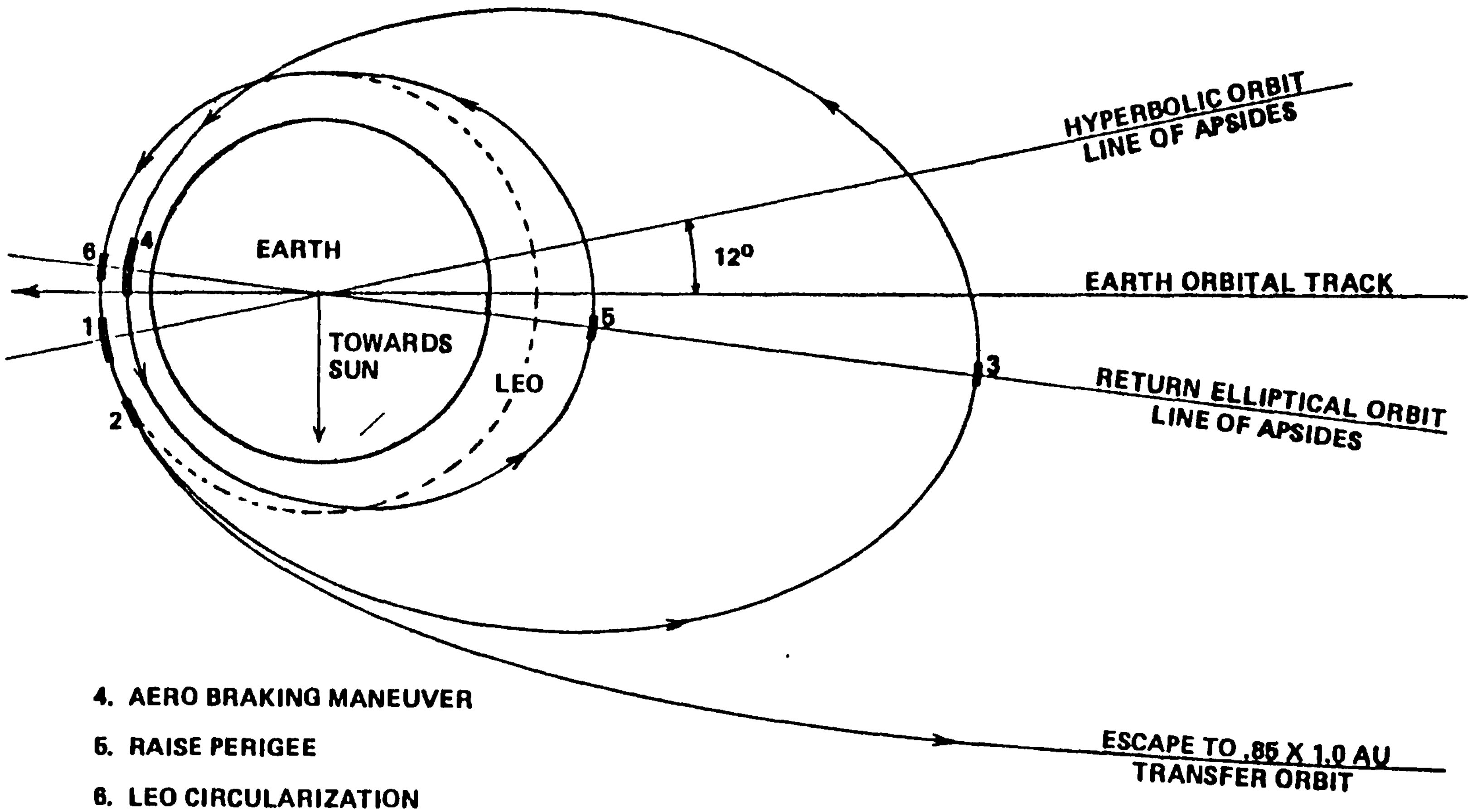
PAGE 7a

Figure 12.

OTV INJECTION MISSION PROFILE

NWC⁻ 104

1. INJECTION INTO EARTH ESCAPE
2. SOIS STAGING AND OTV RETRO
3. ADJUST PERIGEE



4. AERO BRAKING MANEUVER
5. RAISE PERIGEE
6. LEO CIRCULARIZATION

F21

[illegible]

	EVENT	ΔV		TIME	
	[illegible]			0	
1	SEPARATE	10	(~~RCS~~)	7.061	
	PHASE	—		11	
	ESCAPE INJECT	10,743		.25	
4	STAGE	10	(~~RCS~~)	.17	
	~~COAST~~				
	RETRO	V_1 1312		.1	
	COAST	—		~~72~~	20.08
	TRANS INJECT	~~V~~ 10	(~~RCS~~)	.01	
	COAST	35	(~~RCS~~)	~~12~~	20 08
	LEO CIRC	~~V~~ 9486		.05	
	TRIM	10	(~~RCS~~)	.05	
	DOCK	—		4	
	RESERVES	430		—	

S015

0	STAGE	—	
1	ORIENT	5	.2
2	COAST	30	3899
3	ORIENT	5	.2
4	LEO CIRC	4209	.2
5	RESERVES	85	—

Table 2 Mission Timeline and ΔV Budget

26 M1=12279 Figure 13
RUN
JNOTV2 11:05 AM 02-Dec-80

USABLE MAIN PROP. MASS 130000

'OMINAL BURNOUT MASS 12279

START MISSION MASS 249750

INJECTED MASS 105300

SOIS PROP. MASS 39665.4

SOIS BURNOUT MASS 6981.02

PAYLOAD MASS 58653.6

DO YOU WANT DETAILS(YES OR NO)? YES
MAIN ENG. ISP = 464.6
AUX. PROP. ISP = 220

EVENT	DELTA V	PROP. USAGE	LOSSES	MASS
STARTMISSION				249750
SEPERATE	10	352.6	13.9	249384
PHASE	0	0	21.7	249362
ESCAPE INJECT	10743	127825	48.5	121488
STAGING	10	171.5	.3	16016.3
RETRO INJECT	1312	1345.8	19.2	14651.2
COAST	0	0	39.5	14611.7
TRANS. INJECT	10	20.6	0	14591.1
'OAST	65	68.7	58.5	14463.9
.EROMANEUVER	0	0	105.642	14358.3
COAST	0	0	1206.34	13151.9
PHASE INJECT	220	192.1	19.1	12940.7
COAST	0	0	5.9	12934.8
LEO CIRC.	400	341.5	19.1	12574.1
TRIM	10	17.8	.1	12556.3
DOCK	0	0	7.9	12548.4
RESERVES	250	208.1	54.4733	12285.8

NOMINAL MAIN PROPELLANT = 129999

RESERVE MAIN PROPELLANT = 208.1

NOMINAL AUX. PROPELLANT = 544.733

RESERVE AUX. PROPELLANT = 54.4733

TOTAL LOSSES = 360.842

$\frac{\partial GW}{\partial P/L} = 3.9829$

$GW = 16141 + 3.9829 * P/L$

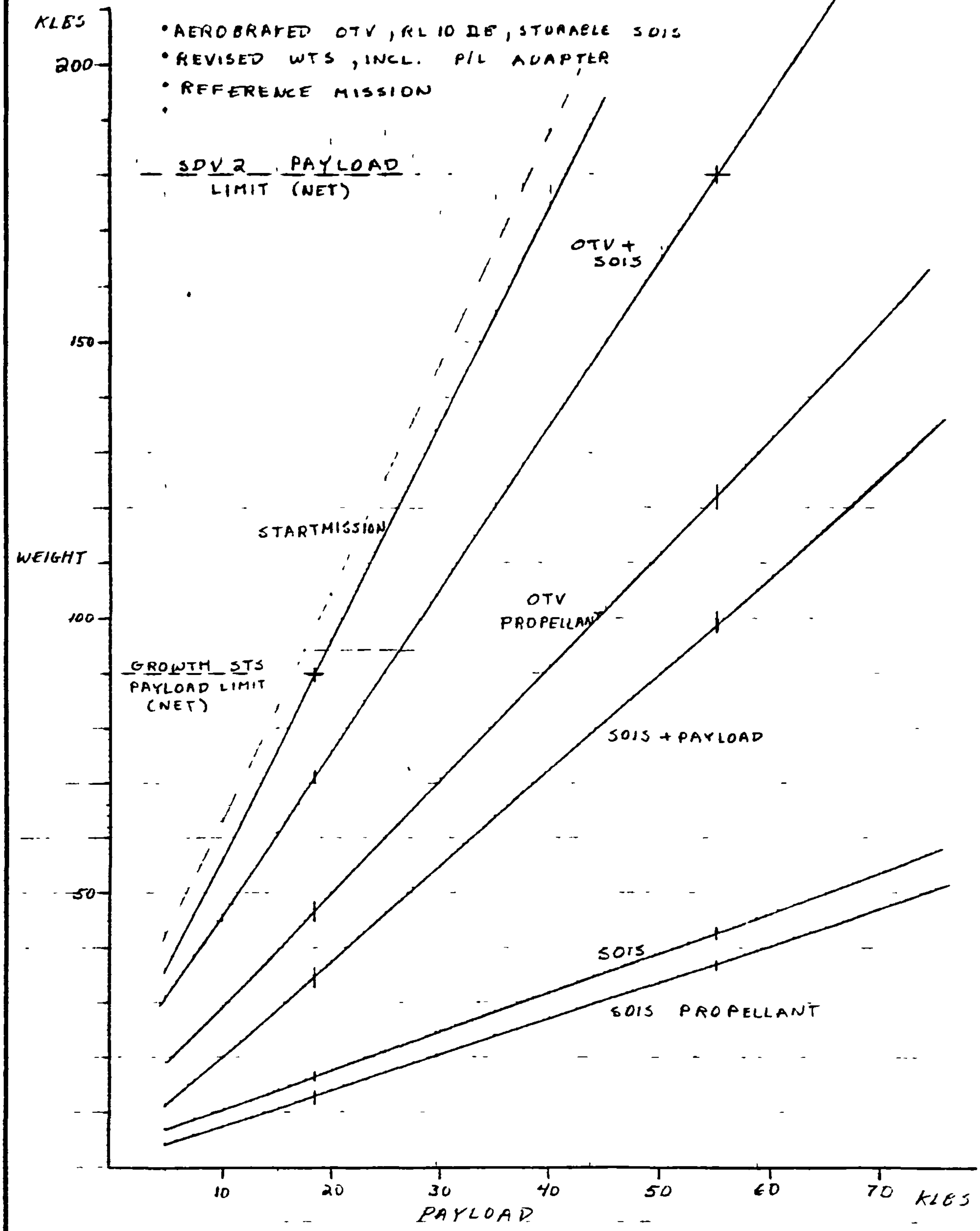
SOIS MASS SEQUENCE

$\frac{\partial \text{SOIS (INCL P/L ADAPT)}}{\partial P/L} = .7487$

$SOIS = 2732 + .7487 * P/L$

EVENT	DELTA V	PROP. USAGE	MASS
'AGING			105300
ORIENT	5	60.6	105239
COAST	30	343.7	104896
ORIENT	5	60.4	104835
HELIO CIRC.	4209	38589.4	66245.9
RESERVES	85	611.2	65634.6

Figure 14



CALC	JCJ	12/...	REVISED	DATE	NUC WASTE PARAMETRIC PERFORMANCE AERO BRAKED OTV, STORABLE SOIS REVISED WTS	
CHECK						
APR						
APR					F24 THE BOEING COMPANY	PAGE

115-047

APPENDIX G

ORBIT TRANSFER SYSTEM DETAIL MASS STATEMENTS

N.T. C[illegible]
8/25/80

GROUP WEIGHT STATEMENT
SOLAR ORBIT INSERTION STAGE
USABLE PROP. (MAIN + RCS) = 7311 LB

STRUCTURE – EXCLUDING TANKAGE		720
CIRCULAR SKIN PANEL – INNER	72	
CIRCULAR SKIN PANEL – OUTER	172	
RADIAL SKIN PANELS (6)	97	
LOWER DECK SKIN PANEL	81	
UPPER DECK DIAGONAL BRACES (12)	50	
INNER RINGS, UPPER AND LOWER (2)	24	
OUTER RINGS, UPPER AND LOWER (2)	60	
CHORDS, UPPER AND LOWER – RADIAL PANELS (12)	34	
MOUNTING/SUPPORT STRUCTURES	50	
INTERFACE RING, SOIC-TO-OTV P/L ADAPTER	60	
ASSY. & INSTALLATION HARDWARE	20	
THERMAL CONTROL		20
AVIONICS		130 ~~100~~ 120
POWER SUPPLY & DISTRIBUTION		~~60~~ [illegible]
SOLAR ARRAY (150 FT²) ▷	30	
POWER CONTROL/DISTRIBUTION	30	
MAIN PROPULSION – INCLUDING TANKAGE		520
NTO TANKS (3)	165	
MMH TANKS (3)	165	
HELIUM BOTTLES (3)	90	
ENGINE – 100 LB FORCE, AJ10-160 (6)	48	
PROPELLANT SYSTEM	40	
PRESSURANT SYSTEM	12	
ATTITUDE CONTROL ▷		60
THRUSTER – 25 LB FORCE, ORBITER RCS VERNIER (12)	30	
PROPELLANT SYSTEM	30	
WEIGHT GROWTH ~15%		7 ~~220~~ [illegible]
DRY WEIGHT		~~1700~~ LB 2084

(CONT'D)

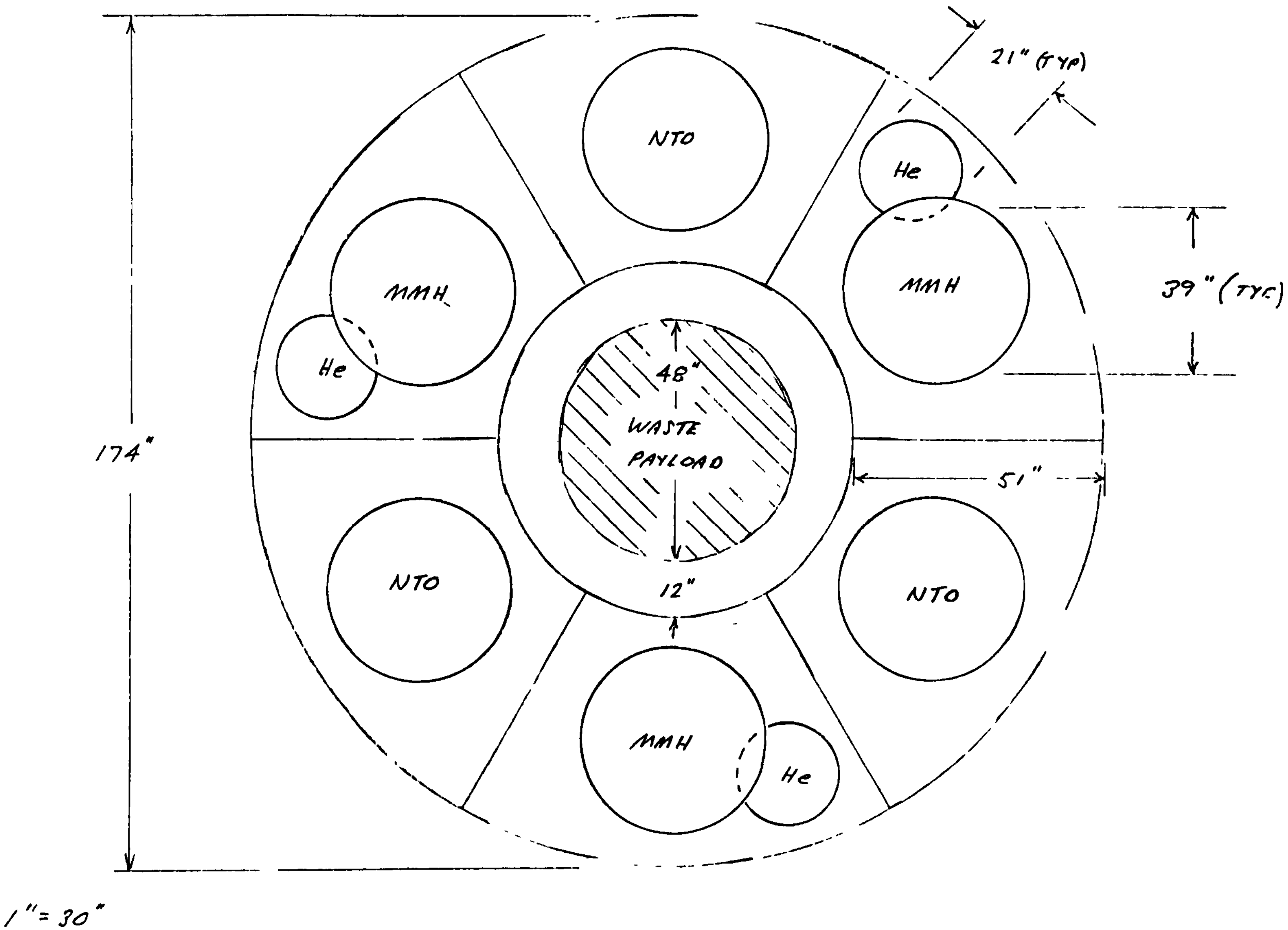
▷ 8-MIL CELLS, 6-MIL COVERS, 2-MIL SUBSTRATE & INTERCONNECTS (~0.2 LB/FT²)
▷ NTO/MMH SYSTEM. PROPELLANT DRAWN FROM MAIN TANKS.

GROUP WEIGHT STATEMENT
SOLAR ORBIT INSERTION STAGE
USABLE PROP(MAIN+RCS) = 7311 LB

RESIDUAL FLUIDS & GASES		322
NTO TRAPPED IN TANKS ~3%	141	
MMH TRAPPED IN TANKS ~3%	87	
NTO TRAPPED IN LINES	47	
MMH TRAPPED IN LINES	29	
HELIUM	18	
BURNOUT WEIGHT - LESS RESERVES		~~2022~~ 2406 LB
USABLE PROP - INCLUDING RESERVES (MAIN + ACS)		7311
STARTBURN WEIGHT		~~9333~~ 9717 LB

$$\text{MASS FRACTION} = \frac{\text{USABLE PROP. WT}}{\text{STARTBURN WT}} = 0.783$$

▷ MAINSTAGE + ACS, MR = 1.616 : 1



TANKAGE ARRANGEMENT
USABLE PROP = 7311 LB

GROUP WEIGHT STATEMENT
IAPOTV
REUSABLE DELIVERY MISSION TO GEO

STRUCTURE - JETTISONABLE BALLUTE (INCL 15% MARGIN)			(– – –)
STRUCTURE - BASIC VEHICLE			(2464)
LH2 TANKS & SUPPORTS		507	
LO2 TANKS & SUPPORTS		416	
BODY SHELL & OTHER STRUCTURES		1441	
SANDWICH PANELS - FWD OF AVIONICS SECTION	362		
SANDWICH PANELS - AFT OF AVIONICS SECTION	128		
FWD RING (P/L ADAPTER LOWER RING)	50		
MAIN SUPPORT RING, FWD YZ LOADS	178		
LH2 TANK SUPPORT RING	35		
LO2 TANK SUPPORT RING	60		
AFT RING (BERTHING ADAPTER UPPER RING)	68 ▷		
AVIONICS RING SECTION ASSEMBLY	300		
MISC. MOUNTING/SUPT. STRUCTURES - AVIONICS/EQUIP.	25		
MAIN SUPPORT FITTINGS, FWD YZ LOADS (3)(Ti)	90		
RMS GRAPPLE FITTINGS (2)(AL)	20		
THRUST STRUCTURE	30		
BERTHING PROBES	20		
UMBILICAL INTERFACE, OTV TO ASE	25		
ASSEMBLY & INSTALLATION HARDWARE	50		
PAYLOAD/DOCKING ADAPTERS		100	
P/L ADAPTER MECHANISMS - LATCH/RELEASE/DEPLOY	100		
UNIVERSAL DOCKING ADAPTER - OTV SIDE	– –		
BALLUTE INSTALLATION FIXED HARDWARE		– –	
THERMAL CONTROL			(315)
ACTIVE THERMAL CONTROL		60	
RADIATOR SYSTEM - FUEL CELLS	60		
INSULATION		255	
CRYO INSULATION - LH2 TANK	90		
CRYO INSULATION - LO2 TANK	45		
CRYO INSULATION - EPS TANKS	2		
PURGE CONTAINMENT - LH2 TANK	33		
PURGE CONTAINMENT - LO2 TANK	23		
PLUMBING - PURGE/REPRESS	52		
MISC.	10		

▷ INCORPORATES 56 LATCH RECEPTACLE FITTINGS @ 0.5 LB EACH

GROUP WEIGHT STATEMENT
IAPOTV
REUSABLE DELIVERY MISSION TO GEO

AVIONICS		(649)
GUIDANCE & NAVIGATION		99
BASIC G&N	99	
GPS	--	
COMMUNICATIONS		95
DATA MANAGEMENT		313
INSTRUMENTATION		142
PROPELLANT LOADING/MEASUREMENT	50	
OTV MEASURING SYSTEM	92	
ELECTRICAL POWER SYSTEM (EPS)		(634)
POWER SOURCE		264
FUEL CELLS (2)	160	
BATTERY (25 AMP-HR)	28	
O2 TANK ASSEMBLY	10	
H2 TANK ASSEMBLY	6	
PLUMBING SYSTEMS	60	
CONVERSION & DISTRIBUTION		370
ELECTRONICS	156	
WIRE HARNESS	214	
MAIN PROPULSION SYSTEM (MPS)		(937)
MAIN ENGINE		404
PROPELLANT SYSTEM		244
LH2 FEED	57	
LH2 FILL, DRAIN, DUMP	82	
LO2 FEED	32	
LO2 FILL, DRAIN, DUMP	73	
THRUST VECTOR CONTROL - INCL ACTUATOR SUPT, LINKAGE		50
PRESSURIZATION & VENT SYSTEM - INCL PNEUMATICS		239
LH2 TANK AUTOGENOUS PRESS SUPPLY	25	
LO2 TANK AUTOGENOUS PRESS SUPPLY	20	
LH2 TANK VENT/RELIEF - GROUND/ASCENT	52	
LH2 TANK VENT/RELIEF - SPACE	30	
LO2 TANK VENT/RELIEF - GROUND/ASCENT	14	
LO2 TANK VENT/RELIEF - SPACE	30	
PNEUMATIC SYSTEM	68	

GROUP WEIGHT STATEMENT
IAPOTV
REUSABLE DELIVERY MISSION TO GEO

ATTITUDE CONTROL SYSTEM (ACS)		(138)
THRUSTERS (6 MODULES / 12 THRUSTERS)		32
TANKAGE / PROPELLANT SYSTEM		106
N2H4 TANK ASSEMBLIES (3)	81	
N2H4 FEED	25	
WEIGHT GROWTH MARGIN - EXCL BALLUTE MARGIN		(633)
MARGIN ON NEW HARDWARE (15%)		565
MARGIN ON EXISTING/PROTOTYPE HARDWARE (5%)		68
OTV DRY WEIGHT		5770 LB
MPS RESIDUAL FLUIDS & GASES		(453)
TRAPPED LH2		14
TRAPPED LO2		134
LH2 FUEL BIAS		62
GH2 IN EMPTY LH2 TANK		145
GO2 IN EMPTY LO2 TANK		223
GHe FOR PNEUMATICS		6
MISSION REQUIREMENT - INCL RESERVE	6	
RESERVE MARGIN - FULL TANK	0	
ACS RESIDUAL FLUIDS & GASES		(11)
TRAPPED N2H4		8
GN2		3
EPS RESIDUAL FLUIDS & GASES		(12)
TRAPPED O2		4
TRAPPED H2		1
TRAPPED PRODUCT WATER		7
MPS RESERVE PROPELLANT - FPR.		(258)
ACS RESERVE PROPELLANT		(22)
MISSION REQUIREMENT		22
RESERVE MARGIN - FULL TANKS		-- ▷
EPS RESERVE REACTANT		(15)
MISSION REQUIREMENT		15
RESERVE MARGIN - FULL TANKS		0

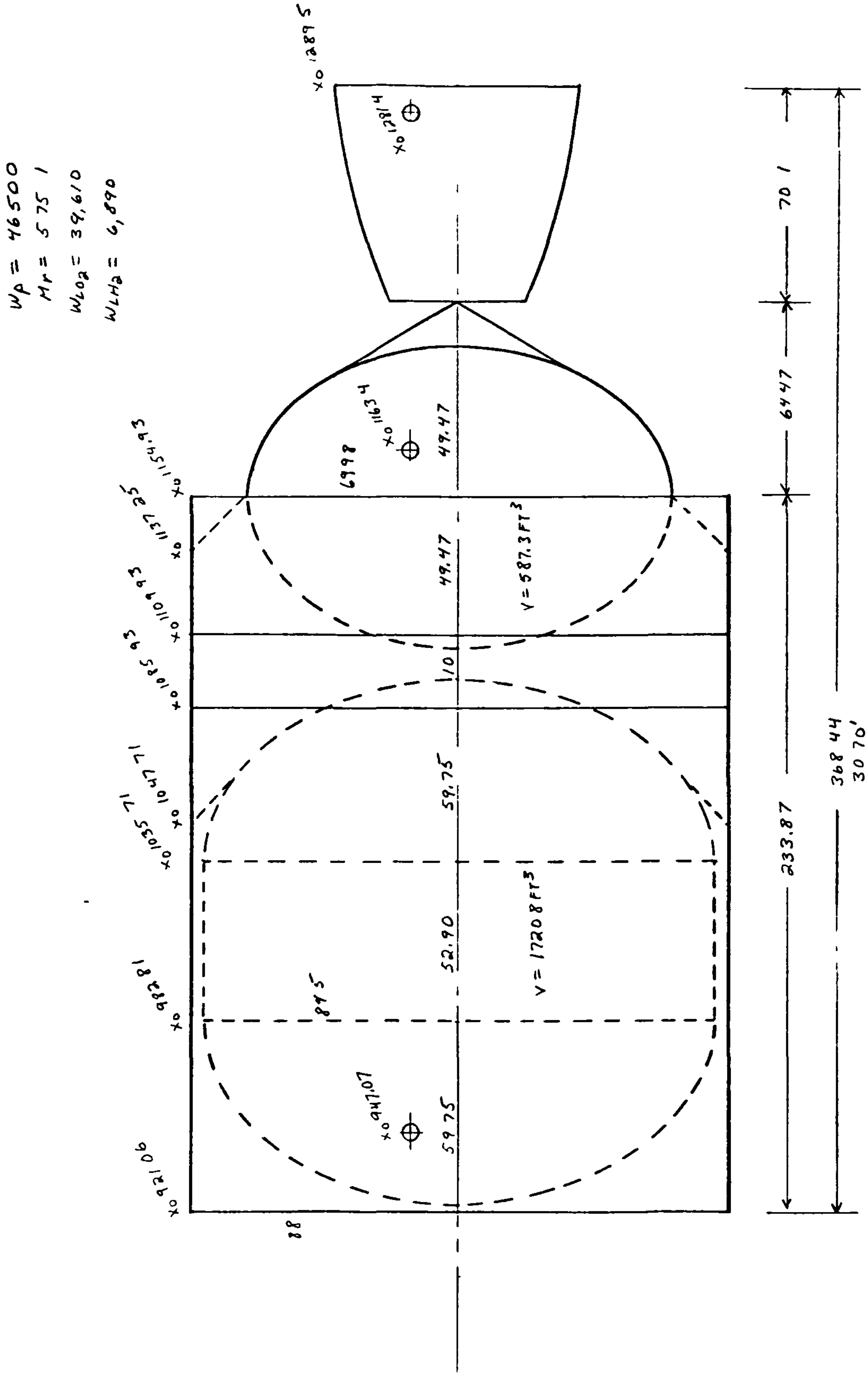
▷ CAN LOAD ADDITIONAL 115 LB BASED ON USE OF 3 IUS TANKS, EACH TANK HAVING A TOTAL USABLE N2H4 CAPACITY OF 120 LB.

GROUP WEIGHT STATEMENT
IAPOTV
REUSABLE DELIVERY MISSION TO GEO

MPS INFLIGHT LOSSES	(181)
LH2 FOR ENGINE CHILLDOWN/START/STOP	70
LO2 FOR ENGINE CHILLDOWN/START/STOP	111
LH2 BOILOFF/VENTING	--
LO2 BOILOFF/VENTING	--
LH2 EXPENDED DURING AEROBRAKING	--
LO2 EXPENDED DURING AEROBRAKING	--
ACS INFLIGHT LOSSES	(--)
N2H4 FOR MAIN PROPELLANT SETTLING	--
MPS NOMINAL PROPELLANT	(46,247)*
ACS NOMINAL PROPELLANT	(223)
EPS NOMINAL PROPELLANT	(73)
OTV GROSS WEIGHT	53,265 LB

$$\text{OTV MASS FRACTION} = \frac{\text{MPS PROPELLANT, NOMINAL + FPR}}{\text{GROSS WEIGHT}} = 0.8731$$

* O/F = 5.75:1



ENGR			REVISED	DATE	IAPOTV RESIZED FOR	JCJ 6/26/80
CHECK					65K STS $w_p = 46500$ @5.75:1	FROM 4/18/80
APR						
APR					G9 BOEING	

STC ... 10
11/30/80

SUMMARY WEIGHT STATEMENT
NWD AEROBRAKED OTV, W_P = 132.8K

STRUCTURE – JETTISONABLE BALLUTE (INCL MARGIN)	1143
STRUCTURE – BASIC VEHICLE	5840
THERMAL CONTROL	480
AVIONICS	793
ELECTRICAL POWER SYSTEM (EPS)	694
MAIN PROPULSION SYSTEM (MPS)	1701
ATTITUDE CONTROL SYSTEM (ACS)	266
SPACE MAINTENANCE PROVISIONS	--
WEIGHT GROWTH MARGIN	1273
(OTV DRY WEIGHT)	(12,190 LB)
MPS RESIDUAL FLUIDS & GASES	1380
ACS RESIDUAL FLUIDS & GASES	17
EPS RESIDUAL FLUIDS & GASES	12
MPS RESERVE PROPELLANT	197
ACS RESERVE PROPELLANT	57
EPS RESERVE PROPELLANT	18
MPS INFLIGHT LOSSES	224
MPS NOMINAL PROPELLANT	132,577 *
ACS NOMINAL PROPELLANT	572
EPS NOMINAL REACTANT	70
(OTV GROSS WEIGHT)	(147,314 LB)

* O/F = 5.75:1

$$\text{OTV MASS FRACTION} = \frac{\text{MPS PROP, NOM + RESERVE}}{\text{GROSS WEIGHT}} = 0.9013$$

END OF MISSION REFERENCE WT = 12,190 − 1143 + (1380 + 17 + 12) + 18 = 12,474 LB

A T C[illegible]
11/5/80

GROUP WEIGHT STATEMENT
NWD AEROBRAKED OTV, W_p = 132.8 K

STRUCTURE – JETTISONABLE BALLUTE (INCL 15% MARGIN)		(1143)
STRUCTURE – BASIC VEHICLE		(5840)
LH2 TANK – INCL SUPPORT STRUTS & ATTACH BRACKETRY		1350
LO2 TANK – INCL SUPPORT STRUTS & ATTACH BRACKETRY		1030
BODY STRUCTURES		3260
FORWARD TRUSS	--	
INTERTANK TRUSS	--	
SANDWICH PANELS (2193 FT², 18 MIL FACE SHTS)	1795	
FORWARD RING (LATCHING INTERFACE)	50	
MAIN SUPPORT RING, FWD YZ LOADS	270	
LH2 TANK SUPPORT RINGS (2)	90	
LO2 TANK SUPPORT RINGS (2)	130	
AFT RING (LATCHING INTERFACE)	70	
AVIONICS RING SECTION	300	
MISC. MOUNTING/SUPPORT STRUCTURES	30	
MAIN SUPPORT FITTINGS, FWD YZ LOADS	150	
RMS GRAPPLE FITTING (2)	20	
THRUST STRUCTURE (BODY MOUNTED CONE, NO BASE RING)	210	
BERTHING PROBES – EXTENDABLE	40	
UMBILICAL INTERFACE, OTV-TO-ASE	25	
ASSEMBLY & INSTALLATION HARDWARE	80	
BALLUTE INSTALLATION FIXED HARDWARE		100
LATCHING MECHANISM – FWD INTERFACE		100
UNIVERSAL DOCKING ADAPTER		--
THERMAL CONTROL		(480)
ACTIVE THERMAL CONTROL		60
AVIONICS COOLING	--	
FUEL CELL COOLING	60	
INSULATION		278
MLI INSULATION – LH2 TANK (23 LAYERS)	210	
MLI INSULATION – LO2 TANK (15 LAYERS)	66	
MLI INSULATION – FWD TRUSS	--	
MLI INSULATION – INTERTANK TRUSS	--	
MLI INSULATION – EPS TANKS	2	
MLI PURGE/REPRESS		122
PURGE CONTAINMENT – LH2 TANK	33	
PURGE CONTAINMENT – LO2 TANK	33	
PLUMBING – PURGE/REPRESS	56	
MISC. THERMAL CONTROL		20

GROUP WEIGHT STATEMENT

NWO AEROBRAKED OTV, W_p = 132.8K

AVIONICS		(793)
GUIDANCE & NAVIGATION		159
BASIC G&N	99	
GPS	60	
COMMUNICATIONS		95
DATA MANAGEMENT		349
RENDEZVOUS & DOCKING		--
INSTRUMENTATION		190
PROPELLANT LOADING/MEASUREMENT	90	
OTV MEASURING SYSTEM	100	
ELECTRICAL POWER SYSTEM (EPS)		(694)
POWER SOURCE		264
FUEL CELLS (2)	160	
BATTERY (25 AMP-HR)	28	
O2 TANK ASSY	10	
H2 TANK ASSY	6	
PLUMBING SYSTEMS	60	
CONVERSION & DISTRIBUTION		430
ELECTRONICS	156	
WIRE HARNESS	274	
MAIN PROPULSION SYSTEM (MPS)		(1701)
MAIN ENGINE (2 RL10-IIB)		808
PROPELLANT SYSTEM		286
LH2 FEED	110	
LH2 FILL, DRAIN, DUMP	30	
LO2 FEED	120	
LO2 FILL, DRAIN, DUMP	26	
THRUST VECTOR CONTROL		100
PRESSURIZATION & VENT		355
LH2 TANK AUTOGENOUS PRESS SUPPLY	53	
LO2 TANK AUTOGENOUS PRESS SUPPLY	47	
LH2 TANK VENT/RELIEF - GROUND/ASCENT	165	
LH2 TANK VENT/RELIEF - SPACE	30	
LO2 TANK VENT/RELIEF - GROUND/ASCENT	30	
LO2 TANK VENT/RELIEF - SPACE	30	
PNEUMATICS		152
PLUMBING SYSTEM	61	
HELIUM BOTTLE	91	

GROUP WEIGHT STATEMENT

NWD AEROBRAKED OTV, W_p = 132.8K

ATTITUDE CONTROL SYSTEM (ACS)	(266)
THRUSTER MODULES (12) - INCL 24 THRUSTERS	65
TANKAGE/PROPELLANT SYSTEM	201
N2H4 TANK ASSEMBLIES (6)	162
N2H4 FEED	39
N2H4 FILL, DRAIN, DUMP - SPACE BASING	--
N2 BOTTLES	--
N2 PRESS PLUMBING	--
SPACE MAINTENANCE PROVISIONS	(--)
CRITICAL AVIONICS ASSEMBLIES REMOVAL	--
FUEL CELL REMOVAL	-
MAIN ENGINE REMOVAL	--
THRUSTER MODULE REMOVAL	

WEIGHT GROWTH MARGIN - EXCLUDE BALLUTE MARGIN	(1273)
MARGIN ON NEW HARDWARE (15 %)	1177
MARGIN ON EXISTING/PROTOTYPE HARDWARE (5 %)	96
OTV DRY WEIGHT - INCL BALLUTE	12,190 LB
MPS RESIDUAL FLUIDS & GASES	(1380)
TRAPPED LH2	19
TRAPPED LO2	201
LH2 FUEL BIAS	125
GH2 IN EMPTY LH2 TANK	420
GO2 IN EMPTY LO2 TANK	575
GHe FOR PNEUMATICS - TOTAL	40
ACS RESIDUAL FLUIDS & GASES	(17)
TRAPPED N2H4	11
GN2 - TOTAL	6
EPS RESIDUAL FLUIDS & GASES	(12)
TRAPPED O2	4
TRAPPED H2	1
TRAPPED PRODUCT WATER	7

GROUP WEIGHT STATEMENT

NWD AEROBRAKED OTV, W_p = 132,81

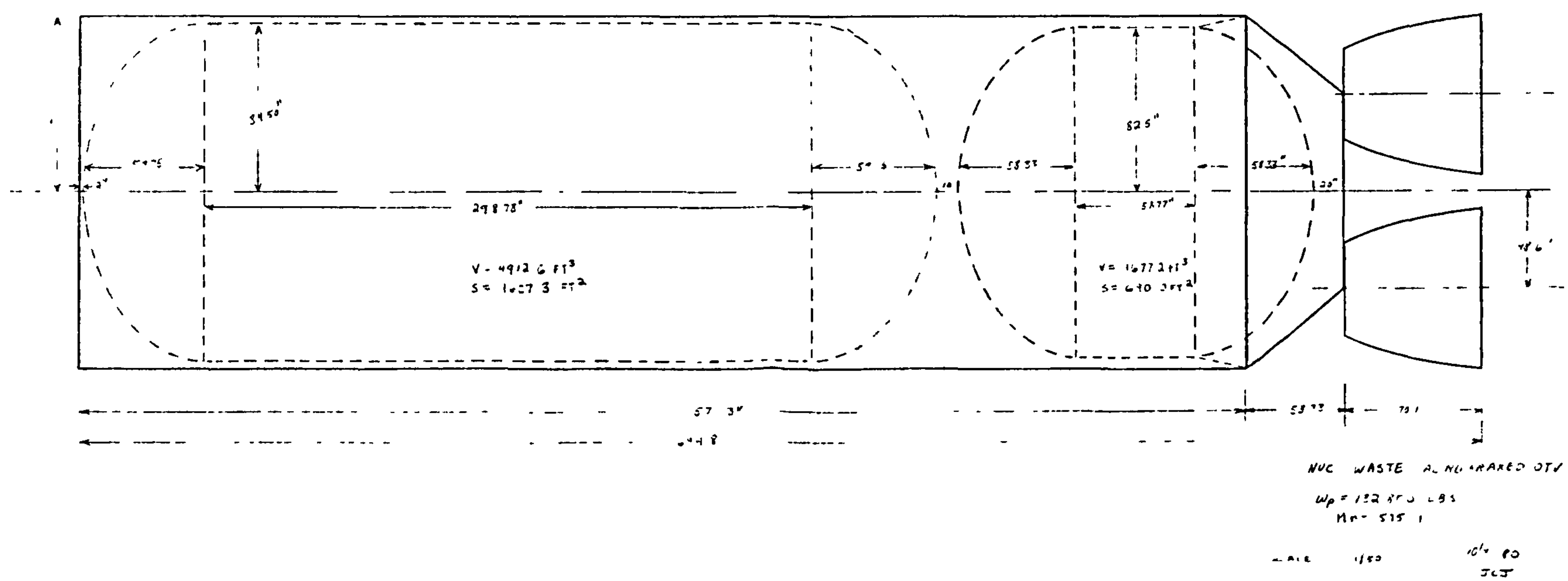
MPS RESERVE PROPELLANT	(197) *
ACS RESERVE PROPELLANT	(57)
EPS RESERVE REACTANT	(18)
MPS INFLIGHT LOSSES	(224)
LH2 FOR CHILLDOWN/START/STOP } (5 FIRINGS) (BOTH ENGINES)	52
LO2 FOR CHILLDOWN/START/STOP } (5 FIRINGS) (BOTH ENGINES)	72
LH2 BOILOFF/VENT LOSS	--
LO2 BOILOFF/VENT LOSS	--
LH2 FOR BALLUTE INFLATION/COOLING	--
LH2 EXPENDED THRU ENGINE(S) DURING AEROBRAKING	14
LO2 EXPENDED THRU ENGINE(S) DURING AEROBRAKING	86
MPS NOMINAL PROPELLANT	(132,577) *
ACS NOMINAL PROPELLANT	(572)
EPS NOMINAL REACTANT	(70)
OTV GROSS WEIGHT	147,314 LB

$$\text{MASS FRACTION} = \frac{\overset{132,577+197}{\text{MPS PROPELLANT, NOMINAL + RESERVE}}}{\underset{147,314}{\text{GROSS WEIGHT}}} = 0.9013$$

* O/F = 5.75:1

THE BOEING COMPANY

NUMBER
REV LTR



SHEET

D1 4802 8100 REV 8/71

J.T. [illegible]
12/3/80

SUMMARY WEIGHT STATEMENT
LONG LIFE OTV, W_p = 61.9 K

STRUCTURE	2970
THERMAL CONTROL	554
AVIONICS - O.T.	703
AVIONICS - S.O.I.	334
ELECTRICAL POWER SYSTEM - O T	664
ELECTRICAL POWER SYSTEM - S.O I	492
MAIN PROPULSION SYSTEM	1842
ATTITUDE CONTROL SYSTEM	138
WEIGHT GROWTH MARGIN	943
(OTV DRY WEIGHT)	(8640 LB)
MPS RESIDUAL FLUIDS & GASES	745
ACS RESIDUAL FLUIDS & GASES	12
EPS RESIDUAL FLUIDS & GASES	12
MPS RESERVE PROPELLANT	700
ACS RESERVE PROPELLANT	27
EPS RESERVE REACTANT	18
MPS INFLIGHT LOSSES	439
MPS NOMINAL PROPELLANT	61,237
ACS NOMINAL PROPELLANT	270
EPS NOMINAL REACTANT	70
(OTV GROSS WEIGHT)	(72,170 LB)

STAGE MASS FRACTION 0.8582

END OF MISSION REF WT = 8640 + (745 + 12 + 12) + 18 = 9427 LB

R. T. Conrad
12/3/80

GROUP WEIGHT STATEMENT
LONG LIFE OTV, W_P = 620 K

STRUCTURE		(2970)
LH2 TANK - INCL SUPPORT STRUTS		660
LO2 TANK - INCL SUPPORT STRUTS		450
BODY STRUCTURES		1800
SANDWICH PANELS (994 FT²; 10 MIL FACE SHTS)	550	
LONGITUDINAL STIFFENERS - P/L ADAPTER (8)	25	
FORWARD RING	50	
MAIN SUPPORT RING, FWD YZ LOADS	100	
LH2 TANK SUPPORT RINGS (2)	40	
LO2 TANK SUPPORT RING	60	
AFT RING (LATCHING INTERFACE)	70	
AVIONICS RING SECTION - O.T. AVIONICS/EQUIP	300	
MOUNTING STRUCTURES - S.O.I. AVIONICS/EQUIP	75	
MISC. MOUNTING/SUPPORT STRUCTURES	30	
P/L ADAPTER ATTACH FTG'S & BOLTS (8 PLACES)	20	
MAIN SUPPORT FITTING, FWD YZ LOADS (3)	90	
RMS GRAPPLE FITTING	10	
THRUST STRUCTURE (BODY MOUNTED CONE, WITH BASE RING)	270	
BERTHING PROBES - FIXED	20	
UMBILICAL INTERFACES	25	
ASSEMBLY & INSTALLATION HARDWARE	65	
METEOROID SHIELD {10 MIL ALUM; 60° SIDE REGIONS} (362 FT²)		60
THERMAL CONTROL		(554)
SUN SHIELD INSTALLATION (240 FT²)		150
RADIATOR SYSTEM - FUEL CELL COOLING		60
INSULATION		228
MLI - LH2 TANK (40 LAYERS) } 0.30 MIL ENCLOSED SHEETS	177	
MLI - LO2 TANK (20 LAYERS) } 0.30 MIL ENCLOSED SHEETS	49	
MLI - EPS TANKS	2	
MLI PURGE		96
PURGE CONTAINMENT - LH2 TANK	33	
PURGE CONTAINMENT - LO2 TANK	33	
PURGE PLUMBING - LH2 TANK	16	
PURGE PLUMBING - LO2 TANK	14	
MISC. THERMAL CONTROL		20

GROUP WEIGHT STATEMENT
LONG LIFE OTV, W_p = 62.0 K

AVIONICS - O.T.		(703)
GUIDANCE & NAVIGATION		99
COMMUNICATIONS		95
DATA MANAGEMENT		349
INSTRUMENTATION		160
PROPELLANT LOADING/MEASUREMENT	60	
OTV MEASURING SYSTEM	100	
AVIONICS - S.O.I.		(334)
GUIDANCE & NAVIGATION		45
COMMUNICATIONS		67
DATA MANAGEMENT		172
INSTRUMENTATION		50
ELECTRICAL POWER SYSTEM - O.T.		(664)
POWER SOURCE		264
FUEL CELLS (2)	160	
BATTERY (25 AMP-HR)	28	
O2 TANK ASSY	10	
H2 TANK ASSY	6	
PLUMBING SYSTEM	60	
CONVERSION & DISTRIBUTION		400
ELECTRONICS	156	
WIRE HARNESS	244	
ELECTRICAL POWER SYSTEM - S.O.I.		(492)
POWER SOURCE		212
SOLAR ARRAY	20	
BATTERY (89 AMP-HR)	192	
CONVERSION & DISTRIBUTION		280
ELECTRONICS	120	
WIRE HARNESS	160	

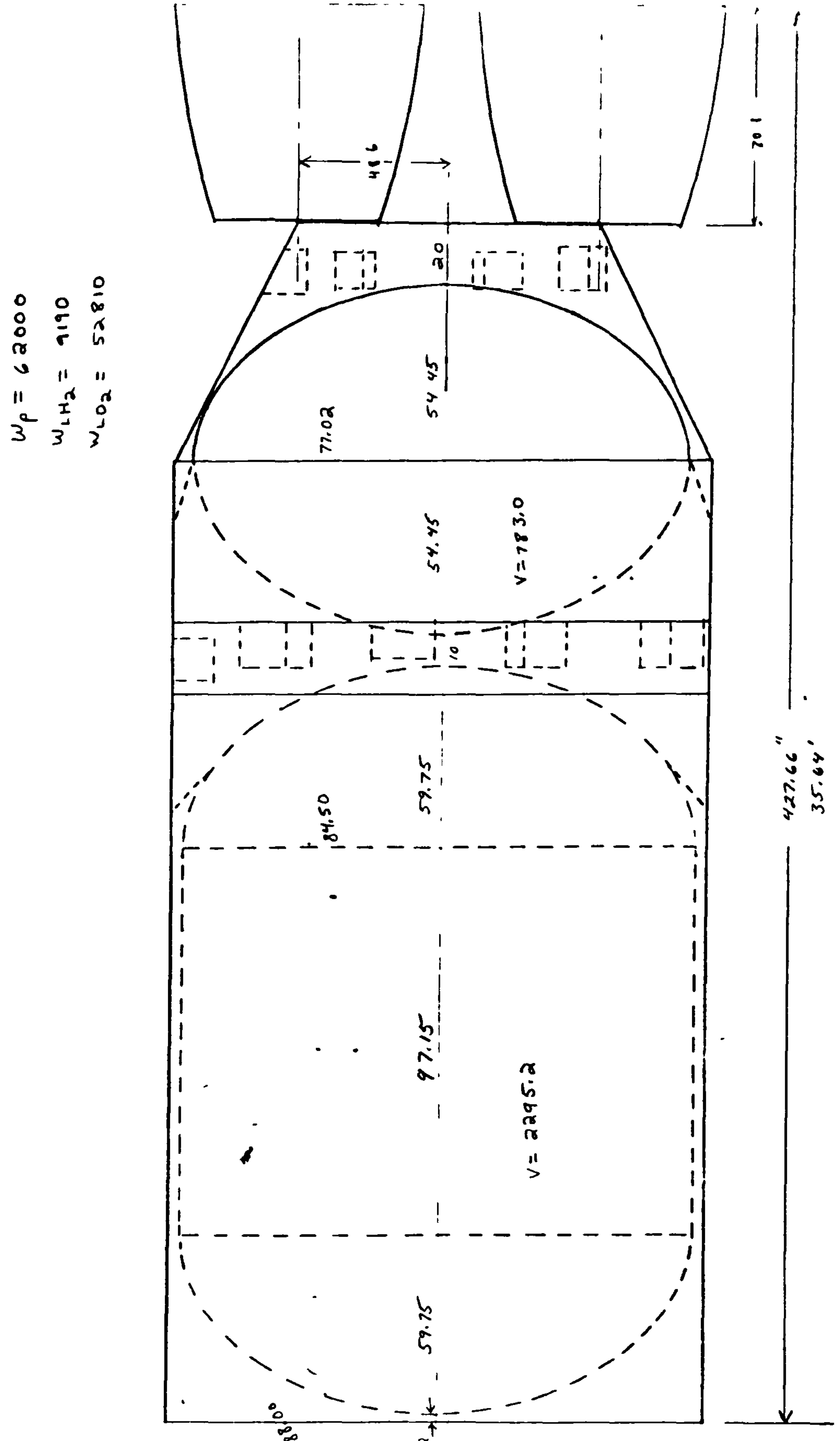
GROUP WEIGHT STATEMENT
LONG LIFE OTV, W_p = 620 K

	MAIN PROPULSION SYSTEM*		(1842)
	MAIN ENGINE – RL10 IIB (2)		808
	PROPELLANT SYSTEM		486
50	LH2 FEED	210*	
	LH2 FILL, DRAIN, DUMP	30	
50	LO2 FEED	220*	
	LO2 FILL, DRAIN, DUMP	26	
	THRUST VECTOR CONTROL		100
	PRESSURIZATION & VENT		362
10	LH2 TANK AUTOGENOUS PRESS SUPPLY	63	
10	LO2 " " " "	57	
25	LH2 TANK VENT/RELIEF – GROUND/ASCENT	105	
20	LH2 " " " – SPACE	50	
10	LO2 " " " – GROUND/ASCENT	37	
20	LO2 " " " – SPACE	50	
	PNEUMATICS		86
10	PLUMBING	71	
	HELIUM BOTTLE	15	
	ATTITUDE CONTROL SYSTEM		(138)
	THRUSTERS (4 MODULES / 12 THRUSTERS)		32
	TANKAGE / PROPELLANT SYSTEM		106
	N2H4 TANK ASSEMBLIES (3)	81	
	N2H4 FEED	25	
	WEIGHT GROWTH MARGIN		(943)
	MARGIN ON NEW HARDWARE (15%)		840
	MARGIN ON EXISTING/PROTOTYPE HARDWARE (5%)		103
	(OTV DRY WEIGHT)		(8640 LB)

* DUAL FAILURE TOLERANT

GROUP WEIGHT STATEMENT
LONG LIFE OTV, W_p = 61.9K

MPS RESIDUAL FLUIDS & GASES	(745)
TRAPPED LH2	19
TRAPPED LO2	201
LH2 FUEL BIAS	58
GH2 IN EMPTY LH2 TANK	196
GO2 IN EMPTY LO2 TANK	268
GHe FOR PNEUMATICS - TOTAL	3
ACS RESIDUAL FLUIDS & GASES	(12)
TRAPPED N2H4	8
GN2 (85% N2/15% He)	4
EPS RESIDUAL FLUIDS & GASES	(12)
TRAPPED O2	4
TRAPPED H2	1
TRAPPED PRODUCT WATER	7
MPS RESERVE PROPELLANT	(700)
ACS RESERVE PROPELLANT	(27)
EPS RESERVE REACTANT	(18)
MPS INFLIGHT LOSSES	(439)
LH2 FOR ENGINE CHILLDOWN/START/STOP } 2 FIRINGS PER ENGINE	68
LO2 " " " " "	66
LH2 BOILOFF/VENT LOSS	305
LO2 " " "	0
MPS NOMINAL PROPELLANT	(61,237)
ACS NOMINAL PROPELLANT	(270)
EPS NOMINAL REACTANT	(70)
(OTV GROSS WEIGHT)	(72,170 LB)



ENGR	JCJ	12/1/80	REVISED	DATE	NUKE WASTE LLOTV $W_p = 62000$
CHECK					
APR					
APR					G21 BOEING

P.T Cont'd
10/31/80

SUMMARY WEIGHT STATEMENT
CRYOGENIC SOIS, W_P = 27.1 K

STRUCTURE	1450
THERMAL CONTROL	370
AVIONICS	307
ELECTRICAL POWER SYSTEM (EPS)	482
MAIN PROPULSION SYSTEM (MPS)	815
ATTITUDE CONTROL SYSTEM (ACS)	138
WEIGHT GROWTH MARGIN	458
(DRY WEIGHT)	(4020 LB)
MPS RESIDUAL FLUIDS & GASES	390
ACS RESIDUAL FLUIDS & GASES	12
MPS RESERVE PROPELLANT - F.P.R.	464
ACS RESERVE PROPELLANT	31
MPS INFLIGHT LOSSES	59
MPS NOMINAL PROPELLANT	26,600
ACS NOMINAL PROPELLANT	307
(GROSS WEIGHT)	(31,883 LB)

STAGE MASS FRACTION 0.8489

END OF MISSION REFERENCE WEIGHT = 4020 + 390 + 12 + 31 = 4453 LB

R.T Cenino
10/31/80

GROUP WEIGHT STATEMENT
CRYOGENIC SOIS, W_p = 27.1 K

STRUCTURE			(1450)
LH2 TANK – INCL SUPPORT STRUTS			313
LO2 TANK – INCL SUPPORT STRUTS			217
BODY STRUCTURES			880
SANDWICH PANELS (682 FT², 10 MIL FACE SHTS)		377	
LONGITUDINAL STIFFENERS (8)		43	
FORWARD RING		30	
LH2 TANK SUPPORT RING		20	
LO2 TANK SUPPORT RING		30	
AFT RING		30	
MOUNTING STRUCTURES – AVIONICS/EQUIP		100	
P/L ADAPTER ATTACH FTG'S & BOLTS (8 PLACES)		20	
INTERSTAGE ATTACH FTG'S (8 PLACES)		20	
RMS GRAPPLE FITTING		10	
THRUST STRUCTURE	{BODY MOUNTED CONE / LESS BASE RING}	145	
UMBILICAL INTERFACES		25	
ASSEMBLY & INSTALLATION HARDWARE		30	
METEOROID SHIELDS	{10 MIL ALUM / 60° SIDE REGIONS ONLY		40
THERMAL CONTROL			(370)
SUN SHIELD INSTALLATION (240 FT²)			150
INSULATION			130
MLI – LH2 TANK (40 LAYERS)	} 0.30 MIL ENCLOSED SHIELDS	102	
MLI – LO2 TANK (20 LAYERS)		28	
MLI PURGE			80
PURGE CONTAINMENT – LH2 TANK		33	
PURGE CONTAINMENT – LO2 TANK		17	
PURGE PLUMBING – LH2 TANK		16	
PURGE PLUMBING – LO2 TANK		14	
MISC. THERMAL CONTROL			10

GROUP WEIGHT STATEMENT
CRYOGENIC SOIS, W_P = 27.1 K

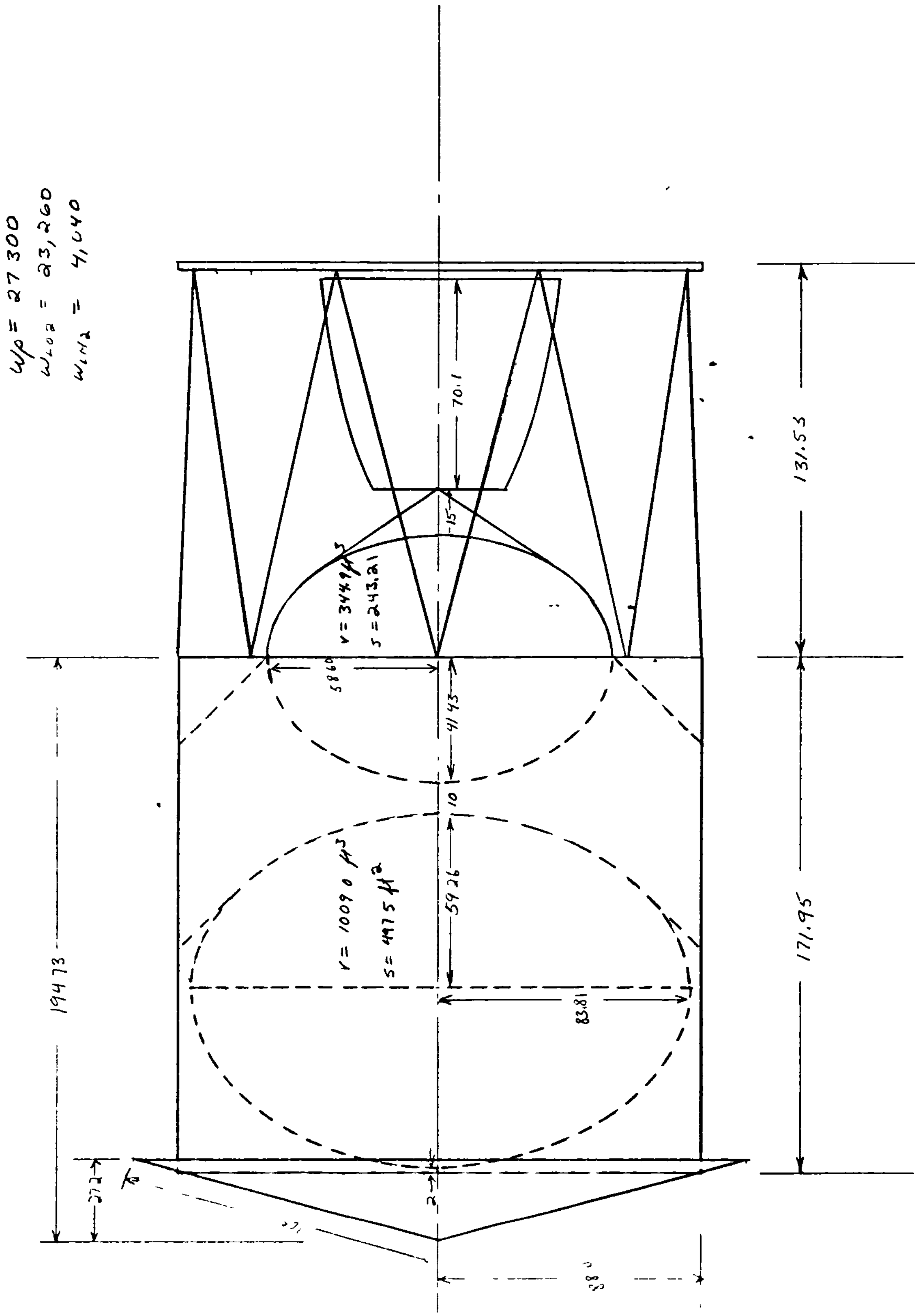
AVIONICS		(307)
GUIDANCE & NAVIGATION		45
COMMUNICATIONS		67
DATA MANAGEMENT		136
RENDEZVOUS & DOCKING		14
INSTRUMENTATION		45
ELECTRICAL POWER SYSTEM (EPS)		(482)
POWER SOURCE		212
SOLAR ARRAY	20	
BATTERY (89 AMP-HR)	192	
CONVERSION & DISTRIBUTION		270
CHARGER/REGULATOR (2)	100	
PWR SWITCHING UNIT (2)	20	
WIRE HARNESS	150	
MAIN PROPULSION SYSTEM (MPS)		(815)
MAIN ENGINE (RL10-IIB)		404
PROPELLANT SYSTEM		138
LH2 FEED	50	
LH2 FILL, DRAIN, DUMP	30	
LO2 FEED	32	
LO2 FILL, DRAIN, DUMP	26	
THRUST VECTOR CONTROL		50
PRESSURIZATION & VENT SYSTEM		166
LH2 TANK AUTOGENOUS PRESS SUPPLY	23	
LO2 TANK AUTOGENOUS PRESS SUPPLY	20	
LH2 TANK VENT/RELIEF – GROUND/ASCENT	49	
LH2 TANK VENT/RELIEF – SPACE	30	
LO2 TANK VENT/RELIEF – GROUND/ASCENT	14	
LO2 TANK VENT/RELIEF – SPACE	30	
PNEUMATIC SYSTEM		57
PLUMBING	52	
HELIUM BOTTLE	5	

GROUP WEIGHT STATEMENT
CRYOGENIC SOIS, W_P = 27.1 K

ATTITUDE CONTROL SYSTEM (ACS)		(138)
THRUSTERS (6 MODULES/12 THRUSTERS)		32
TANKAGE/PROPELLANT SYSTEM		106
N2H4 TANK ASSEMBLIES (3)	81	
N2H4 FEED	25	
WEIGHT GROWTH MARGIN		(458)
MARGIN ON NEW HARDWARE (15%)		419
MARGIN ON EXISTING/PROTOTYPE HARDWARE (5%)		39
SOIS DRY WEIGHT		4020 LB
MPS RESIDUAL FLUIDS & GASES		(390)
TRAPPED LH2		14
TRAPPED LO2		134
LH2 FUEL BIAS		36
GH2 IN EMPTY LH2 TANK		86
GO2 IN EMPTY LO2 TANK		119
GHe FOR PNEUMATICS		1
ACS RESIDUAL FLUIDS & GASES		(12)
TRAPPED N2H4		8
GN2 (85% N2/15% He)		4
MPS RESERVE PROPELLANT - F.P.R		(464)
ACS RESERVE PROPELLANT		(31)

GROUP WEIGHT STATEMENT
CRYOGENIC SOIS, W_p = 27.1 K

MPS INFLIGHT LOSSES	(59)
LH2 FOR ENGINE CHILLDOWN/START/STOP (1)	6
LO2 FOR ENGINE CHILLDOWN/START/STOP (1)	13
LH2 BOILOFF/VENTING	40
LO2 BOILOFF/VENTING	0
MPS NOMINAL PROPELLANT	(26,600)
ACS NOMINAL PROPELLANT	(307)
SOIS GROSS WEIGHT	31,883 LB



ENGR	JCJ	10/23/90	REVISED	DATE	LO2/LH2 5015 MATCHED TO 5042 AEROBRAKED STV Wp = 27,300 LBS	
CHECK						
APR						
APR					G27 **BOEING**	

APPENDIX H

DATA FOR RADIATION SHIELD TRADES

TABLE 1

REFERENCE COMMERCIAL ELEMENTAL WASTE MIX COMPOSITION (PW-4b)

	Constituent	Amount, kg/MTHM		Constituent	Amount, kg/MTHM
Inerts	Na_2O	---	Fission	TeO_2	0.725
	Fe_2O_3	1.511	Products	Cs_2O	2.880
	Cr_2O_3	0.345	(Cont'd.)	BaO	1.567
	NiO	0.141		La_2O_3	1.480
	P_2O_5	0.672		CeO_2	3.323
	Gd_2O_3	---		Pr_6O_{11}	1.482
				Nd_2O_3	4.522
Fission	Rb_2O	0.354		Pm_2O_3	0.123
Products	SrO	1.059		Sm_2O_3	0.924
	Y_2O_3	0.598		Eu_2O_3	0.200
	ZrO_2	4.944		Gd_2O_3	0.137
	MoO_3	5.176			
	Tc_2O_7	1.291			
	RuO_2	2.972	Actinides	U_3O_8	1.169
	Rh_2O_3	0.480		NpO_2	0.865
	PdO	1.483		PuO_2	0.010
	Ag_2O	0.088		Am_2O_3	0.181
	CdO	0.097		Cm_2O_3	0.040
			TOTAL		40.8

TABLE 2

RADIONUCLIDE COMPOSITION OF COMMERCIAL WASTE (PW-4b)

Isotope	Nuclide Mass, kg/MTHM	Payload[a] Nuclide Mass, kg	Oxide Mass, kg/MTHM	Activity Ci/MTHM	Payload[a] Activity, Ci
Rb-85	0.092	7.821	0.101	0	0
Rb-87	0.232	18.360	0.253	1.90E-5	1.50E-3
Total	0.324	25.641	0.354 (Rb_2O)	1.90E-5	1.50E-3
Sr-88	0.406	32.129	0.480	0	0
Sr-90	0.492	38.935	0.579	6.95E+4	5.50E+6
Total	0.898	71.064	1.059 (SrO)	6.95E+4	5.50E+6
Y-89	0.471	37.273	0.598	0	0
Y-90	<0.001	< 0.079	<0.001	6.97E+4	5.52E+6
Total	0.471	37.352	0.598 (Y_2O_3)	6.97E+4	5.52E+6
Zr-90	0.142	11.237	0.193	0	0
Zr-91	0.587	46.453	0.793	0	0
Zr-92	0.641	50.727	0.864	0	0
Zr-93	0.715	56.583	0.961	1.83E+0	1.45E+2
Zr-94	0.774	61.252	1.037	0	0
Zr-96	0.822	65.050	1.096	0	0
Total	3.681	291.302	4.944 (ZrO_2)	1.83E+0	1.45E+2
Mo-95	0.066	5.223	0.094	0	0
Mo-97	1.060	83.885	1.590	0	0
Mo-98	1.140	90.216	1.626	0	0
Mo-100	1.260	99.712	1.866	0	0
Total	3.526	279.036	5.176 (MoO_3)*	0	0
Tc-99	0.976	77.237	1.291	1.66E+1	1.31E+3
Total	0.976	77.237	1.291 (Tc_2O_7)*	1.66E+1	1.31E+3
Ru-100	0.061	4.827	0.080	0	0
Ru-101	0.806	63.784	1.062	0	0
Ru-102	0.008	63.942	1.061	0	0
Ru-104	0.588	46.532	0.769	0	0
Total	2.263	179.085	2.972 (RuO_2)*	0	0
Rh-103	0.389	30.784	0.480	0	0
Total	0.389	30.784	0.480 (Rh_2O_3)*	0	0

(a) For 5,500 kg CERMET payload

TABLE 2

RADIONUCLIDE COMPOSITION OF COMMERCIAL WASTE (PW-4b) (Continued)

Isotope	Nuclide Mass, kg/MTHM	Payload[a] Nuclide Mass, kg	Oxide Mass, kg/MTHM	Activity Ci/MTHM	Payload[a] Activity, Ci
Pd-104	0.197	15.590	0.257	0	0
Pd-105	0.233	18.439	0.304	0	0
Pd-106	0.366	28.964	0.476	0	0
Pd-107	0.188	14.878	0.244	8.98E-2	7.11E+0
Pd-108	0.128	10.129	0.166	0	0
Pd-110	0.027	2.137	0.035	0	0
Total	1.139	90.137	1.438(PdO)*	8.98E-2	7.11E+0
Ag-109	0.082	6.489	0.088	0	0
Total	0.082	6.489	0.088(Ag_2O)*	0	0
Cd-110	0.043	3.403	0.049	0	0
Cd-111	0.018	1.424	0.020	0	0
Cd-112	0.009	0.712	0.011	0	0
Cd-114	0.012	0.950	0.014	0	0
Cd-116	0.003	0.237	0.004	0	0
Total	0.085	6.726	0.097(CdO)**	0	0
Te-125	0.011	0.871	0.014	0	0
Te-125	0.138	10.921	0.173	0	0
Te-130	0.428	33.871	0.535	0	0
Total	0.577	45.663	0.722(TeO_2)**	0	0
Cs-133	1.200	94.964	1.257	0	0
Cs-134	0.007	0.554	0.007	8.55E+3	6.77E+5
Cs-135	0.370	29.281	0.392	3.27E-1	2.59E+1
Cs-137	1.160	91.799	1.224	1.01E+5	7.99E+6
Total	2.737	216.598	2.880(Cs_2O)	1.10E+5	8.71E+6
Ba-134	0.202	15.986	0.226	0	0
Ba-136	0.020	1.583	0.022	0	0
Ba-137	0.241	19.072	0.269	9.45E+4	7.48E+6
Ba-138	0.940	74.388	1.049	0	0
Total	1.403	111.029	1.567(BaO)	9.45E+4	7.48E+6
La-139	1.260	99.712	1 480	0	0
Total	1.260	99.712	1.480(La_2O_3)	0	0

(a) For 5,500 kg CERMET payload

TABLE 2

RADIONUCLIDE COMPOSITION OF COMMERCIAL WASTE (PW-4b) (Continued)

Isotope	Nuclide Mass, kg/MTHM	Payload[a] Nuclide Mass, kg	Oxide Mass, kg/MTHM	Activity Ci/MTHM	Payload[a] Activity, Ci
Ce-140	1.420	112.374	1.749	0	0
Ce-142	1.280	101.295	1.574	0	0
Total	2.700	213.669	3.323 (CeO_2)	0	0
Pr-141	1.230	97.338	1.482	0	0
Total	1.230	97.338	1.482 (Pr_6O_{11})	0	0
Nd-142	0.022	1.741	0.026	0	0
Nd-143	0.746	59.036	0.871	0	0
Nd-144	1.270	100.504	1.484	0	0
Nd-145	0.651	51.518	0.759	0	0
Nd-146	0.665	52.626	0.774	0	0
Nd-148	0.354	28.014	0.411	0	0
Nd-150	0.171	13.532	0.198	0	0
Total	3.879	306.971	4.522 (Nd_2O_3)	0	0
Pm-147	0.106	8.388	0.123	9.82E+4	7.77E+6
Total	0.106	8.388	0.123 (Pm_2O_3)	9.82E+4	7.77E+6
Sm-147	0.078	6.173	0.090	0	0
Sm-148	0.087	6.885	0.101	0	0
Sm-149	0.354	28.014	0.411	0	0
Sm-150	0.191	15.115	0.221	0	0
Sm-151	0.021	1.662	0.024	5.63E+2	4.46E+4
Sm-152	0.048	3.799	0.055	0	0
Sm-154	0.019	1.504	0.022	0	0
Total	0.798	63.152	0.924 (Sm_2O_3)	5.63E+2	4.46E+4
Eu-151	0.003	0.237	0.004	0	0
Eu-153	0.137	10.842	0.158	0	0
Eu-154	0.033	2.612	0.038	4.78E+3	3.78E+5
Total	0.173	13.691	0.2200 (Eu_2O_3)	4.78E+3	3.78E+5
Gd-154	0.016	1.266	0.019	0	0
Gd-155	0.005	0.396	0.006	0	0
Gd-156	0.084	6 647	0.097	0	0
Gd-158	0.012	0.950	0.014	0	0
Gd-160	0.001	0.079	0.001	0	0
Total	0.118	9.338	0.137 (Gd_2O_3)	0	0

(a) For 5,500 kq CERMET payload

TABLE 2

RADIONUCLIDE COMPOSITION OF COMMERCIAL WASTE (PW-4b) (Continued)

Isotope	Nuclide Mass, kg/MTHM	Payload[a] Nuclide Mass, kg	Oxide Mass, kg/MTHM	Activity Ci/MTHM	Payload[a] Activity, Ci
U-235	0.008	0.633	0.010	1.81E-5	1.43E-3
U-236	0.004	0.317	0.005	2.69E-4	2.13E-2
U-238	0.979	77.475	1.154	3.26E-4	2.58E-2
Total	0.991	78.425	1.169(U_3O_8)	6.13E-4	4.85E-2
Np-237	0.762	60.302	0.865	5.37E-1	4.25E-1
Total	0.762	60.302	0.865(NpO_2)	5.37E-1	4.25E-1
Pu-239	0.005	0.396	0.006	3.22E-1	2.55E+1
Pu-240	0.003	0.237	0.003	5.82E-1	4.61E+1
Pu-241	0.001	0.079	0.001	9.80E+1	7.86E+3
Total	0.009	0.712	0.010(PuO_2)	9.89E+1	7.83E+3
Am-241	0.129	10.209	0.142	4.41E+1	3.49E+4
Am-243	0.035	2.770	0.039	6.83E+0	5.41E+2
Total	0.164	12.979	0.181(Am_2O_3)	4.48E+2	3.55E+4
Cm-244	0.036	2.849	0.040	2.94E+3	2.33E+5
Total	0.036	2.849	0.040(Cm_2O_3)	2.94E+3	2.33E+5
Reprocessing Chemicals	--	211.1	--	--	--

(a) For 5,500 kg CERMET payload

TABLE 3

GAMMA SPECTRUM FROM ORIGEN

Group No.	Upper Boundary	Rate (part/sec)
1	5.5 MeV	1.27×10^{8}
2	5.0	1.94×10^{8}
3	4.5	4.11×10^{8}
4	4.0	6.51×10^{8}
5	3.5	1.02×10^{9}
6	3.0	1.63×10^{9}
7	2.6	3.50×10^{9}
8	2.2	2.55×10^{12}
9	1.8	8.22×10^{14}
10	1.35	1.04×10^{16}
11	0.9	3.37×10^{17}
12	0.4	1.39×10^{16}
13	0.225	3.57×10^{12}
14	0.175	8.37×10^{12}
15	0.125	8.39×10^{11}
16	0.075	4.46×10^{14}
17	0.045	5.85×10^{12}
18	0.035	3.49×10^{13}
	0.025	
	TOTAL	3.63×10^{17}

TABLE 4

GAMMA SPECTRUM INPUT TO ANISN

Group No.	Upper Boundary	Rate (part/sec)
1	10 0 M	0
2	8.0	0
3	6.5	1.22×10^{8}
4	5.0	6.04×10^{8}
5	4.0	1.67×10^{9}
6	3.0	2.51×10^{9}
7	2.5	1.27×10^{12}
8	2.0	2.57×10^{14}
9	1.66	1.03×10^{15}
10	1.33	7.65×10^{15}
11	1.0	6.97×10^{16}
12	0.8	1.35×10^{17}
13	0.6	1.35×10^{17}
14	0.4	7.96×10^{15}
15	0.3	5.97×10^{15}
16	0.2	1.06×10^{13}
17	0.1	3.72×10^{14}
18	0.05	1.15×10^{14}
	0.01	
	TOTAL	3.63×10^{17}

TABLE 5

SPONTANEOUS FISSION NEUTRONS FROM ^{244}Cm

Abundance, neutrons/(sec)(g of nuclide)

4.2×10^5	2.1×10^5
8.7×10^5	2.2×10^5
8.9×10^5	2.9×10^5
7.7×10^5	2.4×10^5
7.9×10^5	1.8×10^5
7.8×10^5	1.4×10^5
6.8×10^5	1.1×10^5
6.1×10^5	8.2×10^4
5.5×10^5	5.9×10^4
5.0×10^5	4.1×10^4
4.6×10^5	3.1×10^4
4.4×10^5	2.5×10^4
3.6×10^5	2.8×10^4
3.0×10^5	8.2×10^3
2.8×10^5	8.6×10^3
2.7×10^5	5.7×10^3
2.6×10^5	3.9×10^3

TABLE 6

NEUTRON SPECTRUM INPUT TO ANISN

Group No.	Upper Boundary	Rate (n/sec)
1	14.92 Mev	6.48×10^{6}
2	12.2	3.28×10^{7}
3	16.0	9.04×10^{7}
4	8.18	4.52×10^{8}
5	6.36	1.14×10^{9}
6	4.96	1.48×10^{9}
7	4.06	3.36×10^{9}
8	3.01	2.59×10^{9}
9	2.46	5.36×10^{8}
10	2.35	3.43×10^{9}
11	1.83	6.64×10^{9}
12	1.11	6.11×10^{9}
13	0.55	4.65×10^{9}
14	0.111	5.76×10^{8}
15	$3\ 35 \times 10^{-3}$	2.96×10^{6}
16	5.83×10^{-4}	2.15×10^{5}
17	1.01×10^{-4}	1.42×10^{4}
18	2.9×10^{-5}	2.05×10^{3}
19	1.01×10^{-5}	4.41×10^{2}
20	3.06×10^{-6}	6.87×10^{1}
21	$1\ 12 \times 10^{-6}$	1.51×10^{1}
22	$4\ 14 \times 10^{-}$	4.38
	10^{-8}	
	TOTAL	3.05×10^{10}

TABLE 7

CONTENTS OF THE DLC-23/CASK
22-18 LIBRARY OF COUPLED NEUTRON AND GAMMA-RAY CROSS SECTIONS
(Revised March 1975)

Element	MAT Number	Source	POPOP4 ID
H	1001	ENDF/B-II	010101
He-4	1270	ENDF/B-IV	No - prod.
Be	1007	ENDF/B-II	040101, 040401
B-10	1009	ENDF/B-II	050201
C	1040	ENDF/B-II	060102, 060301
N	4133 MOD 3	DNA	See Note 4
O	1013	ENDF/B-II	086301, 080201
Na	1059	ENDF/B-II	110101, 113301
Mg	1014	ENDF/B-II	120101, 120301
Al	1015	ENDF/B-II	130101, 130301
Si	530	PDS-16	140101, 140301
K	5005	ENDF/B-III, Pre.	190101, 190301
Ca	540	PDS-16	200101, 200301
Ti	1016	ENDF/B-II	220101, 220301
Cr	1121	ENDF/B-III	See Note (9)
Mn	1019	ENDF/B-III	250101
Fe	4180 MOD 0	DNA	See Note 4
Ni	1123	ENDF/B-III	See Note (9)
Cu	1087	ENDF/B-III	290101
Zr	7033	LENDL	See Note (10)
Mo	1111	ENDF/B-III	420101
Sn	7039	LENDL	See Note (10)
Ta	1126	ENDF/B-III	731303, 731107
W	1060-1063	ENDF/B-III	740107
Pb	43	ORNL-UK	820102, 820301
U-235	102	ENDF/A-700	925101, 925301, 925801
U-238	103	ENDF/A-700	928112, 925301, 925801
Pu-239	104	ENDF/A-700	928112, 925301, 925801
Pu-240	1105	ENDF/B-II	928112, 925301, 925801

TABLE 8

CERMET COMPOSITION

Element	Percent	Element	Percent
O	12.13	Ba	2.02
P	0.42	La	1.81
Cr	0.34	Ce	3.89
Fe	24.01	Pr	1.77
Ni	6.29	Nd	5.58
Cu	7.36	Pm	0.15
Rb	0.47	Sm	1.15
Sr	1.29	Eu	0.25
Y	0.68	Gd	0.17
Zr	5.30	U235	0.0115
Mo	10.39	U236	0.0058
Tc	1.40	U238	1.409
Ru	3.26	Np237	1.10
Rh	0.56	Pu239	.0072
Pd	1.64	Pu240	.0043
Ag	0.12	Pu241	.001
Cd	0.12	Am	.164
Te	0.83	Cm	.036
Cs	3.94		

Density = 6.7 g/cc

TABLE 9

COMPOSITION OF SHIELDING MATERIALS

Stainless Steel	7.98 g/cc
Fe	68.75%
Cr	17 %
Ni	12 %
Mo	2.25%
Air	10^{-3} g/cc
N	80 %
O	20 %
Depleted Uranium	18.7 g/cc
U235	0.3 %
U238	99.7 %
Tantalum	16.6 g/cc
Ta	100 %
Graphite	2.25 g/cc
C	100 %

TABLE 10

NEUTRON GROUP STRUCTURE, SOURCE SPECTRUM,
AND
FLUX-TO-DOSE CONVERSION FACTORS

Neutron Group	Upper Energy (eV)		Flux-to-Dose Conversion Factors [(mrem/hr)/neut/cm^2/sec]	
1	1.492	7	2.088	-1
2	1.220	7	1.656	-1
3	1.000	7	1.476	-1
4	8.180	6	1.476	-1
5	6.360	6	1.404	-1
6	4.960	6	1.332	-1
7	4.060	6	1.296	-1
8	3.010	6	1.260	-1
9	2.460	6	1.260	-1
10	2.350	6	1.296	-1
11	1.830	6	1.332	-1
12	1.110	6	1.188	-1
13	5.500	5	5.400	-2
14	1.110	5	6.480	-3
15	3.350	3	4.320	-3
16	5.830	2	4.680	-3
17	1.010	2	4.680	-3
18	2.900	1	4.500	-3
19	1.010	1	4.320	-3
20	3.060	0	4.140	-3
21	1.120	0	3.960	-3
22	4.140	-1	3.780	-3
	1.000	-2		

TABLE 10 (Continued)

GAMMA-RAY GROUP STRUCTURE, SOURCE SPECTRUM,
AND
FLUX-TO-DOSE CONVERSION FACTORS

Gamma Group	Upper Energy (eV)		Flux-to-Dose Conversion Factors [(mR/hr/g-r/cm²/sec]	
1	1.000	7	9.792	-3
2	8.000	6	8.280	-3
3	6.500	6	6.840	-3
4	5.000	6	5.760	-3
5	4.000	6	4.752	-3
6	3.000	6	3.960	-3
7	2.500	6	3.492	-3
8	2.000	6	2.988	-3
9	1.660	6	2.412	-3
10	1.330	6	1.908	-3
11	1.000	6	1.602	-3
12	8.000	5	1.260	-3
13	6.000	5	9.216	-4
14	4.000	5	6.372	-4
15	3.000	5	4.392	-4
16	2.000	5	2.376	-4
17	1.000	5	1.404	-4
18	5.000	4	3.024	-4
	1.000	4		

TABLE 11

QUADRATURE COEFFICIENTS USED IN ANISN

S_4 Quadrature

Cosine	Weight
-1.0	0
-0.8535534	0.1464466
-0.3535534	0.3535534
+0.3535534	0.3535534
+0.8535534	0.1464466

S_8 Quadrature

Cosine	Weight
-0.9759	0
-0.9511897	0.0604938
-0.7867958	0.0907407
-0.5773503	0.1370371
-0.2182179	0.2117284
0.2182179	0.2117284
0.5773503	0.1370371
0.7867958	0.0907407
0.9511897	0.0604938

TABLE 12

Shield Material	2 r/hour cm	2 r/hour Weight*	1 r/hour cm	1 r/hour Weight*
Tantalum	9.4	7,865 kg	11.0	10,179 kg
Depleted Uranium	8.0	7,385 kg	12.7	12,452 kg
Steel	23 6	11,166 kg	27.0	13,504 kg

Compound shield

8.5 cm TA - 10 cm graphite

8,477 kg @ 0.9 r/hour

* Weights for Ta, DU and compound shields include .77 cm inner and .13 cm outer steel shells.

APPENDIX I

FLIGHT SUPPORT SYSTEM DETAIL MASS STATEMENTS

R.T. C[illegible]
12/2/80

SUMMARY WEIGHT STATEMENT
FLIGHT SUPPORT SYSTEM FOR SINGLE WASTEBALL

ITEM	WEIGHT (LB)		
	20,400 LB WASTEBALL	25,000 LB WASTEBALL	37,050 LB WASTEBALL
PRIMARY SUPPORT STRUCTURE ASSEMBLY	800	970	1380
SECONDARY STRUCTURES	40	50	80
COOLANT (H_2O) SYSTEM	160	220	340
AVIONICS – STS INTERFACE	50	50	50
THERMAL CONTROL PROVISIONS	15	20	25
WEIGHT MARGIN – 15%	160	192	282
(DRY WEIGHT)	(1225)	(1502)	(2157)
HELIUM	5	8	13
COOLANT – H_2O	3020	4460	7500
(GROSS WEIGHT)	(4250)	(5970)	(9670)

GROUP WEIGHT STATEMENT
FLIGHT SUPPORT SYSTEM FOR SINGLE 25K WASTEBALL

PRIMARY SUPPORT STRUCTURE ASSEMBLY		970
LONGERON FITTING ASSEMBLY (2)	480	
KEEL FITTING ASSEMBLY	175	
MAIN X-LOAD STRUTS (2)	42	
MAIN Z-LOAD STRUTS (4)	25	
MAIN YZ-LOAD STRUTS (4)	152	
SECONDARY STRUTS	50	
ASSEMBLY & INSTALLATION HARDWARE	46	
SECONDARY STRUCTURES		50
SUPPORT STRUTS – H2O TANKS	40	
UMBILICAL INTERFACES	10	
COOLANT (H2O) SYSTEM		220
H2O TANK ASSEMBLIES (4)	140	
H2O FEED, FILL, DRAIN	20	
HELIUM BOTTLE	40	
He FEED, FILL, VENT	20	
AVIONICS – STS INTERFACE		50
THERMAL CONTROL PROVISIONS		20
WEIGHT MARGIN – 15%		192
DRY WEIGHT		1502 LB
HELIUM		8
COOLANT – H2O		4460
GROSS WEIGHT		5970 LB

SUMMARY WEIGHT STATEMENT
FLIGHT SUPPORT SYSTEM FOR TWIN 37,050 LB WASTEBALLS

ITEM	WEIGHT (LB)	
PRIMARY SUPPORT STRUCTURE ASSEMBLY	2100	S
P/L SUPPORT/TRANSFER CRADLE	300	m
DOCKING RING INSTALLATION - INCL STRUTS	800	M
COOLANT PLUMBING	20	S
AVIONICS - STS INTERFACE	70	*
THERMAL CONTROL PROVISIONS	30	S
WEIGHT MARGIN - ABOVE ITEMS - 15%	500	
(SUBTOTAL WEIGHT)	(3820)	
COOLANT SYSTEM - REMOTELY LOCATED	340	
WEIGHT MARGIN - COOLANT SYSTEM - 15%	50	
(DRY WEIGHT)	(4210)	
PRESSURANT - GHe	13	
COOLANT - H2O	7500	
(GROSS WEIGHT)	(11,723)	

GROUP WEIGHT STATEMENT
FLIGHT SUPPORT SYSTEM FOR TWIN 37,052 LB WASTEBALLS

PRIMARY SUPPORT STRUCTURE ASSEMBLY		2100
MAIN FRAMES (2)	430	
LATERAL BULKHEADS (2)	450	
TRUNNION FITTINGS (4)	400	
KEEL FITTINGS (2)	200	
P/L RETENTION/DEPLOYMENT FTG'S - XZ (2)	200	
P/L RETENTION/DEPLOYMENT FTG'S - Y (2)	100	
PRIMARY STRUTS (4)	100	
BRACE STRUTS (6)	90	
ACTUATOR INSTALLATION - P/L CRADLE ROTATION (2)	80	
ASSEMBLY & INSTALLATION HARDWARE	50	
P/L SUPPORT/TRANSFER CRADLE		300
SUPPORT TRUSS ASS'Y	240	
TRANSFER RAIL INST'L	60	
DOCKING RING INSTALLATION		800
DOCKING RING ASSEMBLY - ACTIVE HALF	400	
SUPPORT STRUTS - ACTUATOR/ATTENUATOR TYPE (8)	400	
COOLANT PLUMBING		20
AVIONICS - STS INTERFACE		70
THERMAL CONTROL PROVISIONS		30
WEIGHT MARGIN - ABOVE ITEMS - 15%		500
(SUBTOTAL WEIGHT)		(3820 LB)
COOLANT SYSTEM - REMOTELY LOCATED		340
H2O TANK ASSEMBLIES	235	
H2O FEED, FILL, DRAIN	20	
HELIUM BOTTLES	65	
He FEED, FILL, VENT	20	
WEIGHT MARGIN - COOLANT SYSTEM - 15%		50
(DRY WEIGHT)		(4210 LB)
PRESSURANT - GHe		13
COOLANT - H2O		7500
(GROSS WEIGHT)		(11,723 LB)

11/11/60

WEIGHT CALC
(TITANIUM)

<u>PAYLOAD SUPPORT/TRANSFER CRADLE</u> — 300 LB

<u>SUPPORT TRUSS ASSEMBLY</u> — 240 LB

MAIN STRUT TUBES (4) — 60 LB

4" OD TUBES, 0.1 IN. WALL, 75 IN. LENGTH

DIAGONAL STRUT TUBES (8) — 24 LB

2" OD TUBES, .050 IN WALL, 60 IN. LENGTH

DIAGONAL TIE-STRUT TUBES (4) — 8 LB

2" OD TUBES, .050 IN. WALL, 43 IN. LENGTH

VERTICAL TIE-STRUT TUBES (2) — 2 LB

1.5" OD TUBES, .050 IN. WALL, 28 IN. LENGTH

CROSS BEAM BASIC SECTION — 32 LB

X-AREA = 3 IN^2, 66 IN. LENGTH

END FITTING - MAIN STRUT - UPR & LWR (4) — 26 LB

SEE SKETCH, 4 @ 6.5 LB/FTG

END FITTING - MAIN STRUT - FWD & AFT (4) — 32 LB

ESTIMATE 8 LB/FTG

(CONT'D)

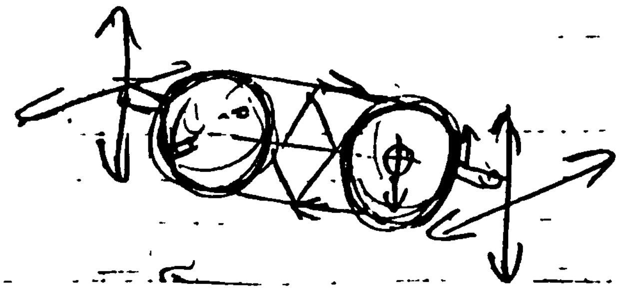
SUPPORT TRUSS ASSEMBLY

MID FITTING – MAIN STRUT – UPR & LWR (2)	20 LB
ESTIMATE 10 LB/FTG	
MID (Y-LOAD / TRUNNION) FTG – MAIN STRUT – FWD & AFT (2)	30 LB
ESTIMATE 15 LB/FTG	
ASSEMBLY & INSTALLATION HARDWARE	6 LB
ESTIMATE 6 LB	

TRANSFER RAIL INSTALLATION — 60 LB

ESTIMATE 60 LB TOTAL

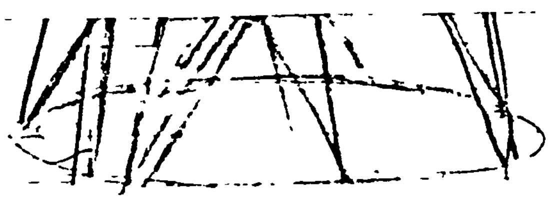
TASK 3: CONCEPTUALIZE OTU PAYLOAD SUPPORT STRUCTURE



R.T. CONRAD
11/6/80

SUMMARY WEIGHT STATEMENT
P/L SUPPORT FRAME

	DUAL PAYLOAD	SINGLE PAYLOAD
BASIC FRAME ASSEMBLY	240	120
PAYLOAD INTERFACE RING ASS'Y-UPR	120	120
PAYLOAD INTERFACE RING ASS'Y-LWR	120	—
PAYLOAD GUIDE RAIL INSTALLATION	50	—
DOCKING RING ASSEMBLY	300	150
MLI INSULATION	20	10
INSTRUMENTATION AND WIRING	20	20
WEIGHT GROWTH MARGIN -15%	130	63
TOTAL	1000 LB	483 lb ≈ 500lb.



11/6, 80

INTERSTAGE ASSEMBLY
CRYO SOIS TO AEROBRAKED OTV

SANDWICH SHELL (484 FT^2, 10 MIL FACE SHTS)	268
LONGITUDINAL STIFFENERS (8)	32
FORWARD RING	50
AFT RING	50
SEPARATION/ATTACH FTG'S - FWD (8)	20
SEPARATION/ATTACH/FTG'S - AFT (8)	20
EXPLOSIVE BOLTS, BOLT CATCHERS, WIRING - FWD	20
EXPLOSIVE BOLTS, BOLT CATCHERS, WIRING - AFT	10
WIRING INSTL, TRAY, ETC - SOIS TO OTV	20
WEIGHT GROWTH MARGIN - 15%	70
	560 LB

End of Document