NASA CR-161,992

A Reproduced Copy

OF

NASA-CR-161992

19820014143

Reproduced for NASA

by the

NASA Scientific and Technical Information Facility

FFNo 672 Aug 65

ANALYSIS OF SPACE SYSTEMS FOR THE SPACE DISPOSAL OF NUCLEAR WASTE FOLLOW-ON STUDY

VOLUME 2

TECHNICAL REPORT

1982

D180-26777-2

Contract NAS8-33847

DPD 609

DR 4

Submitted to
The National Aeronautics and Space Administration
George C. Marshall Space Flight Center
by
Boeing Aerospace Company
Seattle, Washington 98124

(NASA-CR-161992) ANALYSIS OF SPACE SYSTEMS FOR THE SPACE DISPOSAL OF NUCLEAR WASTE FOLLOW-ON STUDY. VOLUME 2: TECHNICAL REPORT Final Report (Boeing Aerospace Co.,

N82-22017

Unclas

N82-22017 #

ANALYSIS OF SPACE SYSTEMS FOR THE SPACE DISPOSAL OF NUCLEAR WASTE FOLLOW-ON STUDY

VOLUME 2

TECHNICAL REPORT

1982

D180-26777-2

Contract NAS8-33847

DPD 609

DR 4

Submitted to
The National Aeronautics and Space Administration
George C. Marshall Space Flight Center
by
Boeing Aerospace Company
Seattle, Washington 981244

FOREWORD

This 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 as a follow-on effort to NASA contract NAS8-33847 from June of 1981 until February of 1982. The objective of the follow-on study was to define the major impacts on the space system concepts selected in the 1980 study that would result from changes in the reference nuclear waste mix from the PW-4b mix used in the 1980 study.

Information developed during the study period is contained in this two-volume final report as listed below:

Volume 1 Executive Summary
Volume 2 Technical Report

Inquiries regarding this study should be addressed to:

C. C. (Pete) Priest
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 following people provided significant contributions to the technical content and preparation of this report in the areas shown:

R. P. Reinert:	Study Manager
J. C. Jenkins:	Orbit Transfer System Performance Evaluations
J. J. Olson:	Flight Support System Concept Development
R. C. Hall and M. L. Jannssen:	SOIS Reliability Studies
B. L. Siegel:	Thermal Analysis
S. McGee:	Documentation Coordination and Technical Editing

Acknowledgment is also made to Mr. William Galloway, the NASA/MSFC contracting officer representative for the bulk of the study, for the guidance and cooperation provided during the technical portion of the study; to his replacement, Mr. Claude C. (Pete) Priest, for valuable support during preparation of the final reports; and to Robert F. Nixon of the MSFC staff. Also, special thanks must go to Ralph E. Best, Manager of Alternative Concepts at the Department of Energy (DOE) Office of Nuclear Waste Terminal Storage Integration (ONI), 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.

CONTENTS

Page

Page

Page

FIGURES

Page

1.0 INTRODUCTION

Since 1970 a number of concepts for space systems for nuclear waste disposal have been studied and evaluated. This study has built upon the results of the 1980 MSFC/BAC study of space disposal to identify the effects of alternative waste mixes on the space systems identified for the reference PW-4b mix in the 1980 study. 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 to mined geological repositories. This introduction provides a brief overview of the study background, scope, objective, guidelines and assumptions, and contents.

1.1 BACKGROUND

NASA and DOE are conducting a sustaining level assessment of nuclear waste disposal in space. The 1980 study of space systems for disposal of nuclear wastes (contract NAS8-33847) investigated reasonable alternative space system concepts (space transportation systems, payload protection systems, and space destinations) to dispose of the reference nuclear waste (Purex PW-4b waste mix in cermet form) used in previous MSFC studies. The output of this study resulted in selection of several alternative space system concepts with high merit for further indepth study and evaluation.

The follow-on effort described in this report emphasized the determination of the effects of variations in the waste mix on the space systems concept to allow determination of the space systems effect on total system risk benefits when used as a complement to the DOE reference mined geological repository.

1.2 OBJECTIVES

The overall objective of the current NASA/DOE joint space system study is to determine if the space option for disposal of nuclear waste can be used to decrease the overall risk associated with disposal of high-level radioactive waste in the reference DOE mined geological repository.

The objective of this study was to determine the effect of alternative waste mixes on the space systems defined in the 1980 MSFC study. The study was conducted in association with parallel studies performed by Battelle Columbus Laboratories (BCL) and Battelle Pacific Northwest Laboratories (PNL). Specific objectives of the Boeing study included (1) determination of the parameters required for characterization of alternative waste mixes or forms and the ranges of these parameters for the waste mixes or forms compatible with space disposal, (2) establishment of waste form categories representing all parameters and the full range of their bounds as determined in task 1, (3) assessment of the effect of the representative waste form categories on the space systems selected in the 1980 study effort and selection of a reference space system approach for each category, and (4) definition of the selected space system.

1.3 SCOPE

The study was conducted during a 9-month contract period (6-1-81 to 2-28-82), divided into a 7-month technical effort followed by 2 months for preparation and delivery of the final report. Maximum use was made of the parallel study efforts conducted by Battelle and of past studies, particularly the 1980 MSFC/BAC study, to focus the resources of this study on its immediate objectives, restricting additional analyses and definition to only those areas specific to the study. Analyses were sufficient to (1) scope the full range of parameters characteristic of alternative waste payloads and (2) assess the impact on alternative space systems to a sufficient depth to allow comparison with the alternative systems defined in the 1980 study in the areas of technical feasibility, safety, performance, and reliability.

1.4 GUIDELINES AND ASSUMPTIONS

General guidelines and assumptions used in the study are as follows:

1. Maximum use was made of past studies and other associated data as appropriate.
2. The reference concept for nuclear waste disposal in space was based on the concepts recommended in the 1980 space systems study.
3. The waste mixes and forms to be analyzed were those defined in the DOE Waste Mixes for Space Disposal Study.
4. Waste mix thermal loading was restricted to values low enough to preclude post-burial meltdown.
5. Liquid and powder states for waste forms were not considered.
6. Cost estimates for elements of the space systems are in 1980 dollars.
7. Containment requirements to be used as a starting point for the waste payload systems study are defined in the 1980 space systems study. Requirements were reviewed and appropriate recommendations made during the course of this study.
8. Estimates of waste form quantity for disposal were based on a 4480 MTHM/year rate of HLW generation.
9. Consideration of destinations other than the reference circular heliocentric orbit at 0.85 AU was eliminated.

10. The waste payload primary container thickness was held constant at 223.5 mm for all waste payload configurations.

1.5 CONTENTS

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

Section 2.0 summarizes the study effort to determine space disposal waste mix parameters. Parameters identified and their importance to the space disposal concept are described in section 2.1. Section 2.2 describes the values for the parameters.

Section 3.0 characterizes the waste forms defined in section 2.0 by incorporating manufacturing and other constraints to define the physical configuration of waste forms and to provide a basis for waste payload design in sections 4.0 and 7.0.

Section 4.0 describes the impact of the alternative waste forms on the space system. Primary impacted areas include the waste payload system, launch system, and orbit transfer system. The effort concentrated on defining space systems suitable for the disposal of waste forms whose quantities required relatively low launch rates (less than five launches per year).

Section 5.0 summarizes the rationale for selection of the reference space system described in sections 6.0, 7.0, and 8.0 and provides an overview of the system elements and operation.

Section 6.0 describes the trajectories and performance requirements for the reference space system delivery mission and details the mission sequence for the nominal rescue mission.

Section 7.0 describes in detail the elements of the reference space system, including the waste payload system, launch site facilities, launch vehicle systems, orbit transfer system, and flight support system.

Section 8.0 describes the operations of the reference space system, including launch site operations, flight operations for the nominal delivery mission, and flight operations for the deep-space rescue mission.

Sections 9.0 and 10.0 summarize the conclusions and recommendations resulting from the study.

2.0 DETERMINATION OF THE CHARACTERISTICS OF WASTE FORMS SUITABLE FOR SPACE DISPOSAL

The objective of this task was to identify characteristics of candidate waste forms relevant to the design of waste payloads for space disposal and to establish the range of values of these characteristics that bound the full range of candidate waste forms.

The approach taken was to use the waste mixes selected by Battelle Northwest Laboratories in a parallel contract. Waste mixes selected include iodine 129 in the form of lead iodide, elemental technetium, and the Oak Ridge National Laboratories cermet high-level waste mix. A survey of past studies indicated that these three mixes represent the full range of waste form parameters specified in past studies. Waste form characteristics were identified in six areas relevant to the space disposal mission. Areas were identified from review of past studies and from the general engineering background of the 1980 studies. Once identified, characteristics were evaluated by a comprehensive literature search conducted by Boeing Aerospace Company. This search provided values for most parameters and identified areas where further research would be required to establish values.

2.1 IDENTIFICATION OF WASTE FORM PARAMETERS

Parameters identified are shown in Figure 2.1-1, with relevant mission areas specified. Parameters were identified in six primary areas: nuclear, strength of materials, mechanical, thermal, manufacturing, and chemical and crystal structure. Parameters identified were evaluated for their relevance to mission areas of risk, flight rate, and waste payload design. Emphasis was placed on identification of parameters relevant to risk and flight rate. Fabrication parameters were identified as a consequence of risk, flight rate, or as required for the level of detail necessary for waste payload concept definition in section 7.0. This effort provided a guide to the relative importance of evaluating the identified parameters.

2.2 EVALUATION OF PARAMETERS

Values of the parameters identified in section 2.1 for the candidate waste forms are shown in Figure 2.2-1. Areas where further research is required to establish parameter values are noted.

AREA	PARAMETER	RELEVANT MISSION AREA		
		RISK	FLIGHT RATE	DESIGN/FAB
NUCLEAR	QUANTITY (Kg /MTHM)		✓	
	TYPE OF NUCLEAR RADIATION EMITTED	✓	✓	✓
	HALF LIFE (YEARS)	✓		
STRENGTH OF MATERIALS	ULTIMATE TENSILE STRENGTH OR MODULUS OF RUPTURE (Pa)	✓		
	YIELD STRENGTH IN COMPRESSION (0.2%OFFSET) (Pa)	✓		
	YIELD STRENGTH IN TENSION (0.2%OFFSET) (Pa)	✓		
	POISSON'S RATIO	✓		
	YOUNG'S MODULUS (Pa)	✓		
MECHANICAL	DENSITY (g/CM)		✓	✓
	MELTING POINT (°C)	✓		
	BOILING POINT (°C)	✓		
	COEFFICIENT OF LINEAR EXPANSION (10^{-6}/°C)			✓
THERMAL	THERMAL LOADING (W/kg)	✓	✓	✓
	THERMAL CONDUCTIVITY $\left(\frac{W}{m\cdot °C}\right)$	✓		✓
	SPECIFIC HEAT (CAL/g.°C)	✓		✓
MANUFAC-TURING	FABRICATION			✓
	HANDLING AND ASSEMBLY			✓
CHEMICAL OR CRYSTAL STRUCTURE	CHEMICAL STABILITY	✓		
	WASTE FORM LOADING $\left(\frac{\text{Kg ISOTOPE}}{\text{Kg WASTE FORM}}\right)$		✓	

Figure 2.1-1. Waste Form Parameters

2.2.1 Nuclear

Primary nuclear parameters include the quantity of each waste mix, radiation emission characteristics, and half-life.

Quantity is expressed in kilograms of waste form per metric ton of heavy metal and ranges from 0.40 kg for the lead iodide waste form to 47.38 kg for the cermet. Quantity is the primary factor in determination of the flight rate for each waste form.

Radiation emission characteristics determine the waste payload shield requirements. Mass of the shield required for each waste form is the second key factor in determining flight rate. Radiation emission for the lead iodide and technetium waste forms is low-energy beta particles at relatively low levels of emission. Effective shielding for these levels of radiation is provided by even minimal containment provisions. The cermet waste form emits high-energy gamma radiation and 13 MeV neutrons. Extensive shielding is required (up to 20 cm of steel and graphite reduces the radiation level to about 0.6 rem/hr at a distance of 1m from the waste form outer surface).

Determination of half-life is required to aid the evaluation of long-term risks. Half-lives for the waste mixes considered range from about 200,000 years for the technetium

ORIGINAL PAGE IS
OF POOR QUALITY

AREA	PARAMETER	WASTE MIX/FORM TYPE: PbI_2^{129}	WASTE MIX/FORM TYPE: Tc^{99}	WASTE MIX/FORM TYPE: MOD. HLW IN ORNL CERMET
NUCLEAR	QUANTITY (kg/MTHM)	0.43	0.75	47.33
	TYPE OF NUCLEAR RADIATION EMITTED	LOW ENERGY β-@ 0.189 mev	LOW ENERGY β-@ 0.292 mev 6.2×10^{8} dis/g-sec	HIGH ENERGY γ n @ 13 mev
	HALF LIFE (YEARS)	1.7×10^{7}	2.12×10^{5}	VARIES
CHEMICAL OR CRYSTAL STRUCTURE	CHEMICAL STABILITY	RELATIVELY INERT	TARNISHES IN MOIST AIR	SOME OXIDATION
	WASTE FORM LOADING (kg ISOTOPE / kg WASTE FORM)	0.56	1.00	0.61
MECHANICAL	DENSITY (g/cm^3) THEORETICAL/NET (1)	6.16 / 5.54 (AS CAST)	11.50/10.93 (BILLET)	6.50/6.50 (BILLET)
	MELTING POINT (°C)	402	2172	1200 (2)
	BOILING POINT (°C)	832	4977	NA
	COEFFICIENT OF LINEAR EXPANSION (10^{-6}/°C)	(3)	(3)	16.0
STRENGTH OF MATERIALS	ULTIMATE TENSILE STRENGTH OR MODULUS OF RUPTURE (Pa)	(3)	(3)	6.9×10^{8} (4)(5)
	YIELD STRENGTH IN COMPRESSION (0.2% OFFSET) (Pa)	(3)	(3)	3.35×10^{8} (4)(5)
	YIELD STRENGTH IN TENSION (0.2% OFFSET) (Pa)	(3)	(3)	3.19×10^{8}
	POISSON'S RATIO	(3)	.309 (4)	0.29-0.33
	YOUNG'S MODULUS (Pa)	(3)	3.22X10[illegible] (4)	1.92X10[illegible] (4)(5)
THERMAL	THERMAL LOADING (W/kg)	8×10^{-5}	1.2×10^{-2}	1.00
	THERMAL CONDUCTIVITY (W/m-°C)	(3)	5.0	9.5
	SPECIFIC HEAT (CAL/g-°C)	0.04	0.058	0.14
MANUFACTURING	HANDLING AND ASSEMBLY	GLOVE BOX	GLOVE BOX	REMOTE ONLY
	FABRICATION	CASTABLE IN PLACE	POWDER METALLURGY UNIAXIAL PRESS SINTER 1.0 DIA./LENGTH 60mm MAX DIM.	POWDER METALLURGY UNIAXIAL PRESS AND SINTER 1.0 DIA./LENGTH 60mm MAX DIM.

NOTES: (1) INCLUDES EFFECT OF POROSITY, VOIDS, INCLUSIONS
(2) FABRICATION TEMPERATURE
(3) DATA UNAVAILABLE IN LITERATURE: RESEARCH REQUIRED
(1) VALUE MEASURED AT ROOM TEMPERATURE
(5) SOLUTION HEAT TREATED

Figure 2.2-1. Parameter Values for Waste Mixes and Forms

to 1.7 x 10^7 years for the iodine 129. The cermet waste form contains a variety of active waste oxides with a wide range of half-lives.

2.2.2 Chemical or Crystal Structure

Chemical and structural parameters include chemical stability and waste form loading. All three candidate waste forms are relatively inert. At room temperature, technetium will tarnish slowly in moist air; cermet, in effect, rusts. The primary container serves as a corrosion barrier for all waste forms considered.

Waste form loading, in terms of kilograms of waste mix per kilogram of waste form, ranges from 0.56 kg for the lead iodide to 1.00 kg for the elemental technetium.

2.2.3 Mechanical

Waste form mechanical properties are important for determining flight rate (density and accident effects), risk level (melting point and boiling point), and waste payload design (coefficient of linear expansion). Values for these candidate waste form parameters are tabulated in Figure 2.2-1. The coefficients of linear expansion for lead iodide and technetium were not found in the literature search and may have to be determined by further research.

2.2.4 Strength of Materials

These properties include ultimate tensile strength or modulus of rupture, yield strength in compression, yield strength in tension, Poisson's ratio, and Young's modulus. These variables are useful primarily in the evaluation of accident effects.

Values are shown in Figure 2.2-1. Values of all properties for lead iodide and of strength properties for technetium were not found in the literature and may require further research to determine. Values for cermet were obtained by using the analogous characteristics of the essentially similar superalloy hastalloy-G. Further refinement of cermet properties would require a dedicated research effort.

2.2.5 Thermal

Thermal properties can impose limits on maximum waste payload size due to contraints imposed by passive radiative heat dissipation in the space environment and by post-burial meltdown.

Waste form thermal loading governs the amount of heat to be dissipated. Values range from 8×10^{-5} W/kg for lead iodide to about 1 W/kg for the cermet.

Thermal conductivity governs the center temperature of the waste form. All waste forms have a high enough thermal conductivity to prevent exceeding waste form temperature limits at the center of the waste form based on radiative heat dissipation at the space disposal destination.

Specific heat is important for calculation of transient temperature response during evaluation of action conditions. Values range from a low of 0.04 for lead iodide to approximately 0.14 for the cermet waste form.

2.2.6 Manufacturing

Handling and assembly constraints imposed by radiation and fabrication characteristics govern the waste form manufacturing processes and assembly of the waste payload. Radiation emission from iodine 129 and technetium are low enough to allow glovebox handling. The high gamma and neutron radiation levels of the cermet waste form require remote handling at all stages of waste form fabrication and assembly.

Fabrication of the lead iodide waste form can be accomplished by casting it in place inside the steel radiation shield and primary container. This is made possible by the relatively low melting temperature of 402°C. The refractory nature of technetium and the cermet requires fabrication by powder metallurgy processes using uniaxial press and sintering techniques.

3.0 CHARACTERIZATION OF ALTERNATIVE WASTE FORMS

The objective of this section is to take the waste forms defined in section 2.0 and incorporate manufacturing and other constraints to define the physical configuration for each waste form to be used as a basis for waste payload design in sections 4.0 and 7.0.

The approach used was to coordinate with Battelle Northwest Laboratories in a parallel contract to determine fabrication constraints applicable to selected waste forms, allowing definition of candidate waste form configurations for use in the waste payload design effort. Two basic configurations were defined for the three candidate waste forms, illustrated in Figure 3.0-1.

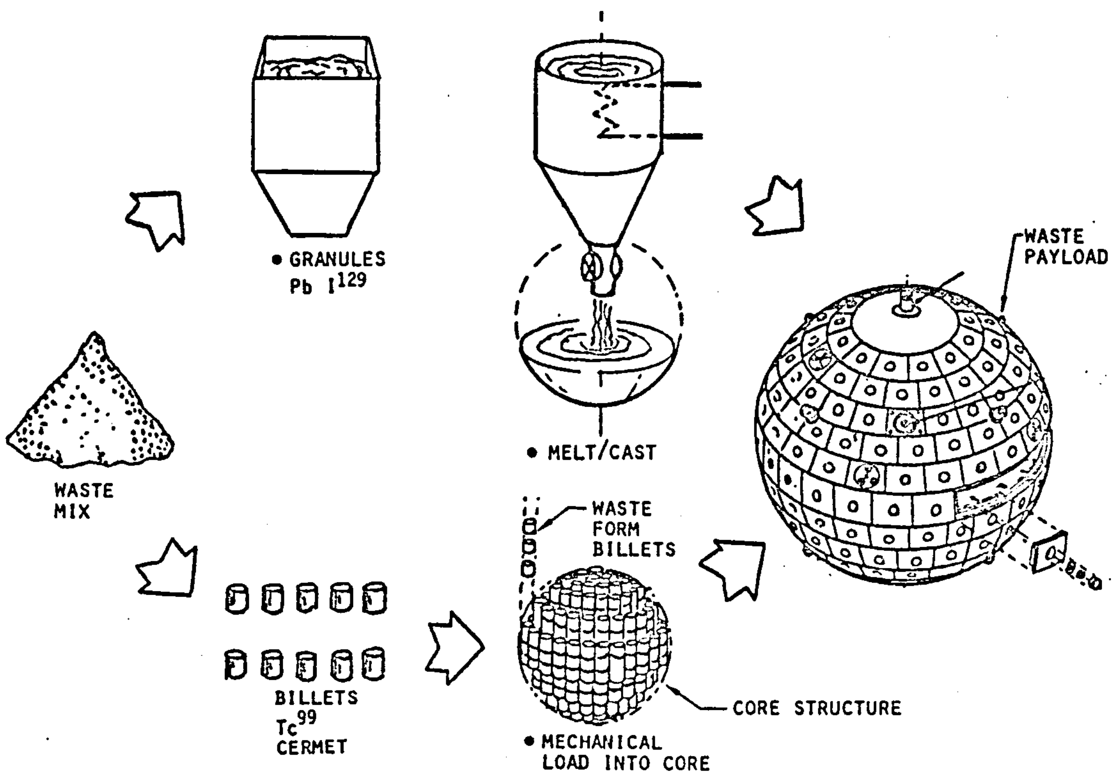


Figure 3.0-1. Waste Form Configurations

The technetium and cermet waste forms are fabricated as right cylindrical billets with height equal to diameter. Corners are rounded to accommodate the uniaxial press and sintering process. Size of the individual billets is limited by constraints imposed by the press and sintering fabrication process to approximately 50 mm maximum dimension (height or diameter). Several thousand of the technetium or cermet billets would be stacked in a hexagonal close-packed array to provide maximum volumetric efficiency in packing the spherical radiation shield and primary container. Both the exact size and the total number of billets are selected as functions of the payload size to maximize packing density.

The lead iodide waste form used for disposal of iodine 129 would be melted and cast in place within the spherical radiation shield and primary container to yield a monolithic spherical waste form. Although, theoretically, 100% volumetric efficiency could be approached using this method, a more conservative figure of 90% was assumed to allow for voids and shrinkage during the casting process.

4.0 ASSESSMENT OF ALTERNATIVE WASTE MIX FORM IMPACT ON SPACE SYSTEMS

Primary space system areas impacted by the adaptation of alternative waste mixes include space disposal destinations and the trajectories required to reach them, the waste payload system and systems for the protection of the waste payload, and launch and orbit transfer systems as a consequence of reduced launch system capacity and launch rate.

Flight support systems were treated parametrically using data from the 1980 study. Waste payload protection systems were deemphasized as the result of the decision at the first working group meeting to use the orbiter as the basis for protection of the waste payload.

The 0.85 AU heliocentric orbit destination was selected as the study baseline as it was the best characterized destination studied and because earlier studies show it satisfies all stability requirements for permanent disposal of nuclear waste in space.

The basic task became the evaluation of the impact on the waste payload system itself and on orbit transfer and launch system components. The evaluation was conducted in three parts. In the first, the impact of alternative waste forms on the waste payload system was determined. Parametric expressions were developed for the net mass of waste form delivered as a function of waste payload system total mass for each of the three alternative waste forms. In the second, the capability of launch systems optimized for low launch rates (less than five launches per year) was determined. In the third, candidate orbit transfer systems were characterized parametrically and matched to the launch systems and waste payload systems to determine the most effective system in terms of net mass of waste form delivered per mission for low-launch-rate scenarios.

4.1 ALTERNATIVE WASTE MIX/FORM IMPACT ON WASTE PAYLOAD SYSTEM CONCEPTS

4.1.1 Candidate Waste Payload Configurations

Candidate waste payload configurations developed to accommodate the waste forms defined in section 3.0 are shown in Figure 4.1-1.

Both configurations use the shield concept developed in the MSFC 1980 study. The shield assembly is the primary barrier against release of the waste form and encases the core and waste form billets inside a seamless shell of Inconel 625 superalloy, 224 mm (8.8 in.) thick. This shell is further protected by a layer of graphite in the form of 228 interlocking tiles, 50 mm (1.97 in.) thick, and a final outer steel shell, 4.3 mm (0.19 in.) thick.

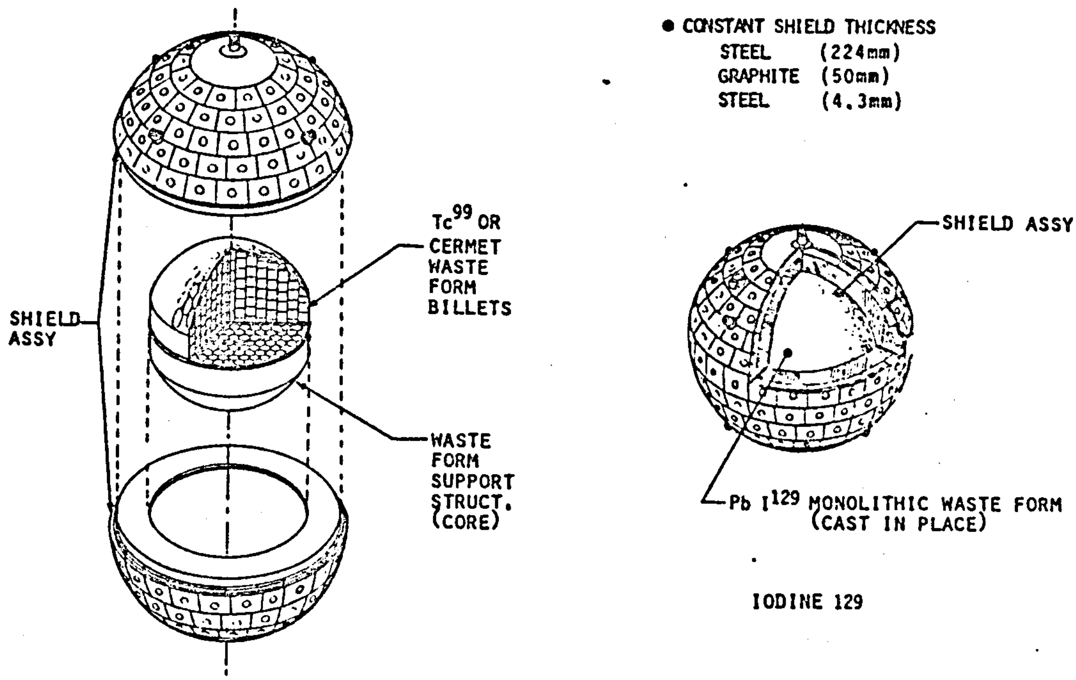


Figure 4.1-1. Waste Payload Configurations

Technetium or cermet waste form billets are stacked in bores drilled in a solid stainless steel waste form support structure or core. The shield assembly is fabricated in two halves, which are assembled around the core and electron beam welded into a single seamless unit.

In contrast, the iodine 129 waste form is cast in place inside an assembled spherical shield. The molten lead iodide is poured in through a small aperture which is welded shut following the casting operation. Closeout tiles are installed over the welded plug in the metal shield.

Parametric mass breakdowns for the three reference waste mixes are shown in Figures 4.1-2 through 4.1-4. Each figure presents the mass of the shield assembly, waste form billets, and, if used, the waste form support structure or core as a function of gross mass of the waste payload.

4.1.2 Cermet Waste Payload Characteristics

Figure 4.1-2 shows the mass breakdown of the cermet waste payload. The relatively high proportion of the waste form mass devoted to the shield assembly is apparent, as is the increased volumetric efficiency realized in the waste payload with increasing total

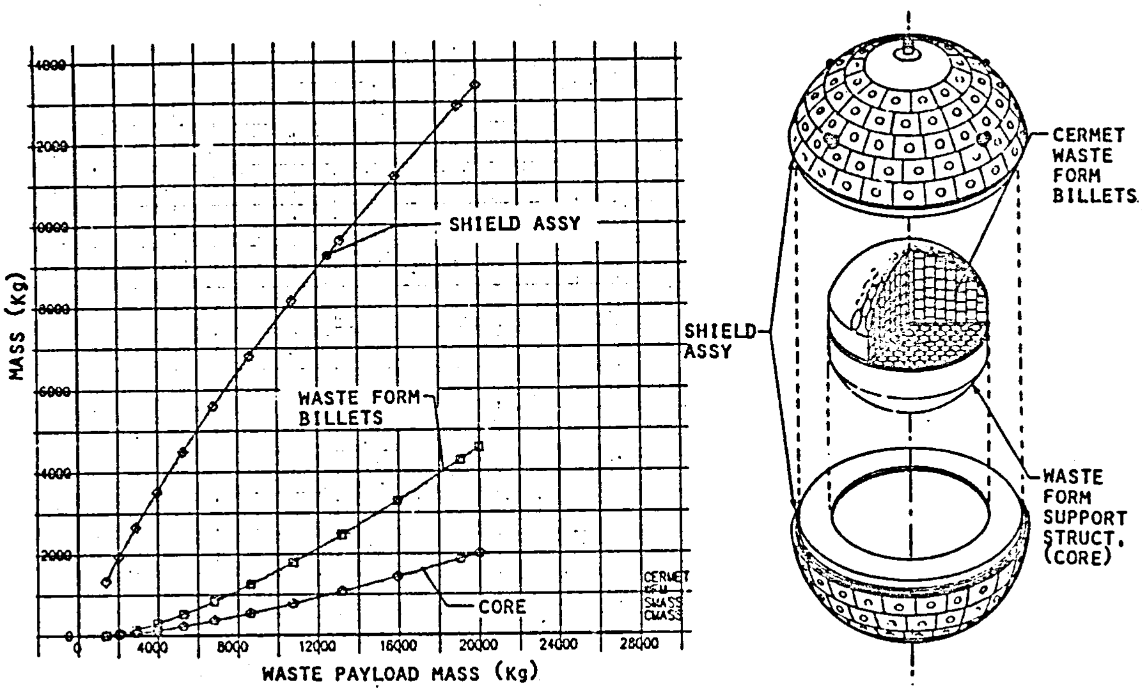


Figure 4.1-2. Waste Payload Parametric Mass Breakdown—Cermet

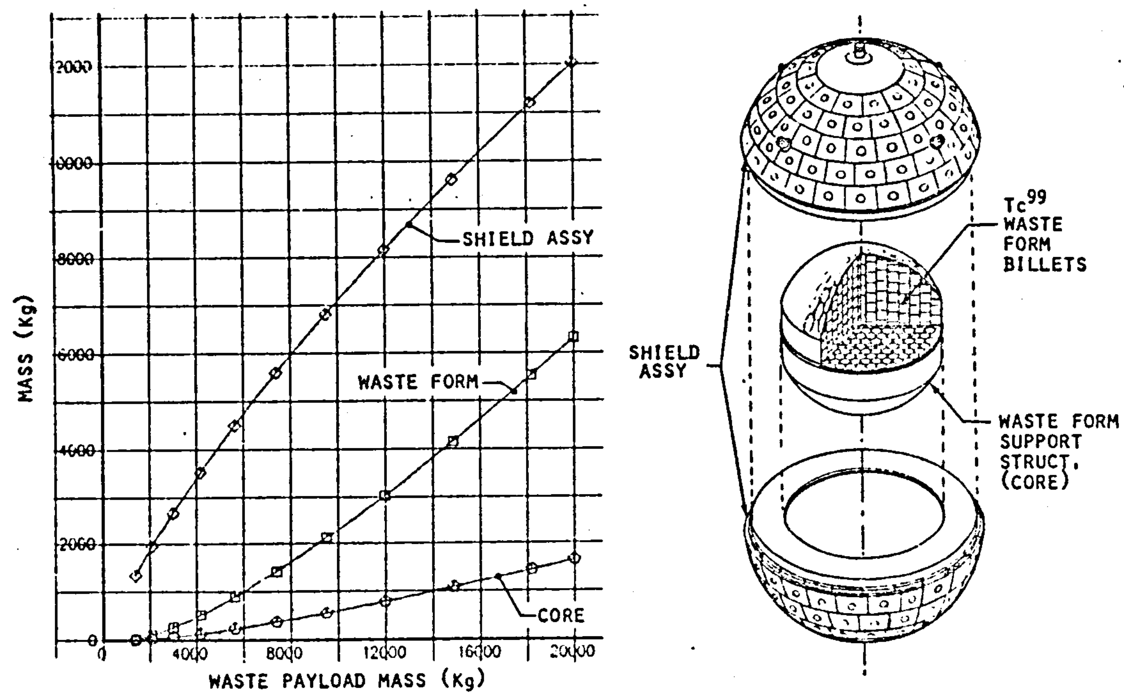


Figure 4.1-3. Waste Payload Parametric Mass Breakdown—Technetium 99

mass. The strong effect of volumetric efficiency on the net mass of waste form delivered in each waste payload for the cermet is apparent.

4.1.3 Technetium Waste Payload Characteristics

Figure 4.1-3 presents the mass breakdown for the technetium 99 waste form. The increased mass of waste form for a given waste payload can be seen as can the relatively reduced impact of the mass of the core. The increased efficiency is due to the high density of the technetium 99 elemental waste form.

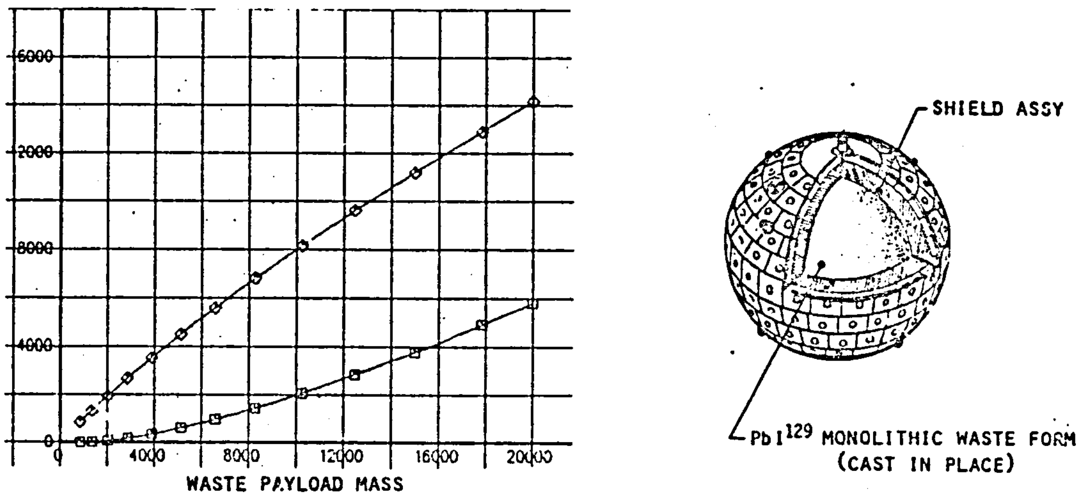


Figure 4.1-4. Waste Payload Parametric Mass Breakdown—Lead Iodide (PbI_2^{129})

4.1.4 Iodine 129 Waste Payload Characteristics

Figure 4.1-4 shows the mass breakdown for the lead iodide waste form. Because of the monolithic cast nature of the waste form, no mass is required for a waste form support structure. The effect of economy of scale on packaging efficiency can be seen.

4.1.5 Parametric Characterization of Candidate Waste Payload System Masses

The ratio of total waste payload mass to the mass of waste form delivered is compared for the three candidate waste forms in Figure 4.1-5.

Technetium 99 shows the most favorable ratio due to its high density. The lead iodide is the second most efficient due to the high volumetric efficiency of the cast-in-place method of waste payload fabrication. The relatively low density of the cermet was the least efficient in terms of packaging of the three waste forms considered.

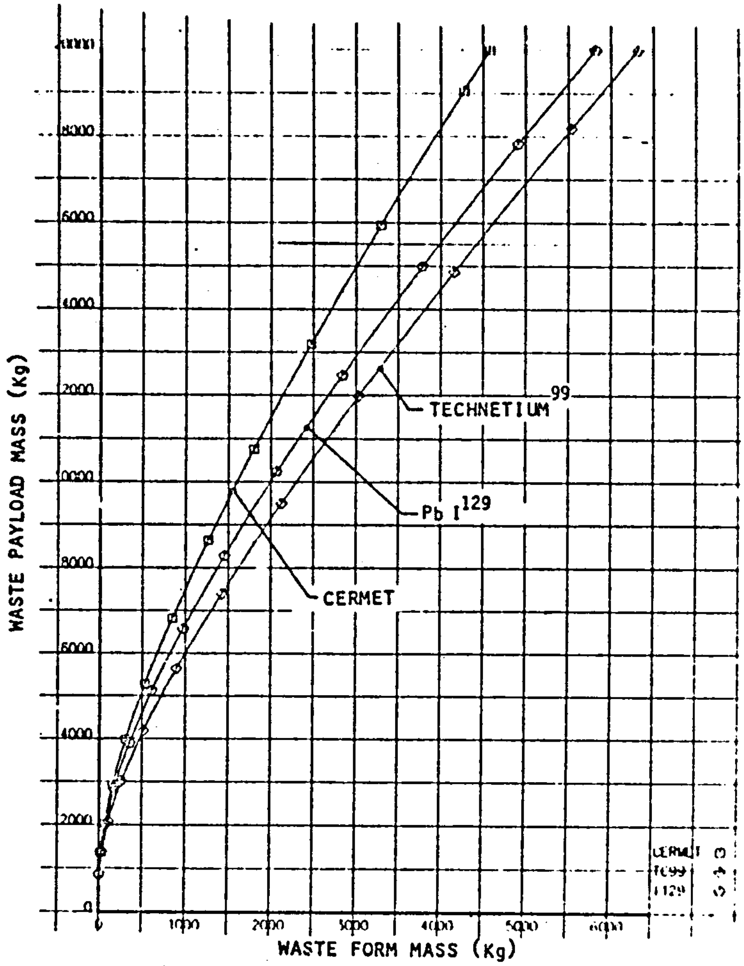


Figure 4.1-5. Complete Waste Payload Mass Versus Waste Form Mass

These parametric characterizations, relating waste form mass delivered to gross mass of the waste payload, served as the basis for total space system performance estimates in section 4.4.

4.1.6 Cermet Waste Payload Thermal Analysis

A thermal analysis of the cermet waste payload was conducted with two objectives: (1) to determine that core melting did not occur with the reference waste form thermal loading in the destination orbit at 0.85 AU and (2) to determine the maximum waste form thermal loading compatible with avoidance of core melting using radiative heat dissipation at the reference destination.

Thermal modeling and analysis of the waste payload were performed for the reference destination orbit and waste form. An upper limit for the waste form temperature was determined to be about 700°C, well below the 1200°C melting point. Additional analysis showed that a waste form thermal loading of 0.022 W/cm^3 would be required to attain the 1200°C melting temperature. A detailed description of the thermal analysis conducted is in Appendix C.

4.2 LAUNCH SYSTEMS

The objective of the launch systems task was to evaluate the candidate launch systems defined in the 1980 MSFC study to determine the best choice of launch systems for candidate waste forms requiring only two to five launches per year, instead of the 50 to 60 launches per year required by the cermet waste form.

The approach used was to apply the risk criteria used in the 1980 study and to reevaluate the cost effectiveness of the candidates based on the reduced launch rate. The revised cost trades resulted in selection of the following two candidate launch systems:

1. For single launch missions, the existing space transportation system (STS), capable of transporting 29,500 kg (65,000 lb) to low Earth orbit (LEO).
2. For dual launch missions, two STS launches with on-orbit rendezvous.

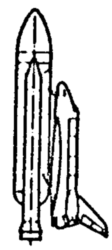
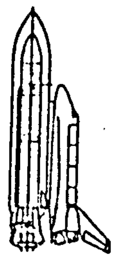
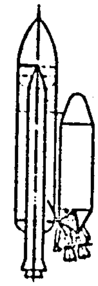
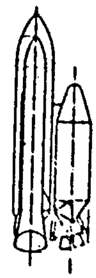
4.2.1 Candidate Launch Systems

Candidate launch vehicles from the 1980 study are illustrated in Figure 4.2-1, along with key characteristics in the areas of risk, cost, and performance. Major elements of each candidate are listed.

Risk is expressed in terms of whether or not the vehicle possesses an intact abort capability. In the event of a malfunction, winged orbiters 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 derived vehicles (SDV) do not have 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 (DDT&E) required to implement the candidate, the production cost per unit, and the cost per flight.

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)	• EXTENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • SOLID ROCKET BOOSTER (2)	• EXTENDABLE 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/L MOD
(4) WHEN DEVELOPED IN ASSOCIATION WITH DEVELOPMENT OF UPRATED SHUTTLE.
(5) TO 270 Km

Figure 4.2-1. Launch Vehicle Characteristics

4.2.2 Selection of Low-Launch-Rate Systems

Launch system selection was accomplished by comparison of life cycle costs (LCC) for the candidate vehicles. 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. The life cycle cost was calculated by multiplying the launch cost from Figure 4.2-1 by the total number of flights required to dispose of the reference cermet waste form. This total was added to the DDT&E cost estimate to derive an estimate of each system's mission life cycle costs.

Figure 4.2-2 compares launch system life cycle costs and shows some of the key assumptions used in their calculation. The ordinate shows an estimate of launch system life cycle cost in billions of dollars. Cumulative mass in thousands of metric tons is plotted on the abscissa, 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 represents DDT&E for initial deployment

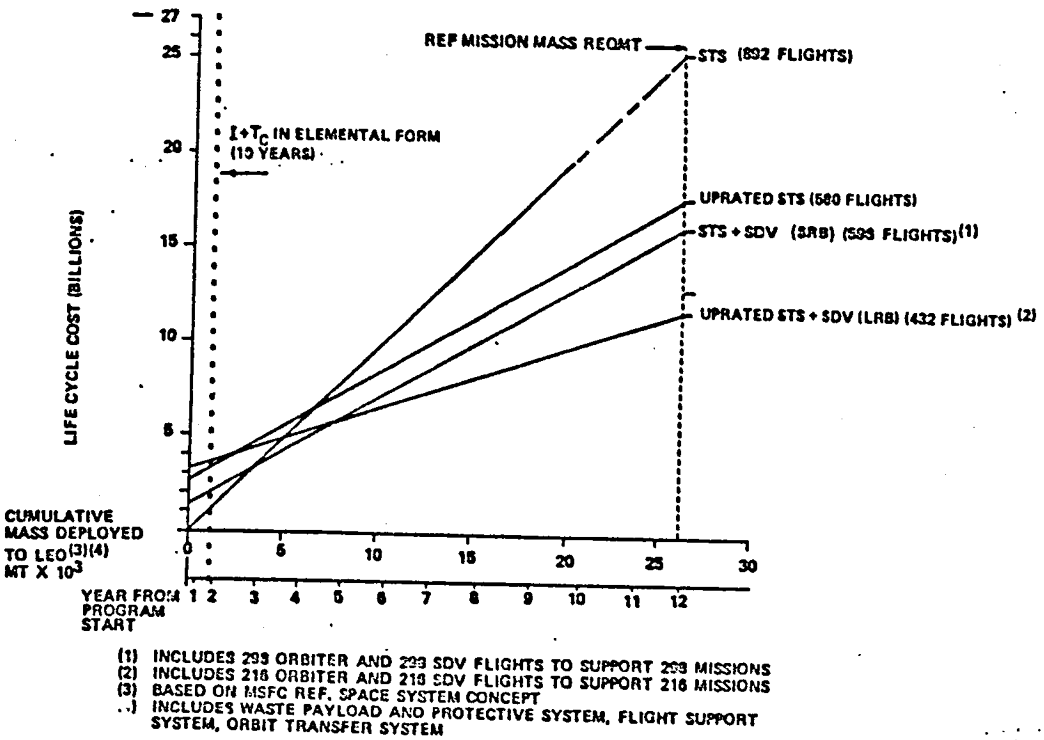


Figure 4.2-2. Launch Vehicle Life Cycle Cost Comparison

of the launch system; values range from zero for the reference shuttle to about $3.2 billion for the uprated shuttle teamed with the liquid rocket booster (LRB) version of the shuttle-derived cargo launch vehicle. The slope of each line is proportional to the cost per flight.

Vertical dotted lines represent the reference cermet mission mass transported to LEO for the reference mission (approximately 27,000t over 10 years) and for the sum of both iodine 129 and technetium (approximately 2,000t over 10 years).

The choice of the most cost-effective launch system for both cermet and Tc 99 plus iodine is apparent. The combination of uprated STS plus SDV with LRB's is the most cost-effective for the launch rate required by the cermet waste payload by approximately $4 billion. At the low launch rate required by the iodine and technetium waste forms, the existing 65K STS is the most cost-effective choice, showing a total cost of $1 billion less than the cost of the next most effective candidate. The cost savings is due to elimination of DDT&E expenses for launch vehicle development, made possible by use of an existing system. The risk advantages of the winged orbiter are retained.

4.2.3 Launch System Performance

Performance characteristics of selected single- and dual-launch systems for space disposal of nuclear waste are summarized in Figure 4.2-3 for both high- and low-launch-rate systems. The performance parameter shown for each option is the net mass at startburn (just after deployment from the launch system) of the entire orbit transfer system, including injection and placement stages and the waste payload. The mass of airborne support equipment (ASE) retained in the orbiter cargo bay has been subtracted from the launch system capacity. These values were used as the basis for total system performance estimates in section 4.4.

WASTE MIX/FORM / UTILIZATION OPTION	HIGH LAUNCH RATE (CERMET/HLW)	LOW LAUNCH RATE PbI^{129} OR Tc^{99}
SINGLE LAUNCH	UPRATED STS: ORBIT TRANSFER SYSTEM MASS AT STARTBURN 32,250 Kg	STANDARD SHUTTLE: ORBIT TRANSFER SYSTEM MASS AT STARTBURN: 26,625 Kg
DUAL LAUNCH	UPRATED STS+SDCLV, LEO RENDEZVOUS. ORBIT TRANSFER SYSTEM MASS AT STARTBURN 81,647 Kg	2 STANDARD SHUTTLES; LEO RENDEZVOUS ORBIT TRANSFER SYSTEM MASS AT STARTBURN 53,678 Kg

Figure 4.2-3. Launch System Performance

4.3 ORBIT TRANSFER SYSTEMS

The objective of the orbit transfer system task was to review the candidate systems identified in the 1980 study in order to define the optimum orbit transfer system for the low-launch-rate STS required for disposal of the iodine and technetium waste forms.

The approach was to apply the new constraints imposed by the revised choice of launch systems to the candidate orbit transfer systems and vehicles identified in the 1980 study. The resulting candidate vehicles, systems, and performance were parametrically characterized.

Five systems were selected as being compatible with the revised launch system constraints, including single-stage systems using cryogenic chemical propellants and electric propulsion, two-stage systems using aerobraking to enable reuse of the injection stage, and use of storable liquid and electric second stages. Performance of all five systems was parametrically characterized for use in total system performance evaluations in section 4.4.

4.3.1 Identification of Candidate Orbit Transfer Systems for Low Launch Rates

The review of candidate orbit transfer systems applied constraints derived from the 1980 study of reuse and from the revised selection of launch vehicles. Reuse options involving all-propulsive chemical propellant injection stages were eliminated for not being cost effective when compared to the aerobraked-return option. Reuse of solar electrical stages (SES) was also determined to be uneconomical.

Use of cryogenic placement stages (CRYO SOIS) was ruled out due to the length constraints imposed by use of the 65K STS without the shuttle-derived vehicle. The 18.3m cargo bay length of the STS is not sufficient to contain both a low-density cryogenic placement stage and an aerobraked injection stage. The far more compact storable propellant and solar electric placement stages can be easily accommodated.

The results of this screening process are illustrated in Figure 4.3-1. Systems identified and their designations include: (1) a single-stage expendable solar electric

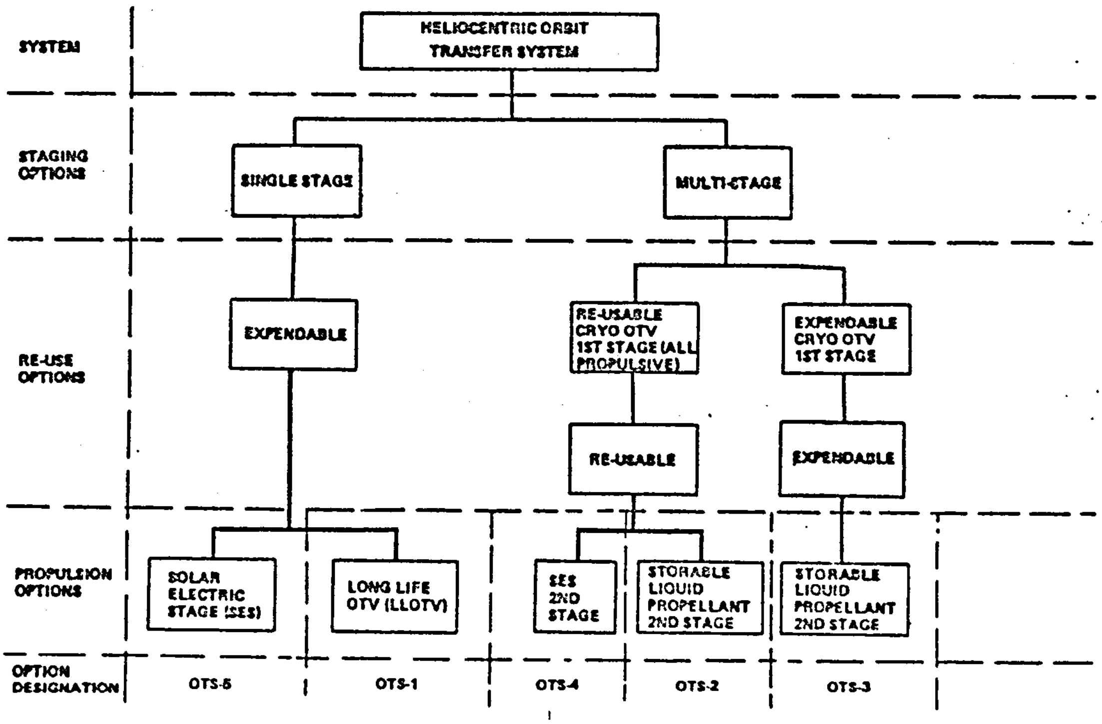


Figure 4.3-1. Orbit Transfer System for Low-Launch-Rate Systems

stage (OTS-5), (2) a single-stage expendable long-life orbital transfer vehicle (OTV) (OTS-1), (3) a multistaged system using a reusable aerobraked cryogenic injection stage and an expendable solar electric propulsion second stage (OTS-4), (4) a multistaged system using a reusable aerobraked cryogenic propellant injection stage with an expendable storable liquid propellant second stage (OTS-2), and (5) a multistaged system using an expendable cryogenic propellant injection stage and an expendable storable liquid propellant second stage for placement (OTS-3).

4.3.2 Definition of Candidate Orbit Transfer System Vehicle Elements

The following vehicles are required for the candidate orbit transfer system options:

1. Injection stages
   a. Reusable, aerobraked, cryogenic injection stage (OTS-2, -4)
   b. Expendable cryogenic injection stage (OTS-3)
2. Placement stages
   a. Storable propellant solar orbit insertion stage (SOIS) (OTS-2, -3)
   b. Solar electric placement stage (OTS-4)
3. Combinations of injection and placement stages
   a. LO_2/LH_2 long-life OTV (OTS-1)
   b. Solar electric stage (OTS-4, -5)

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. Point designs were taken from the 1980 study and are documented there. Parametric mass relationships were extended to cover the smaller propellant loadings required of orbit transfer systems using the existing STS. The resulting point design vehicles are illustrated with some of their key characteristics in Figure 4.3-2.

Parametric relationships for mass at burnout to mass of propellant were generated from the point design mass statements described. The burnout mass versus propellant mass relationships are shown for chemical propellant injection and placement stages in Figures 4.3-3 and 4.3-4, respectively. A series of points were selected from these curves and used for the performance analysis. Point design masses were generated on an individual basis for the two solar electric stages.

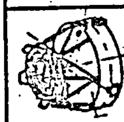
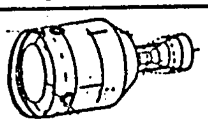
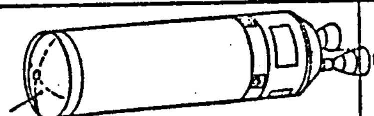
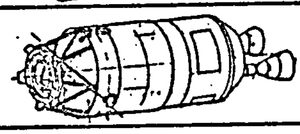
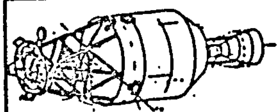
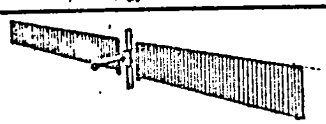
CONFIGURATION	USAGE	PROPELLANT	POINT DESIGN CHARACTERISTICS (Kg) MASS AT BURNOUT	PROPELL-ANT MASS	MASS AT STARTBURN	DIMENSIONS
	PLACEMENT (SOIS)	STORABLE N_2O_4 N_2H_4	1091	3316	4407	4.7 M DIA X 1.52 M LONG
	INJECTION	CRYO LOX LH_2	2931	21,100	24,811	4.7 M DIA X 9.38 M LONG
	INJECTION	CRYO LOX LH_2	8,529	60,135	68,823	4.7 M DIA X 17.7 M LONG
	INJECTION + PLACEMENT (LLOTV)	CRYO LOX LH_2	3,913	28,122	32,735	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 4.3-2. Orbit Transfer System Vehicles and Characteristics

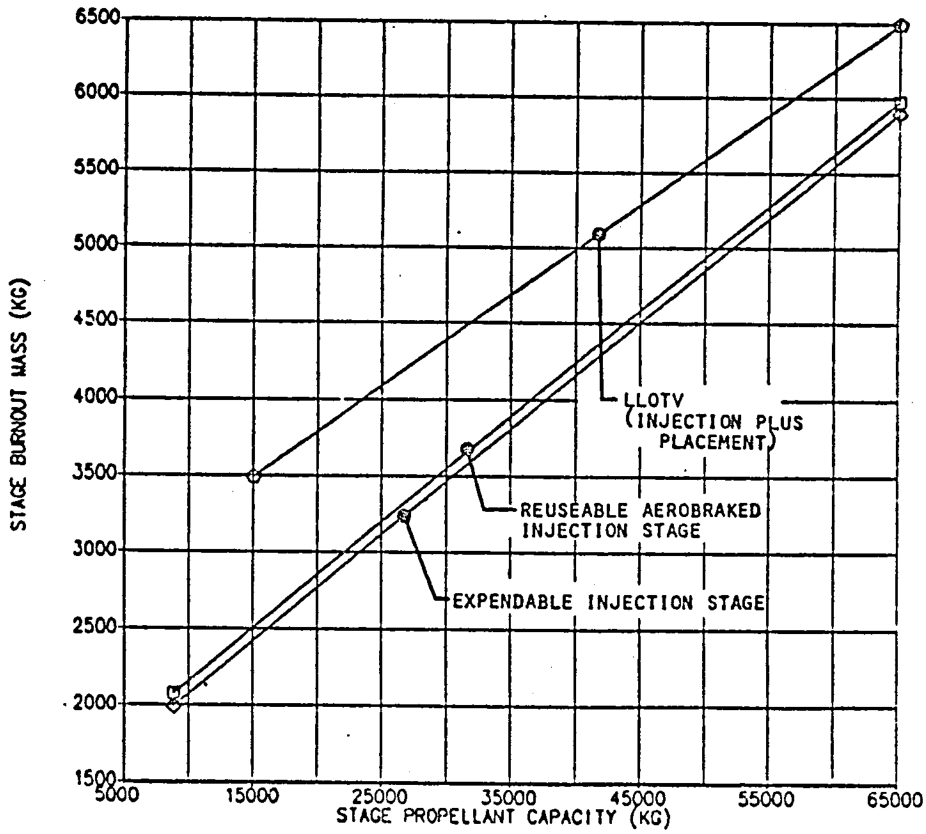


Figure 4.3-3. Parametric Mass Characterization of Injection Stages and LLOTV

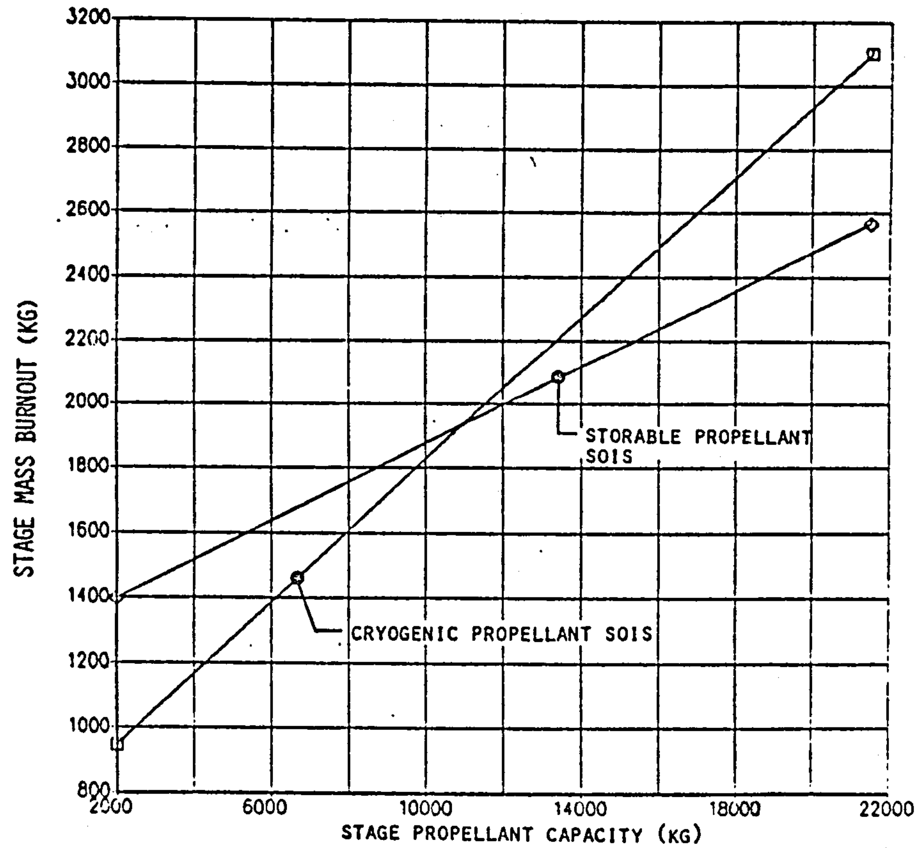


Figure 4.3-4. Parametric Mass Characterization of SOIS Configuration

4.3.3 Parametric Characterization of Candidate Orbit Transfer System Performance

The candidate vehicles described in the previous section were assembled into candidate orbit transfer systems. The following orbit transfer systems were identified in section 4.3.2 for the nuclear waste disposal mission:

1. Solar electric stage (OTS-5)
2. Long-life LO_2/LH_2 OTV (OTS-1)
3. Aerobraked OTV/SES (OTS-4)
4. Aerobraked recoverable OTV/storable SOIS (OTS-2)
5. Expendable OTV/storable SOIS (OTS-3)

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. To obtain these relationships, the vehicle definitions of section 4.3.2 and the mission parameters from the 1980 study (see sec. 6.1) were used as inputs to the Boeing OTV Payload and Sequential Mass Calculation (PSMC) code. 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 reaction control system (RCS) or main engine, and corresponding start-stop losses can be specified. Boiloff and electrical power system 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 interstage is carried by the OTV/SOIS combinations. This is jettisoned by the OTV after injection into heliocentric transfer. A payload adapter mass of 227 kg was added to the SOIS burnout masses. The output of the code provided a parametric characterization of the performance of each candidate system option as illustrated in Figure 4.3-5.

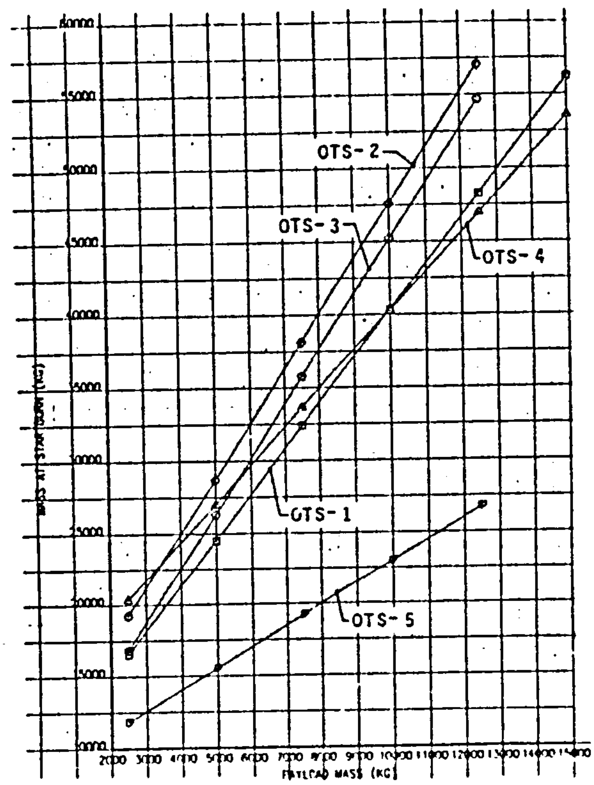


Figure 4.3-5. Orbit Transfer System (OTS) Mass at Startburn Versus Payload Mass to 0.85-AU Destination

Orbit transfer system mass is plotted on the ordinate as a function of delivered payload mass, plotted on the abscissa. Variables include both solar electric and storable propellant SOIS propulsion and two injection stage options: aerobrake return for recovery and expending the injection stage without recovery. Single-stage options include the solar electric stage and the LLOTV. The mass at startburn of each candidate system can be determined for any waste payload mass between 2500 and 15,000 kg.

These performance characteristics, along with those of sections 4.1 and 4.2, provided the basis for the total system performance comparisons in section 4.4.

4.3.4 SOIS Reliability Optimization

A primary input to the concurrent risk analysis being conducted in a separate BCL contract was reliability figures for the orbit transfer system vehicles. Figures were required for the injection stage and SOIS of the reference space system (see sec. 7.0). Reliability values for the injection stage were available from past studies. The SOIS, however, was sufficiently different in the configuration of its avionics system to require a separate analysis.

A preliminary analysis of the SOIS as defined for the 1980 study indicated it has a total mission reliability prediction of 0.901, with the major failure probability occurring during the dormant cruise phase. The failure potential is nearly equally divided among the four major subsystems.

This level of reliability was determined to be inadequate to meet system safety guidelines. Accordingly, a study was conducted to determine the optimum level of redundancy in the major subsystem areas. The analysis traded mission reliability against weight and indicated that SOIS mission reliability can be increased from the original equipment design value of 0.901 to 0.995. The reliability of the optimized SOIS design is limited to the 0.995 region by certain equipment where redundancy is not practical.

Results of the study are summarized in Figure 4.3-6, which illustrates the relationship between the composite mass of key subsystems (primary avionics, propulsion, and RCS) and system reliability. The circled lines represent individual configurations evaluated. Case 1 is a single-string configuration shown as a reference. The vertical dotted line at 1000-kg system mass intersects the reliability curve at a value of 0.91, which represents the reference configuration from the 1980 study. The second vertical dotted line at about 1290 kg represents the selected value for reliability. At 0.9927, it approaches the asymptote of 0.995 imposed by fundamental system limits. Added mass amounts to 318 kg.

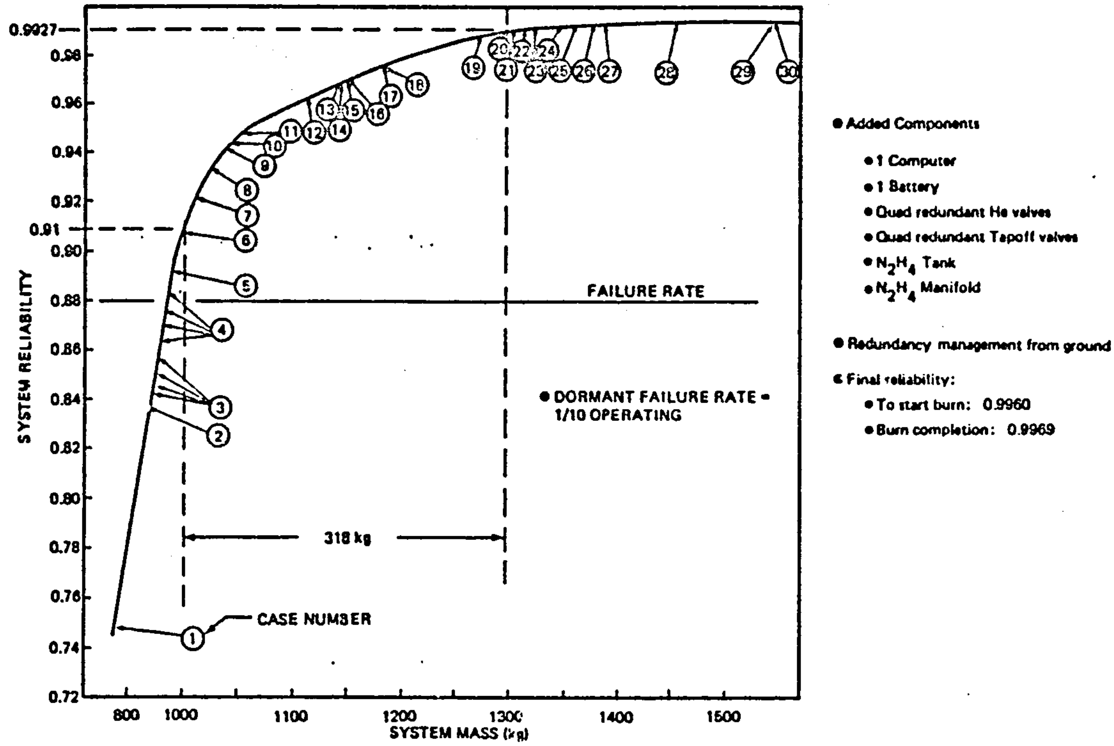


Figure 4.3-6. SOIS Mass and Reliability Optimization Study Results

Added components with reference to the 1980 study configuration are also shown, along with values for the overall stage reliability. Redundancy management is assumed to be handled from the ground.

A detailed description of the methodology and assumptions used in the optimization is contained in Appendix D.

4.4 PARAMETRIC PERFORMANCE CHARACTERIZATION OF LOW-LAUNCH-RATE SYSTEMS FOR THE SPACE DISPOSAL OF NUCLEAR WASTE

The objective of this section was to combine the results of sections 4.1, 4.2, and 4.3 to define the performance of concepts combining launch, orbit transfer, and waste payload systems.

For each launch system, the efficiency of the candidate orbit transfer systems for transport of each candidate waste form was compared, using the parametric mass and performance data from sections 4.1 and 4.2.

Review of the comparison allowed selection of three candidate orbit transfer systems for the single-launch option and two systems for the dual-launch option. These systems are viable candidates for disposal of waste forms requiring low launch rates (equivalent to less than five launches per year of the existing STS).

4.4.1 Candidate Total System Concepts

There are a total of 18 distinct concepts for space disposal of low-launch-rate (Tc^{99} and $Pb\,I_2^{129}$) waste forms. The concepts are formed by combining one of two launch system options (single or dual launch) with one of the five orbit transfer system options, yielding 10 possible space disposal transportation options. One orbit transfer option (use of a single-stage solar electric orbit transfer system) is not compatible with the dual-launch system because of its low demands on paylaod. This yields a total of nine candidate space transportation systems, which are illustrated in Figure 4.4-1, showing the system derivation, characteristics, and designation. Any of the candidate systems can be used to dispose of either of the low-launch-rate waste mixes, yielding a total of 18 concepts.

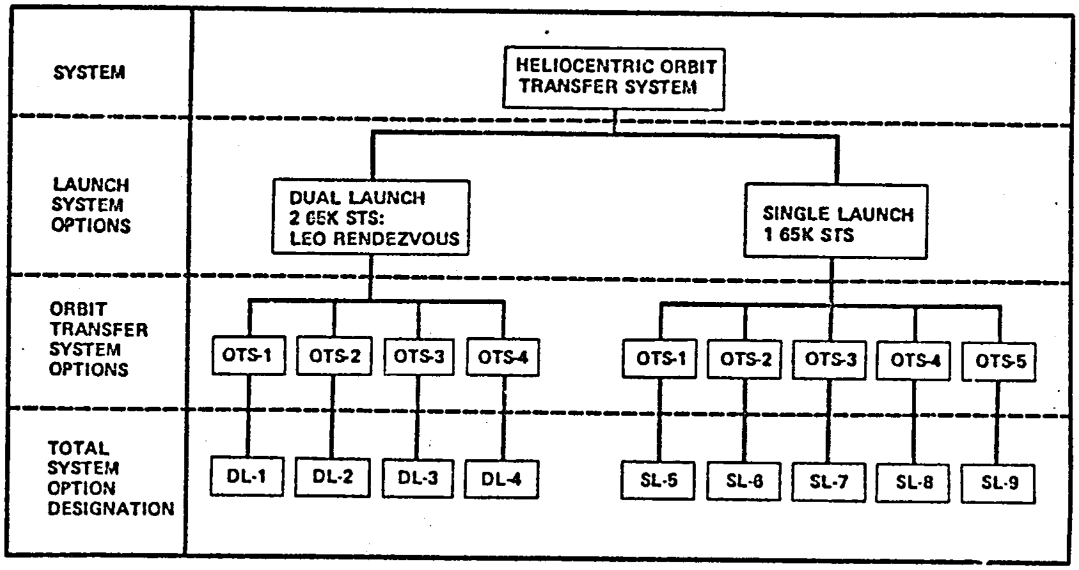


Figure 4.4.1. Identification of Candidate Orbit Transfer Systems for Low Launch Rates

4.4.2 Total System Performance Evaluation

The 18 concepts identified in section 4.4.1 were evaluated for relative performance using a technique which allows direct and simultaneous graphic comparison of total system performance by combining parametric characterization of orbit transfer system performance and waste payload systems.

Figure 4.4-2 summarizes the parametric relationship between launch system, orbit transfer system, and net mass of payload delivered for each candidate waste form. Launch system payloads are shown as vertical dotted lines on the left half of the chart. Candidate orbit transfer system payloads are determined by the intersection of the vertical lines, representing launch system payloads, with the slanted lines, representing orbit transfer system performance.

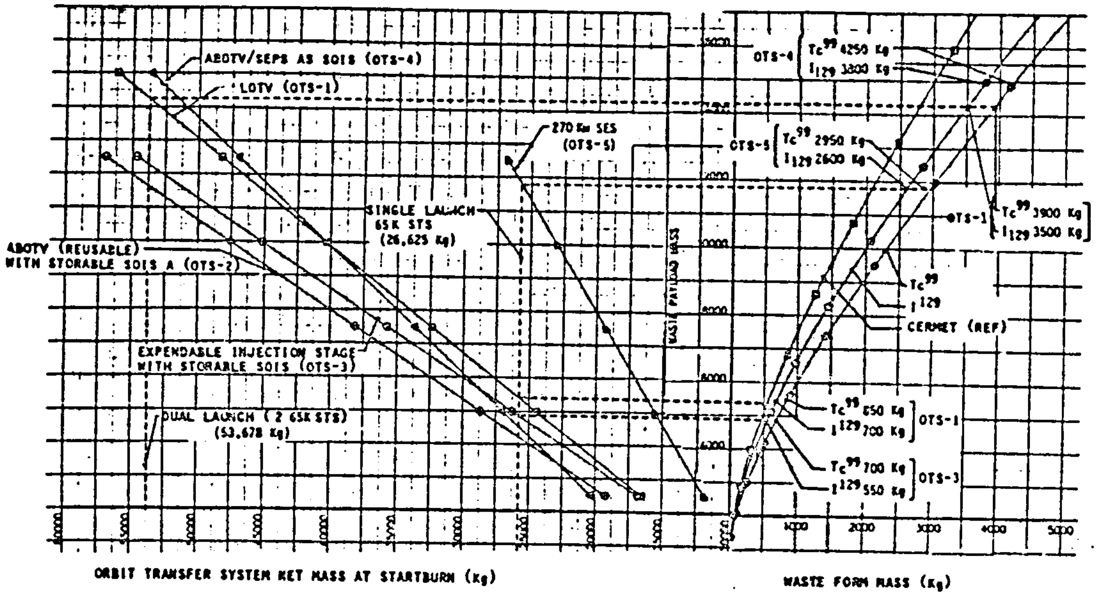


Figure 4.4-2. Parametric Performance Characteristics of Orbit Transfer Systems for Cermet, Pb I_2^{129}, and Tc^{99} Waste Forms

Gross payloads of candidate orbit transfer systems are represented by the horizontal dotted lines which extend into the right half of the plot. Intersection of these lines with the curves representing gross mass of the waste payload to net mass of waste form carried determines the net mass of waste form delivered by each candidate combination of launch system, orbit transfer system, and waste form. Using this technique, performance of any orbit transfer system option can be rapidly evaluated in terms of net waste form delivered per launch.

Typical systems evaluated included the two best chemical propellant options for both single-launch and dual-launch scenarios and the single-stage electric option used in the single-launch mode. Further discrimination of systems would require detailed definition of relative life cycle costs to determine the most cost effective. All candidates are comparable in terms of risk.

Results of the evaluation of candidate systems are summarized in Figure 4.4-3. The performance of low-launch-rate systems using one or two launches of the 65K STS with various orbit transfer systems is shown in terms of waste form mass delivered and equivalent flights per year for the candidate system. Performance of the reference system described in sections 7.0 and 8.0 is shown for reference.

	LAUNCH SYSTEM	ORBIT TRANSFER SYSTEM CODE *	TOTAL SYSTEM OPTION	NET PAYLOAD MASS/MISSION (Kg)	WASTE FORM MASS DELIVERED PER MISSION (Kg) (EQUIVALENT FLIGHT RATE, FLTS/YR)		
					Pb I_2^{129} (1,600 $\frac{Kg}{YR}$)	Tc^{99} (3,000 $\frac{Kg}{YR}$)	CERMET (189,520 $\frac{Kg}{YR}$)
LOW LAUNCH RATE CANDIDATE SYSTEMS	SINGLE LAUNCH 1 65K STS PER MISSION	OTS-1	SL-5	5,400	700 (2)	850 (4)	550 (349)
		OTS-3	SL-7	4,900	550 (3)	700 (4)	450 (421)
		OTS-5	SL-9	11,800	2,600 (3FLTS ea. 2 YRS)	2,950 (1)	2100 (90)
	DUAL LAUNCH 2 65K STS PER MISSION	OTS-1	DL-1	14,200	3,500 (1 FLT ea.2 YRS)	3,900 (4 FLTS ea.5 YRS)	2200 (68)
		OTS-4	DL-4	15,000	3,800 (1 FLT ea 2 YRS)	4250 (2 FLTS ea.3YRS)	3000 (63)
REFERENCE SYSTEM	DUAL LAUNCH 1 UPRATED STS 1 SDCLV (LRB) PER MISSION	OTS-6	REF SYSTEM	30,614	7,850 (1 FLT ea.5 YRS)	8,700 (1 FLT ea.3YRS)	6000 ([illegible]2)

*KEY TO ORBIT TRANSFER SYSTEM CODES.	
OTS 1	SINGLE STAGE CRYOGENIC LONG LIFE OTV; EXPENDABLE
OTS 3	2 STAGE: EXPENDABLE CRYOGENIC INJECTION STAGE; STORABLE SOIS
OTS 4	2 STAGE: REUSABLE CRYOGENIC AEROBRAKED INJECTION STAGE; EXPENDABLE SOLAR ELECTRIC STAGE AS SOIS
OTS 5	SINGLE STAGE 270 Kw SOLAR ELECTRIC; EXPENDABLE
REFERENCE SYSTEM → OTS 6	2 STAGE: REUSABLE CRYOGENIC AEROBRAKED INJECTION STAGE; EXPENDABLE CRYOGENIC SOIS

Figure 4.4-3. *Performance Summary—Low-Launch-Rate Space Transportation Systems for Alternative Waste Mixes*

5.0 REFERENCE SPACE SYSTEMS SELECTION AND OVERVIEW

This section summarizes the rationale for selecting the reference space system, along with an overview of system elements and operation. More detailed information on system elements and operation is contained in sections 6.0, 7.0, and 8.0.

5.1 REFERENCE SPACE SYSTEM SELECTION

The reference space system was selected at a joint working group meeting in August of 1981, between Boeing Aerospace Company, Battelle Northwest Laboratories, Battelle Columbus Laboratories, and the Marshall Space Flight Center. The selected waste mix is the cermet high-level waste mix, with 95% of the cesium and strontium removed, as developed by Battelle Northwest Laboratories. This waste mix was the only one of the three considered which showed the potential for long-term risk reductions when compared to the reference mined geologic repository.

The space system used to transport the reference waste mix from the launch site to the 0.85 AU heliocentric orbit destination was selected from the candidates recommended at the conclusion of the 1980 MSFC Space Disposal Study. The system selected combines the lowest risk of any concept considered with the highest performance of the recommended systems. Of the four systems recommended for further study at the end of the 1980 effort, the reference system was judged to be most compatible with the direction of ongoing NASA studies of future space transportation systems.

5.2 REFERENCE SPACE SYSTEM OVERVIEW

The following section provides a brief overview of the system elements and operations described in detail in sections 7.0 and 8.0.

5.2.1 System Elements

Major elements of the reference system, shown in Figure 5.2-1, include:

1. The waste payload system, which supports and protects the waste form during ascent and orbit transfer operations.
2. The flight support system, which provides a mechanical interface between the waste payload system and the launch vehicle system and which has provisions for mechanical transfer of the waste payload system to the orbit transfer system in LEO.
3. The launch system, which transports the waste payload system and orbit transfer system from the launch site into a 270-km-altitude low Earth orbit. The

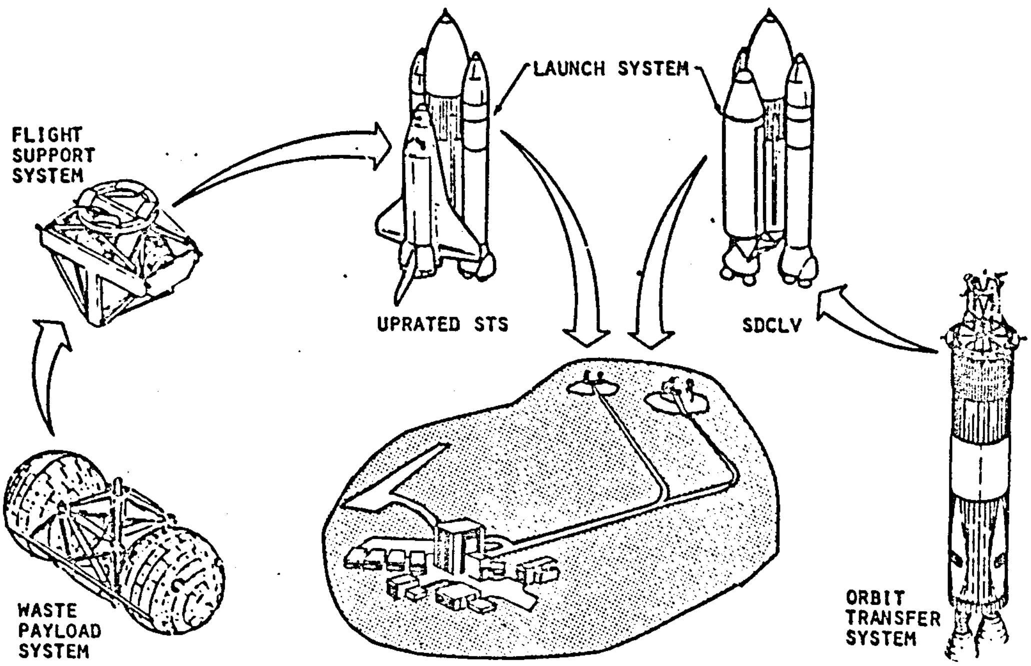


Figure 5.2-1. Reference Space System Major Elements

launch system is composed of two vehicles: one that carries the waste payload and flight support system (FSS) and the other, the orbit transfer system. The waste payload system is carried in an uprated version of the existing STS using liquid rocket boosters. The uprated STS has a payload capacity to LEO of 47,000 kg. The orbit transfer system is carried to LEO in a shuttle-derived cargo launch vehicle which replaces the winged orbiter component of the space transportation system with an expendable cargo shroud and a reusable propulsion and avionics module. The shuttle-derived cargo launch vehicle provides increased internal volume for payload accommodation and has a payload capacity of 84,000 kg.

4. The orbit transfer system, which transports the waste payload from LEO to the destination heliocentric orbit at 0.85 AU. The orbit transfer system is composed of a reusable injection stage and an expendable SOIS. A waste payload adapter on the front of the SOIS allows docking with the orbiter and provides mechanical support for the waste payload during orbit transfer operations.

5. Launch site facilities, which consist of a nuclear payload processing facility (NPPF), for assembly and integration of the waste payload system with the FSS, and the facilities required for turnaround of the launch vehicle systems and the reusable portion of the orbit transfer system.

5.2.2 System Operation

Figure 5.2-2 is a schematic of key mission operations for the reference space system. Key events include:

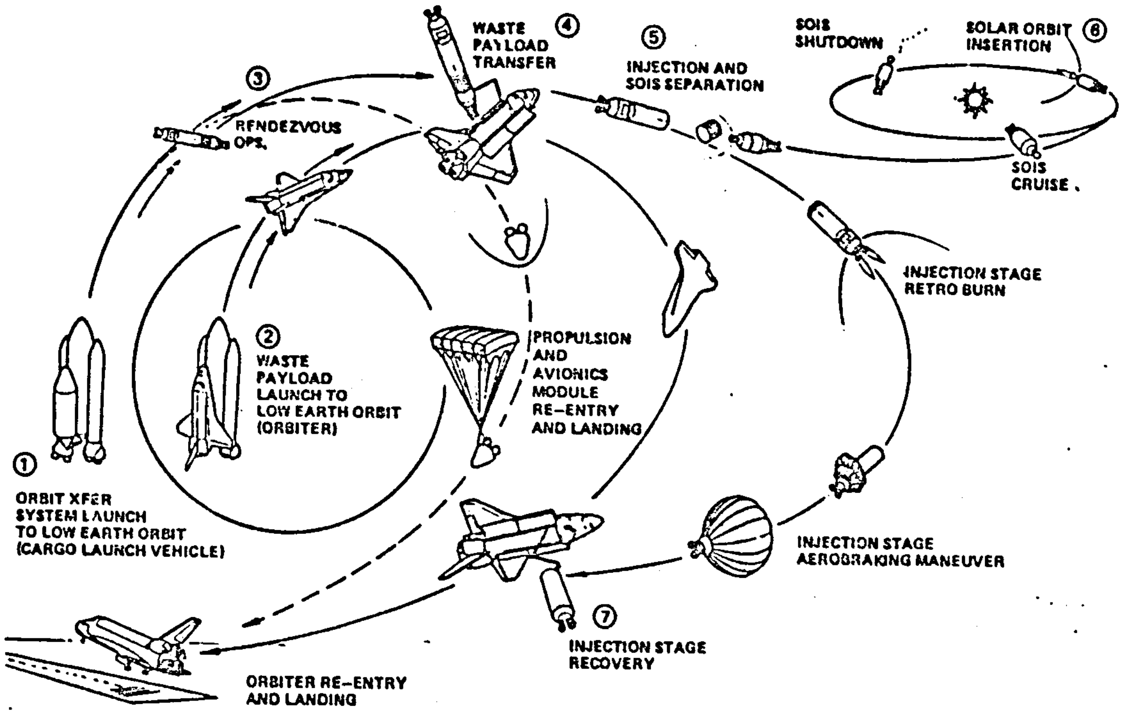


Figure 5.2-2. Reference Space System Mission Operations Summary

1. Launch of the cargo launch vehicle which places the two-stage orbit transfer system into LEO.
2. Launch of the waste payload to LEO in the uprated space shuttle.
3. Rendezvous between the orbit transfer system and the orbiter in LEO.
4. Transfer of the waste payload to the orbit transfer system from the 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 in transfer orbit, injection of the SOIS and the waste payload into the destination heliocentric orbit at 0.85 AU.
7. Recovery of the injection stage for reuse, following a retroburn and aerobraking maneuver which inserts it into LEO.

6.0 REFERENCE SPACE SYSTEM TRAJECTORIES AND PERFORMANCE REQUIREMENTS

Three basic mission profiles are used in the space disposal of nuclear waste: the delivery mission, Earth orbit rescue mission, and the deep-space rescue mission. The delivery mission transports the nuclear waste payload from the launch site to its final destination orbit at 0.85 AU. Rescue missions provide for rendezvous of a rescue vehicle with the waste payload after a delivery mission failure with subsequent transfer of the waste payload to the final destination.

This chapter is divided into three sections. The first describes the mission profiles, trajectory elements, and performance requirements of the delivery mission. The second and third sections describe mission profiles and performance requirements for Earth orbit and deep-space rescue missions, respectively.

6.1 DELIVERY MISSION

The delivery mission takes place in three distinct phases: ascent of the uprated STS, which transports the payload from the launch site to LEO; injection, which places the SOIS and waste payload into the heliocentric transfer orbit; and placement, which leaves the expended SOIS and waste payload in the destination circular orbit.

6.1.1 Uprated STS Ascent Mission Profile

A typical ascent trajectory profile for the uprated shuttle is shown in Figure 6.1-1, which illustrates altitude as a function of range. Major events and time are noted.

The shuttle is launched with the three orbiter space shuttle main engines (SSME) burning in parallel with the two LRB's. The ascent trajectory reaches a maximum dynamic pressure (Q) of 650 lb/ft^2 approximately 60 sec after launch at an altitude of 10,214m. At 108 sec, the total load factor reaches the first stage maximum value of 2.6 g's. LRB separation occurs at approximately 126 sec at an altitude of 45,263m, 51.4 km downrange from the launch site. The LRB's are recovered and returned to the launch site for reuse. After LRB separation, the orbiter continues to ascend, using the three SSME's which provide thrust vector control. The total load factor reaches a maximum value of 3.0 g's (longitudinal) at 415 sec and remains at that value until 470 sec, when the main engine cutoff (MECO) sequence is initiated.

MECO takes place 478 sec after liftoff, when the orbiter has reached an altitude of 110,155m. The external tank (ET) separation occurs at MECO. After a short coasting period, the orbital maneuvering system (OMS) engines are fired at 514 sec to provide the

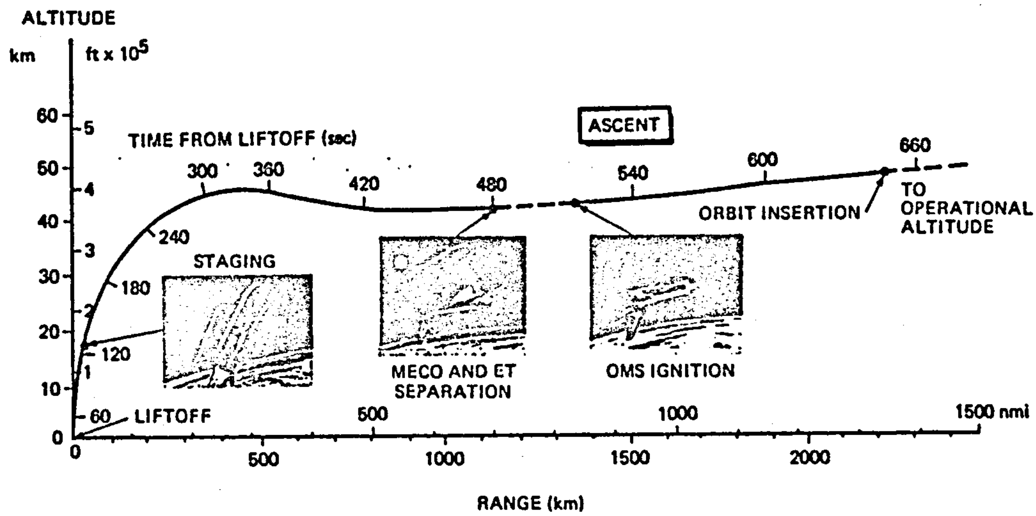


Figure 6.1-1. Uprated STS Ascent Trajectory Schematic

additional velocity needed to insert the orbiter into an elliptical orbit having a minimum apogee of 369 km. The OMS engine cutoff occurs 648 sec after launch at an altitude of 133.9 km, when the orbiter is 299 km from the launch site. An additional OMS burn at apogee is used to circularize at the 370-km orbit altitude. Velocities, ranges, and altitude for major ascent events are shown in Figure 6.1-2.

Event	Time (min)	(sec)	Altitude* (km)	Inertial Velocity** (m/sec)	Range (km)
Ignition (SSME + LRB)		-3.46	0.056	410	0
Liftoff		-3.46	0.056	410	0
Begin pitchover		7.2	0.166	410	0
Maximum dynamic pressure	1	9	13.3	740	6.4
LRB separation	2	4	47.3	1537	38.1
MECO	8	38	117.5	7823	1334
External tank separation	8	50	118.4	7832	1427
OMS-1 ignition	10	39	126	7813	2220
OMS-1 cutoff	12	24	133.9	7894	2995
OMS-2 ignition	43	58	369.0	7615	15731
OMS-2 cutoff	45	34	370.0	7686	16027

• Altitude referenced to orbiter center of gravity above geodetic representation of Earth's surface.

•• Rotational velocity of Earth at KSC latitude of 28.5° N.

Figure 6.1-2. Uprated STS Ascent Trajectory and Events

6.1.2 Shuttle-Derived Cargo Launch Vehicle (SDCLV) Ascent Mission Profile

The SDCLV trajectory profile is essentially identical to that of the uprated STS.

6.1.3 Launch System Coordination

Launch of uprated STS and SDCLV into the same 38-deg inclination orbit, 4 hr apart, minimizes orbit transfer system on-orbit dwelltime and allows two STS contingency launch opportunities within 24 hr of the SDCLV launch.

The SDCLV is launched first using a launch azimuth (referenced to true north) of about 70 deg. This places the SDCLV (and its payload, the orbit transfer system) into a 38-deg inclination orbit. The SDCLV launch takes place as the Earth's rotation places KSC under the 38-deg orbit ground track, as the orbit track overflies KSC to the north.

After 4 hr, the Earth has rotated about 60 deg, bringing KSC under the 38-deg orbit ground track again. The uprated STS is now launched heading south at an azimuth of 110 deg and is inserted into the same 38-deg orbit as the SDCLV, about 400 km behind the SDCLV. This provides the correct initial location for subsequent rendezvous and docking maneuvers. The 4-hr interval between launches allows the SDCLV to complete just under three orbits—sufficient time to allow verification of orbit transfer system status prior to launch commit for the uprated STS which carries the waste payload.

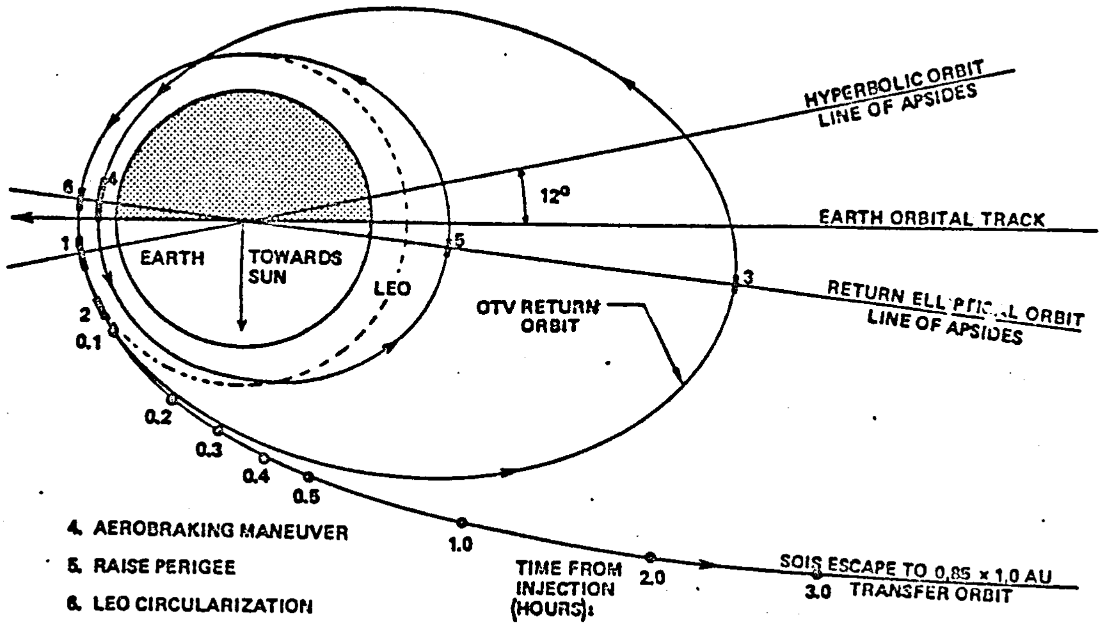
6.1.4 SOIS and Waste Payload Injection Mission Profile

The SOIS and waste payload injection mission profile is illustrated in Figure 6.1-3. The mission consists of two primary phases: injection, which leaves the SOIS and waste payload in the heliocentric transfer orbit, and OTV recovery, which brings the OTV back to LEO for recovery by the shuttle and subsequent reuse.

Injection. The objective of the injection segment of the mission is to achieve the desired aphelion velocity for the transfer from 1 AU to 0.85 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 aphelion velocity. Earth escape is achieved via a hyperbolic orbit. The required hyperbolic velocity at perigee is 11.000 km/sec. Since in LEO, v_c = 7.730 km/sec, the change in velocity required for SOIS injection is 3.270 km/sec.

The geometry of the escape orbit is shown in Figure 6.1-4. Key parameters of the departure hyerbola shown include r_p, the radius at periapsis; e, the orbital eccentricity; a, the semimajor axis; $\emptyset$, the angle between asymptote and line of apsides; and b, the semi-minor axis. Values for these parameters for the reference injection mission are shown in Figure 6.1-4.

1. INJECTION INTO EARTH ESCAPE
2. SOIS STAGING AND OTV RETRO
3. ADJUST PERIGEE



4. AEROBRAKING MANEUVER
5. RAISE PERIGEE
6. LEO CIRCULARIZATION

Figure 6.1-3. Injection Mission Profile Schematic

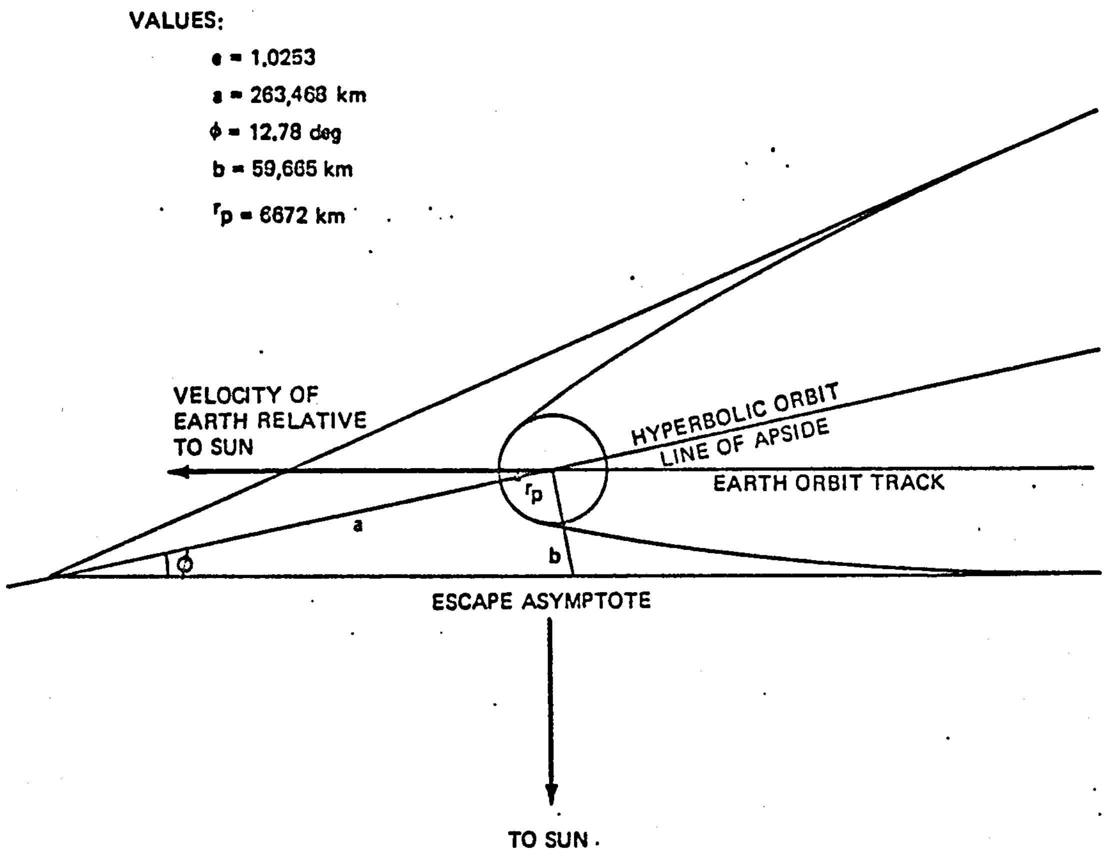


Figure 6.1-4. Reference Mission Departure Hyperbola Geometry

Figure 6.1-5 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 with the plane of the heliocentric transfer orbit is perpendicular to the radius vector from the Sun. Its 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 delta-V penalty for plane change.

OTV Recovery. In the reference mode of operation for the nuclear waste mission, the OTV boosts the SOIS carrying the waste payload into the hyperbolic orbit. The OTV then retrofires into an elliptical orbit returning it to LEO. At LEO a further reduction in velocity circularizes the orbit. The LEO braking maneuver is accomplished by aerobraking. Aerobraking technology was assumed to be already developed in the time frame of interest in the nuclear waste study. Because an aerobraking OTV has much better performance than an all-propulsive OTV, this mode was selected for the reference

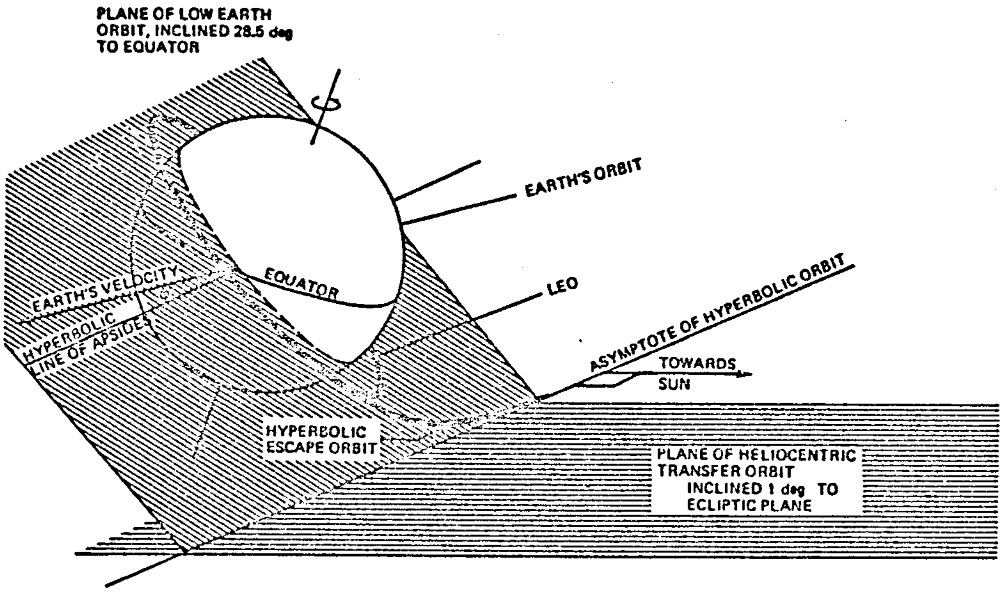


Figure 6.1-5. Orientation of Mission Departure Hyperbola in Relation to Heliocentric Transfer Orbit

mission; in addition, the impact of staging time on total return delta-V is lessened when an aerobraked OTV is used. Aerobraking system weights are essentially independent of the magnitude of velocity reduction in the aerobraking maneuver for the range of velocities of interest here. As a result, the initial retro delta-V essentially becomes the total return delta-V (the delta-V at apogee of the ellipse is about 10 m/sec). The recovery mission profile selected for the aerobraked OTV injection stage limits the initial retro delta-V to 0.4 km/sec and allows a staging time of 410 sec after MECO. The total injection mission timeline and delta-V's are listed in Figure 6.1-6.

6.1.5 Placement Mission Profile

Figure 6.1-7 illustrates the placement mission profile. The injection mission places the SOIS 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 approximate 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. Key parameters of the heliocentric transfer orbit include e, the orbital eccentricity; a and b, the semimajor and semiminor axes, respectively; r_p, the orbit radius at periapsis; and r_a, the orbit radius at apoapsis. Values of these parameters for the reference mission are shown in the figure. Total placement mission timeline and delta-V's are shown in Figure 6.1-8.

	Event	Event Duration (hr)	ET (hr)	Delta-V (m/sec)
INJECT	Start mission		0.000	0
	Ascent & rendez	7.06	7.06	3.04
	LEO ops	11	18.06	0
	Escape inject	0.55	18.61	3391
	Staging	0.17	18.78	3.04
	Retroinject	0.1	18.88	400
	Coast	20.08	38.96	0
	Trans inject	0.01	38.97	3.04
RECOVERY	Coast	20.08	59.05	19.8
	Aeromaneuver	0.08	59.13	0
	Coast	0.75	59.88	0
	Phase inject	0.05	59.93	67.1
	Coast	3	62.93	0
	LEO circ	0.05	62.98	122
	Trim	0.05	63.03	3.04
	Dock	4	67.03	0
	Reserves	0	67.03	76.2

Figure 6.1-6. Injection and OTV Recovery Mission Timeline and Delta-V's

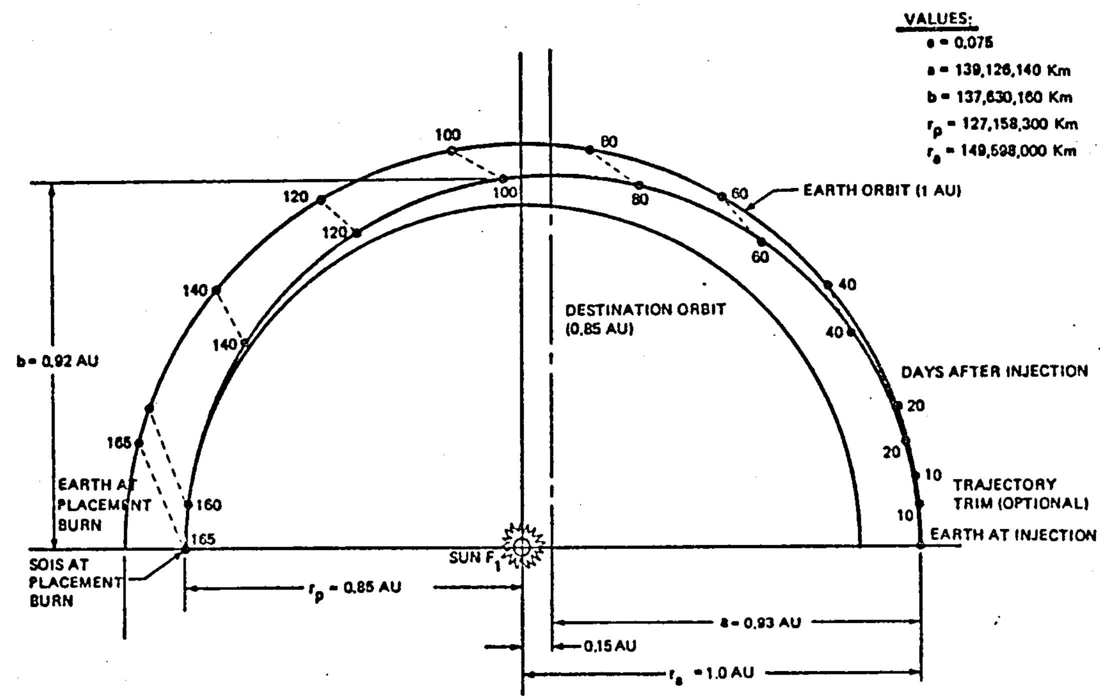


Figure 6.1-7. Placement Mission Profile

Event	Event Duration (hr)	ET (hr)	Delta-V (m/sec)
Staging	0.016	0.016	0.00
Orient	0.016	0.032	1.5
Coast	3899.000	3899.032	3.04
Orient	1.000	3900.032	1.5
Helio circ	0.20	3900.232	1283

Figure 6.1-8. Heliocentric Transfer and Placement Mission Timeline and Delta-V's

6.2 RESCUE MISSION

The study of abort mode options documented in reference 1 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 are needed for both Earth orbit and deep-space rescue locations.

6.2.1 Earth Orbit Rescue Mission Profile

Figure 6.2-1 illustrates the key events for rescue of vehicles stranded in an abort holding orbit. Launch opportunities to heliocentric transfer orbit from the holding orbit

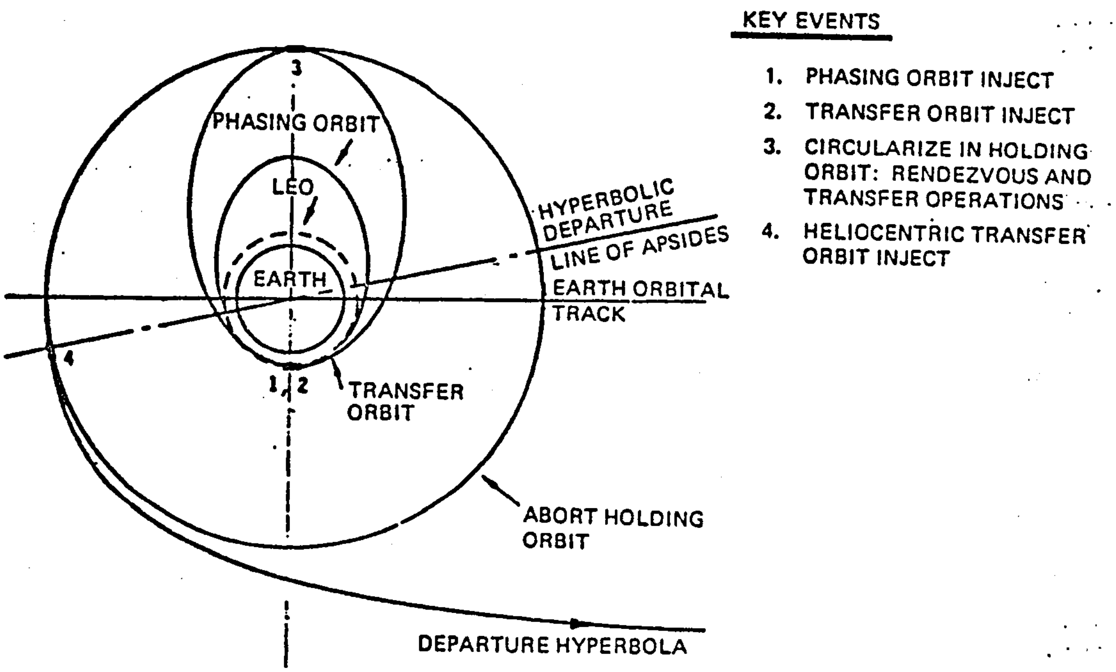


Figure 6.2-1. High Earth Orbit Rescue Trajectory Schematic

occur at 6-month intervals as the plane of the holding orbit becomes perpendicular to the Earth-Sun line. Launch opportunities for transfer of rescue systems to the abort holding orbit occur once per day.

Performance required for rendezvous in Earth orbit was established by an analysis of abort options for failures during the injection burn. Maximum radius for circular storage orbits is about 4×10^4 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.

6.2.2 Deep-Space Rescue Mission Profile

Figure 6.2-2 illustrates the three-burn deep-space mission trajectory. This profile provides for rendezvous with the malfunctioning vehicle at its second perihelion and

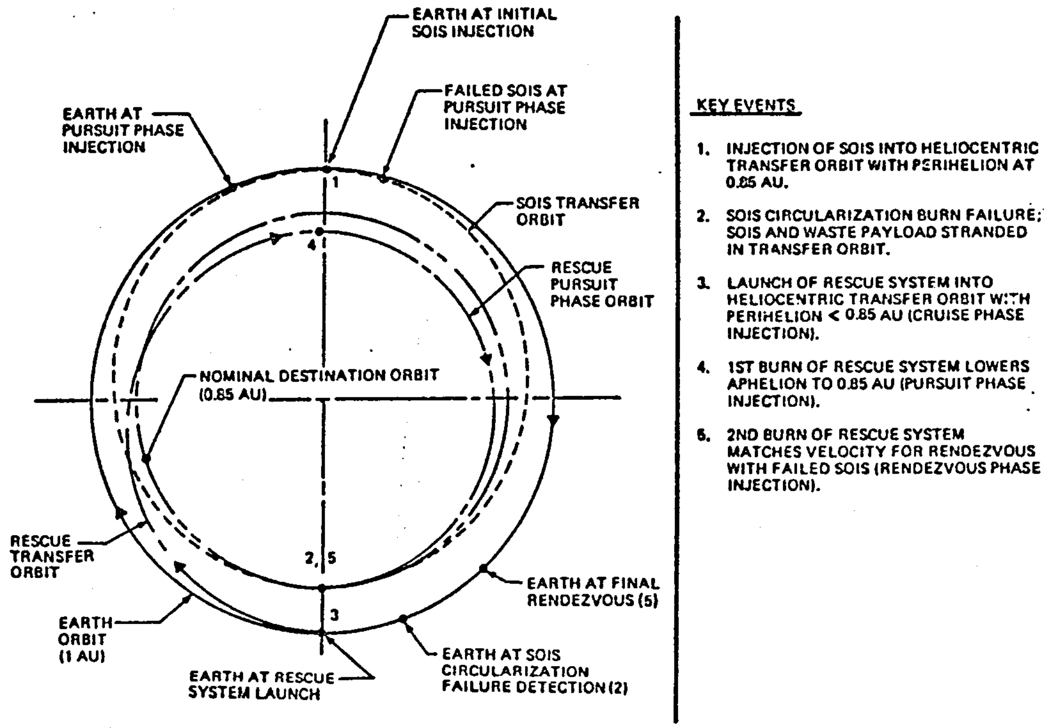


Figure 6.2-2. Deep-Space Rescue Trajectory Schematic

offers reduced delta-V when compared to two-impulse transfers. This trajectory is applicable to a wide range of SOIS failures and provides for maximum mission times of under 2 years.

Performance requirements for deep-space rescue missions are bounded by the case of total SOIS failure. For a typical three-impulse transfer (Fig. 6.2-2), 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.

7.0 DEFINITION OF REFERENCE SPACE SYSTEM ELEMENTS

Primary elements of the reference space system were shown in Figure 5.2-1. They include:

1. The waste payload system (sec. 7.1) which contains and protects the waste payload during all mission phases and which shields personnel and other system elements from the waste payload radiation.
2. Launch site facilities (sec. 7.2) used to assemble the waste payload system and integrate it with the launch vehicle and to service elements of the space transportation system.
3. Launch system (sec. 7.3) used to transport the waste payload and orbit transfer systems from the launch site to a 38-deg inclination, 370-km altitude parking orbit.
4. Orbit transfer system (sec. 7.4) which carries the waste payload from the parking orbit to the destination circular heliocentric orbit at 0.85 AU. A modified version of the orbit transfer system can provide for delivery mission failures by rendezvousing with the failed payload transfer vehicle, taking the waste payload on board, and transferring it to the reference destination.
5. A flight support system (sec. 7.5) which supports the waste payload system in the orbiter cargo bay.

This section is divided into five subsections which describe the characteristics of the primary space system elements in sufficient detail to support estimates of overall performance and risk for the total space disposal concept.

7.1 WASTE PAYLOAD SYSTEM

The waste payload system is the space system element which contains and protects the nuclear waste material as it is transported from the launch site to the final space destination.

The reference waste payload system is illustrated in Figure 7.1-1 which shows the system general arrangement and summary mass properties. The waste payload system consists of two waste payload assemblies weighing 15,332.8 kg each and an interpayload support structure (IPSS) which integrates the two waste payloads into a single structural unit. The interpayload support structure weighs 136.0 kg. The entire waste payload system weighs 30,801.6 kg. Each waste payload contains 3000 kg of cermet waste form, allowing a total of 6000 kg of waste form to be disposed of per mission.

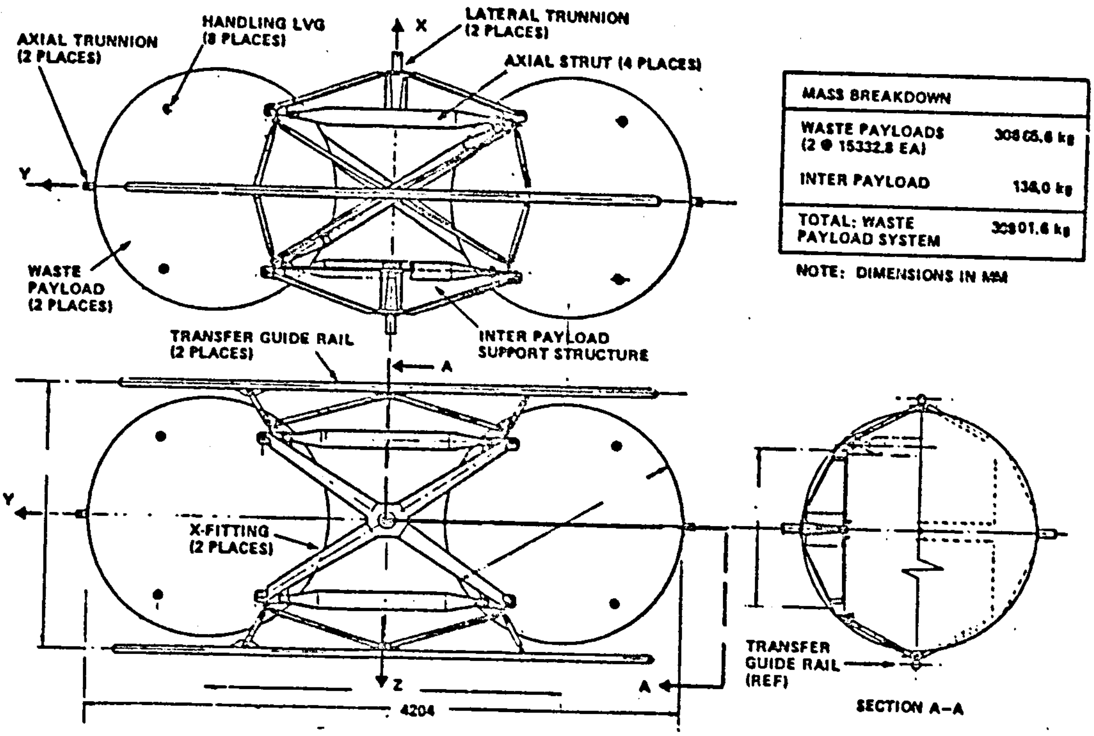


Figure 7.1-1. Waste Payload System General Arrangement

7.1.1 Interpayload Support Structure

The IPSS accomplishes four major functions: waste payload structural support, waste payload transfer, enhancement of waste payload thermal dissipation, and minimization of launch vehicle FSS mass.

Waste Payload Support. The IPSS, in association with the launch vehicle FSS and the orbit transfer system waste payload adapter, supports the waste payloads during all phases of ascent and orbit transfer operations.

Waste Payload Transfer. The IPSS carries guide rails to aid waste payload transfer from the FSS to the orbit transfer system and between a failed solar orbit insertion stage and a rescue vehicle.

Waste Payload Thermal Dissipation Enhancement. By maintaining approximately 1m separation between the waste payloads, the space frame structure provides a free field of view (FOV) for passive waste form thermal dissipation.

Flight Support System Mass Minimization. The same separation minimizes the length of loadpaths in the launch vehicle FSS, minimizing FSS mass.

Figure 7.1-1 shows a three view general arrangement IPSS. Key elements of the interpayload support structure include (1) the two X-fittings, (2) four axial struts, (3) two diagonal struts, (4) two guide rail assemblies, and (5) two lateral trunnions.

The two X-fittings are fabricated from titanium sheet, plate, and bar stock. Each fitting bolts to two of the interface lugs on each waste payload and supports a single lateral trunnion. All struts interface with lugs on the X-fittings.

Four axial and two diagonal struts bolt to lugs on the two X-frames to form a fully triangulated truss connecting the two waste payloads. The struts use conical end fittings machined from titanium bar stock welded to titanium tubing to form an efficient monocoque structure. Integral attachment lugs are machined at the tips of the end fittings.

Two tubular titanium guide rails provide guidance during waste payload transfer operations by rolling on guide rollers mounted on the FSS and waste payload adapter (see secs. 7.4.1 and 7.5). The guide rails are fabricated from titanium tubing with hemispherical ends to aid in transferring between rollers. Lugs on the inboard side of the rails attach to tubular titanium guide rail support struts which interface with the two X-fittings to provide fully triangulated support for the rails.

Two lateral trunnions interface with the FSS and with the waste payload adapter latches to secure the waste payload in position. The lateral trunnions carry only Y-loads in the FSS; X- and Z-loads are carried by the FSS interface trunnions on the outboard ends of the waste payloads.

In the orbit transfer system waste payload adapter, the lateral trunnions carry both Y- and Z-loads. X-loads are reacted through the interface trunnion of the aft waste payload, inducing reaction loads in the Y-direction on the lateral trunnions. The trunnions are machined from steel forgings and held in place by a threaded stud and nut.

7.1.2 Waste Payload Assembly

The waste payload assembly accomplishes four primary functions: containment, radiation shielding, thermal dissipation, and mechanical interfacing with other waste payload system elements.

Containment. The waste payload provides positive containment for the waste form when subjected to the full range of accident and nominal mission conditions specified in the system safety guidelines document (ref. 2). Primary threats to containment which must be withstood include launch pad accidents, terminal velocity impact, heating due to inadvertent reentry, and deep submergence.

Launch pad accidents include blast overpressure and fragment impact due to detonation of the launch vehicle external tank or liquid rocket boosters and exposure to heating from propellant fireballs and ground fires.

Terminal velocity impact is possible as a result of an ascent accident or an inadvertent reentry. Two worst cases exist: impact onto an unyielding surface, which maximizes container mechanical stresses, and impact into soil followed by burial, which imposes maximum thermal stress on the container due to the waste form thermal output. (Dry soil acts as an effective insulator, allowing the thermal output to heat the container with maximum efficiency.)

Reentry due to failures late in the launch vehicle ascent or a decaying orbit impose severe thermal loads on the waste payload container. Vaporization or melting of the container must be withstood to maintain containment.

Deep submergence may follow waste impact on the ocean at any speed. Maximum pressures of up to 1.2×10^7 Pa must be withstood.

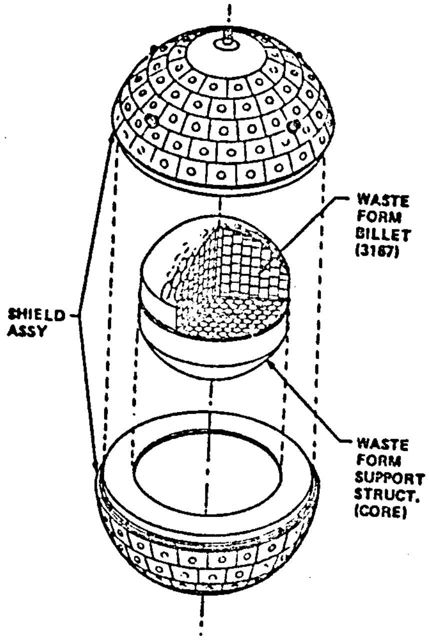
Radiation Shielding. The waste payload provides radiation shielding sufficient to reduce the combined gamma and neutron radiation flux from the reference waste forms carried to a level below 1 rem/hr measured at a distance of 1m from the waste payload surface.

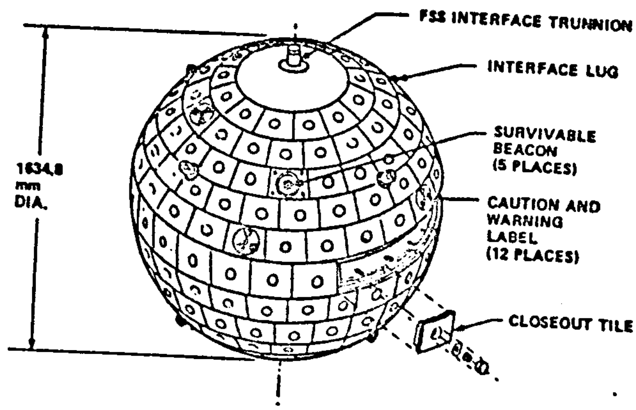
Thermal Dissipation. The waste payload must provide a combination of thermal conductivity from waste form to outer surface, sufficient outer surface area, and outer surface absorbtivity and emissivity values sufficient to allow dissipation of waste form thermal output by passive thermal radiation at the reference 0.85 AU heliocentric orbit destination.

Mechanical Interfaces. The waste payload provides mechanical interfaces for the interpayload support structure, launch vehicle flight support system, and orbit transfer system waste payload adapter to provide mechanical support during ascent and orbit transfer operations.

7.1.2.1 Waste Payload Assembly Description

The general arrangement of the waste payload assembly is shown in Figure 7.1.2-1, along with a summary mass breakdown. The waste payload is spherical and about 1.6m in diameter. It weighs a total of 15,332.8 kg, of which 3000 kg is the reference cermet waste form. Key features include the 228 steel-graphite tiles, which completely cover the outer surface of the waste payload; two cylindrical FSS interface trunnions located 180 deg apart; and eight interface lugs spaced equally on two circles, each located 45 deg from one of the interface trunnions. Additional features include the six survivable beacon





SUMMARY MASS BREAKDOWN	
ELEMENT	MASS, kg
WASTE FORM BILLETS (3167 BILLETS @ 947.289/EA) =	3000.00
CORE (INCLUDES BILLET RETAINERS)	1459.30
SHIELD ASSY INCLUDES PRIMARY CONTAINER AND RADIATION SHIELD, GRAPHITE/STEEL TILES, INTERFACE LVGS & TRUNNIONS, ATTACH HARDWARE	10843.50
SURVIVABLE BEACON SYSTEM (6 @ 5 kg EA)	30.00
TOTAL: WASTE PLD ASSY:	15332.80

Figure 7.1.2-1. Waste Payload Assembly General Arrangement

systems set flush with the tile surface and 12 caution and warning labels which provide proper labeling to identify the waste payload and its contents.

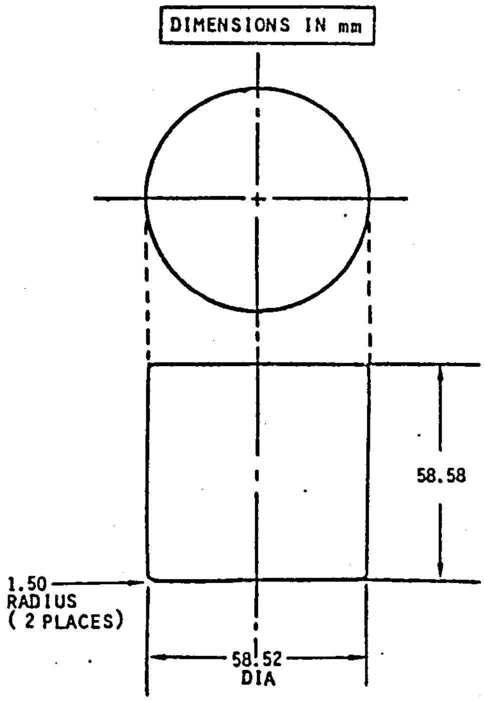
Primary waste payload elements are shown in the exploded view and include the shield assembly, which accomplishes all containment and mechanical interface functions; the waste form support structure (or core), which supports the waste form billets, adds structural integrity to the shield assembly, and aids thermal dissipation; and the cermet waste form billets, which represent the bottom line payload for the entire system. The survivable beacon systems are also primary waste payload elements.

The summary mass breakdown shows the division of mass between the major elements. The shield assembly, at 10,843.5 kg, and the core, at 1459.3 kg, represent 80% of the waste payload mass, illustrating the high priority assigned to providing containment of the waste form. The survivable beacon system, at 30 kg, is almost negligible by comparison. The net delivered waste form mass is 3000 kg. The following sections describe the elements in more detail.

7.1.2.2 Waste Form Billet

The waste form billet is the end product of the high-level waste processing operations and represents the form in which the high-level waste oxides are installed in the waste payload assembly. In each waste payload, 3167 identical billets are carried.

The billet configuration and key characteristics are shown in Figure 7.1.2-2. The billet is a right circular cylinder approximately 60 mm in both diameter and height. The



MATERIAL
- ORNL CERMET
- 67.42% WASTE OXIDE LOADING
- 6.59 g/cm^3

FABRICATION
- POWDER METALLURGY
- UNIAXIAL PRESS & SINTER

HANDLING
- REMOTE ONLY
- NON SMEARING

DIMENSIONS
- NOMINAL DIMENSIONS SHOWN: TOLERANCES TBD
- DIA. SIZED FOR RUNNING FIT IN WASTE FORM SUPPORT STRUCT. BORES

MASS
- 947.3 g

Figure 7.1.2-2. Waste Form Billet Configuration

diameter is sized to provide a running fit in the waste payload core. Corners are radiused to aid fabrication and assembly. The billet is fabricated using a uniaxial press and sintering technique from Oak Ridge National Laboratory (ORNL) cermet material with a 67.42% loading of waste oxides. Overall density is 6.59 g/cm^3, yielding a total billet mass of 947.3g. The resulting material is a hard metal-like substance with considerable strength and resistance to smearing. Handling must be remote only due to the billet's intense gamma ray and neutron emission. Thermal output is approximately 1W per billet.

7.1.2.3 Waste Form Support Structure (Core)

The waste form support structure supports the 3167 individual waste form billets in 241 vertical bores drilled through a solid 316 stainless steel sphere. The core provides a solid metal path for heat conduction from the billets to the shield. By providing a solid, incompressible core, the waste form support structure also augments the structural integrity of the outer shield. Figure 7.1.2-3 shows the core configuration with key dimensions and characteristics noted. Primary elements include the support structure and the billet retainers.

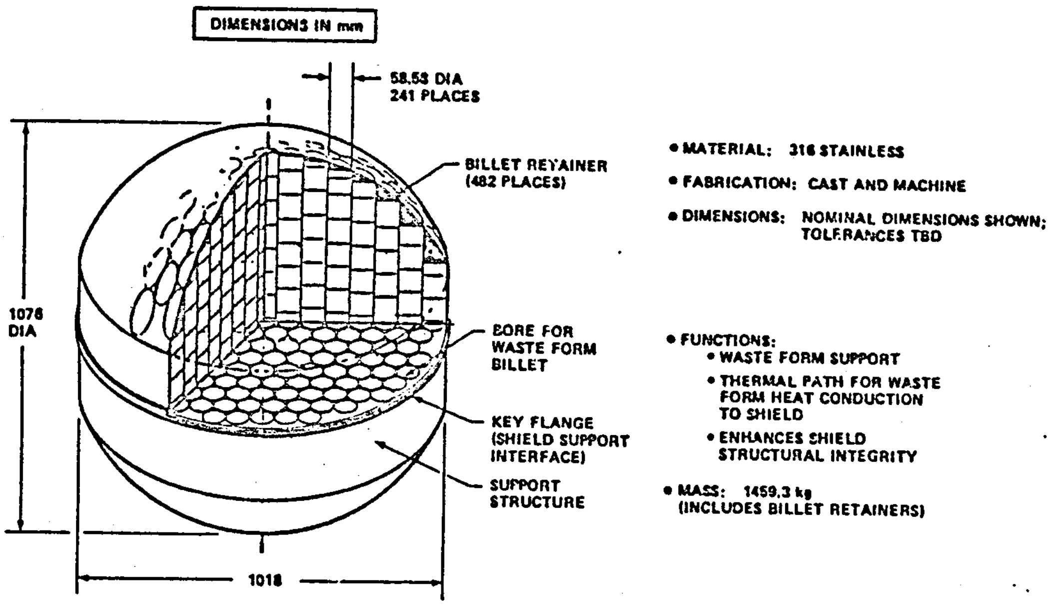


Figure 7.1.2-3. Waste Form Support Structure (Core) Configuration

The support structure is basically a sphere, 1076 mm in diameter, with 241 parallel 58.58-mm-diameter holes bored through in a hexagonal, close-packed arrangement. A cylindrical section is machined into the sphere outer contour, parallel to the bores; a projecting flange, or key circling the cylindrical section, interfaces with a machined groove in the lower shield half to support the core in the shield. The support stucture is fabricated by machining from a 316 stainless steel casting.

The billet retainers are installed in the ends of the boreholes to retain the billets in position after billet loading. The retainers are also fabricated from 316 stainless and are fastened in place with reversible spring-loaded retaining clips to aid in the automated billet loading operation.

Combined mass of the core and retainers is estimated at 1459.3 kg.

7.1.2.4 Shield Assembly

The shield assembly is the primary barrier against release of the waste form. It encases the core and waste form billets inside a seamless shell of Inconel 625 superalloy, 224 mm (8.8 in.) thick. This shell is further protected by a layer of graphite in the form of 228 interlocking tiles, 50 mm (1.97 in.) thick, and a final outer steel sheath, 4.3 mm (0.19 in.) thick. Overall shield assembly mass for the reference waste payload system is 10,843.5 kg. Figure 7.1.2-4 shows the assembled waste payload with key features and dimensions of the shield assembly and core illustrated.

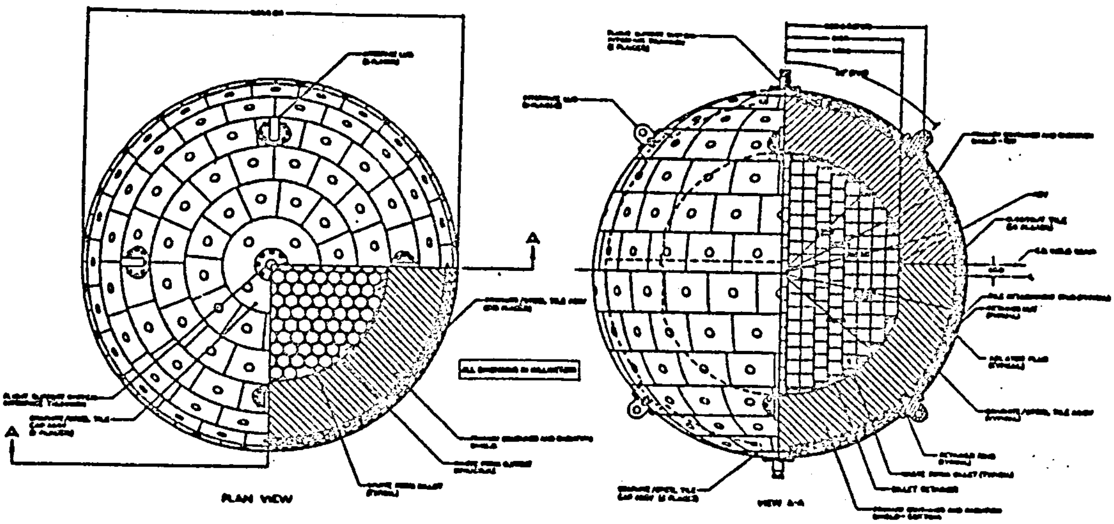


Figure 7.1.2-4. Reference Waste Payload Assembly Features and Dimension

Primary shield elements include the primary container and gamma radiation shield, the composite graphite-steel tile assemblies, and the interface fittings.

Primary Container and Gamma Radiation Shield. The primary container provides protection from blast overpressure, fragment impact, deep immersion, and terminal velocity impact. The structural integrity of the shield and its thickness provide sufficient strength to withstand fragment impact. Its thickness and the essentially incompressible core provide stability against buckling due to external pressure imposed by blast or deep submergence. The shield absorbs the kinetic energy of terminal velocity impact by plastic deformation of shield material and the core. Elongation is limited to values that preclude fracture or cracking which could violate containment.

Additional functions of the primary container include radiation shielding, thermal conduction and capacity, and interface mechanical support.

The 224-mm thickness of the shield attenuates the intense gamma flux from the cermet waste form billets to less than 0.5 rem/hr at 1m. Some neutron flux attenuation is also provided.

The solid container provides good thermal conduction for heat flux from the waste billets. The temperature drop across the shield is limited to 77°C. The substantial thermal capacity of the primary container minimizes fireball radiation thermal effects on the waste payload and the high melting point of the Inconel 625 container material helps to withstand the ground fire environment. Mechanical support for the interface fittings is provided by 10 bosses machined into the outer surface of the container.

The container is fabricated in two halves by machining from Inconel 625 hemispherical forgings. Bosses for attachment or for surface fittings are integrally machined. Tapped holes are provided for the interface fitting bolt circles and the studs used for attachment of the 228 composite tiles. Grooves machined at the inner edge of the joining surface accommodate the core interface flange.

The primary container is assembled into a seamless shell after core installation by electron beam (EB) welding using an automated EB welder (see sec. 8.1, Waste Payload Fabrication and Assembly Operations).

<u>Composite Graphite-Steel Tiles</u>. 288 graphite-steel tile assemblies and two tile cap assemblies completely enclose the primary container. The tiles are the primary method of damping the neutron flux from the waste form billets and act as an ablator during reentry to protect the primary container and core from reentry heating.

The tiles vary in size and shape to accommodate their location at various places on the waste payload. Each tile consists of a 50-mm-thick graphite layer, with a 4.8-mm steel plate bonded to the outside of it. The graphite tiles are stepped so that adjacent tiles overlap, preventing direct paths for neutron escape. Each tile has a central counterbored hole which allows it to be fastened to one of the studs installed in the primary container. After the tile has been placed over the stud, a washer and nut are installed to retain it. A plug of ablator with a steel outer layer is bonded in place over the nut to provide additional neutron attenuation and entry protection. An inward, cup-like extension of the tile outer steel plate is captured under the tile retention nut to provide positive mechanical retention of the steel outer sheath.

All of the tiles except for a single row around the "equator" of the waste payload are preinstalled to minimize operations necessary after the radioactive waste form billets are installed. The clear area remaining allows access for the electron beam used to weld the shield halves together. Following welding, the single row of closeout tiles is installed by an automated tile installation machine.

Interface Fittings. Two types of interface fittings bolted to the bosses on the primary container allow attachment of the IPSS. Two FSS interface trunnions interface with the launch vehicle FSS and the orbit transfer system waste payload adapter. Eight interface lugs provide for ground handling and attachment of the IPSS.

Both types of interface fittings are machined from high-strength steel forgings and attach to the primary container using an integrated eight-bolt mounting flange and a 50-mm-diameter shear pin. The cylindrical trunnions incorporate a groove used as a grapple interface during rescue. The lugs used for ground handling and IPSS attachment have a 25.00-mm borehole to accommodate attach hardware. An edge distance (e/d) of 2.0 is used in the lug design. Like the tiles, the interface fittings are preinstalled on the individual shield halves to provide handling aids and to minimize operations on the waste payload after the radioactive waste form billets are installed.

7.1.2.5 Survivable Beacon System

Six survivable beacon systems are bolted into place, 90 deg apart, on the surface of the waste payload to aid post-accident location. The beacons mount flush to the surface of the adjacent tiles; each beacon replaces one tile. They have an outer layer of ablative material to protect them from entry heating. Location of the six beacons precludes destruction of all six by ground impact or impact of fragments. The beacon is hardened to 50,000g impact loading and against peak overpressures.

The beacons are designed to operate as both radio and sonar beacons. Radio beacon signals are emitted at the standard emergency locator beacon system frequencies of 121.5 and 243.0 MHz using a flush loop-type antenna. A nominal 300-MW-per-channel output is provided at a 100% duty cycle, declining to 100 MW per channel after 48 hr at -30°C. Both radiofrequency (RF) carriers are modulated at audio frequencies using a 300 to 1600 Hz continuous sweep.

Power for both RF and sonar signals is provided by a self-contained lithium battery pack. Beacons are turned on and armed prior to shuttle liftoff. They remain on standby for the shelf life of the battery pack (at least 3 years).

Activation of each individual beacon is accomplished using built-in sensors including: (1) a hardwired command, linked through the SOIS and OTV telemetry links, (2) increasing barometric pressure (decreasing altitude), (3) a thermal switch which responds to reentry heating or ground fires, and (4) an inertia switch responsive to impact or entry g-loading. All of these stimuli will switch the beacon on in the RF mode. Built-in immersion and a backup pressure sensor turn off the RF output and divert power to the sonar transducer and its power amplifiers.

Total mass allocation used for each individual beacon system is 5 kg. For comparison, a NARCO ELT-10 emergency locator beacon (designed for light aircraft with similar RF characteristics and life but without the hardening, entry protection, and sonar beacon) weighs 1.6 kg.

7.2 LAUNCH SITE FACILITIES

Facilities at the launch site are used to build up the waste payload system, integrate it with the uprated STS launch vehicle, and provide support for both uprated shuttle and SDCLV. The reference waste payload quantity requires 30 flights of each vehicle per year.

Specific facility areas include (1) the nuclear payload processing facility NPPF, used to assemble the waste payload system and integrate it with the flight support system; (2) space system support facilities, used to process the launch vehicles; and (3) specialized nuclear waste payload cargo integration facilities, used to provide radiation safety for the launch crew during waste payload/orbiter integration.

7.2.1 Nuclear Payload Processing Facility

The NPPF is illustrated schematically in Figure 7.2.1-1 which shows a cutaway of the approximately 900-m^2 NPPF building. Primary features include the shielded loading

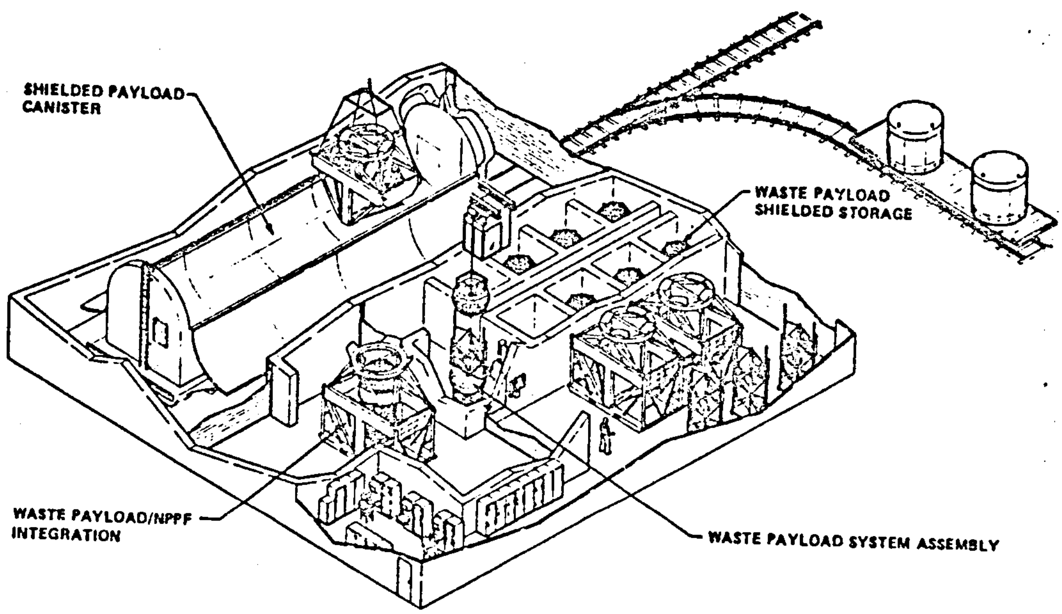


Figure 7.2.1-1. Nuclear Payload Processing Facility (NPPF)

dock, a shielded storage area for up to eight waste payload assemblies, a shielded waste payload system assembly station and FSS integration station, an unshielded space for storage of flight support systems and interpayload support structures, and an office and control area. All areas except for office and control are accessible to a shielded overhead crane.

Areas accessible to the waste payload are shielded by concrete walls and can be closed off when a waste payload is present. With all waste payloads in storage cells, free access to the loading dock, assembly, and integration stations is available for setup. Assembly of waste payload systems and FSS integration is accomplished remotely using the overhead crane, a turntable at the assembly station, and remotely operated wrenches. After the assembled waste payload system is lowered into the FSS, it is rotated into launch position and secured by the FSS rotation and latching system (see sec. 7.5).

In operation, the waste payload assemblies are unloaded in the shielded loading dock after rail transport from the fabrication facilities using the shielded overhead crane. The waste payloads are placed in storage cells prior to assembly and integration. Following integration with the FSS, the resulting waste payload cargo is installed in the shielded payload canister (sec. 7.2.3) for transportation to the launch pad. The canister loading is accomplished in the shielded loading dock area by the shielded overhead crane.

7.2.2 Space Transportation System Support Facilities

Operation of the reference launch and orbit transfer systems at a flight rate of 35 missions per year requires additions to the existing STS support facilities at KSC. Primary additional facilities required are illustrated in Figure 7.2.2-1, along with their key characteristics and the quantities of each which would be required to support 30 launches per year.

Launch Complex. The launch complex consists of the launch pad and associated propellant storage and handling facilities. Primary launch pad elements include fixed and rotating service structures (RSS), flame trenches and flame deflectors, and a water deluge system. The rotating service structure is used to integrate the waste payload cargo element with the uprated space shuttle system. If launch pads are dedicated to specific launch vehicles, only the pad used by the uprated shuttle needs an RSS. Crew access to the uprated shuttle and umbilicals providing hydrogen vent functions for both uprated STS and SDCLV are provided by the fixed service structure. The fixed service structure also mounts a 25-ton general-purpose crane. The main flame trench which runs beneath the mobile launch platform and flame deflectors safely dissipate the plumes from the liquid rocket boosters and space shuttle main engines during launch vehicle liftoff. The water

ORIGINAL PAGE IS
OF POOR QUALITY

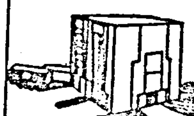
DESIGNATION		PRIMARY ELEMENTS	PRIMARY FUNCTIONS	QUANTITY REQUIRED	COMMENTS
	LAUNCH COMPLEX	• FLAME TRENCH AND DEFLECTORS • RSS • FSS • PROPELLANT STORAGE TANKS, LINE & PUMPS • WATER DELUGE SYSTEM	• CREW ACCESS } UPRATED STS • INTEGRATE WASTE PAYLOAD WITH UPRATED STS ORBITER } UPRATED STS • LOAD PROPELLANT } UPRATED STS & SDCLV • SUPPORT LAUNCH } UPRATED STS & SDCLV	2	IF LAUNCH PADS ARE DEDICATED TO SPECIFIC LAUNCH VEHICLES, ONLY THE PAD USED BY THE UPRATED STS NEEDS A RSS.
	MOBILE LAUNCH PLATFORM	• PLATFORM • TSM (2) • FLAME HOLES	• VEHICLE SUPPORT DURING INTEGRATION, TRANSFER, LAUNCH • TSM'S ARE PRIMARY FLUID UMBILICAL INTERFACE	4	COMMON TO BOTH TYPES OF LAUNCH VEHICLE
	VERTICAL ASSEMBLY BUILDING (VAB)	• CHECKOUT CELLS (4) • TRANSFER AISLE • LAUNCH CONTROL CENTER • WORKSTANDS	• BUILDUP AND CHECKOUT OF • UPRATED STS • SDCLV • PAYLOAD INTEGRATION • SDCLV	1 (1 SPARE CELL)	ALL OF THESE FACILITIES MIGHT BE ORGANIZED AS A SINGLE COMPLEX OR BUILDING
	ORBITER PROCESSING FACILITY	• CHECKOUT BAYS (3) • WORKSTANDS (3)	• REFURBISH ORBITER BETWEEN FLIGHTS • REMOVE FSS/RETURNED PAYLOAD • REFURBISH SDCLV PROPULSION AND AVIONICS MODULE BETWEEN FLIGHTS	2 (1 SPARE CHECKOUT BAY)	
	LRB PROCESSING FACILITY	• HIGH BAY • SUPPORT SHOPS • WORKSTANDS	• REFURBISH LRB'S BETWEEN FLIGHTS • CHECKOUT LRB'S PRIOR TO INTEGRATION IN VAB	1 (NEW DESIGN)	
	E.T. PROCESSING FACILITY	• HIGH BAY • STORAGE • WORKSTANDS	• BUILDUP AND CHECKOUT E.T.'S PRIOR TO INTEGRATION IN VAB	5	
	OTV PROCESSING FACILITY	• HIGH BAY • SUPPORT SHOPS • STORAGE • WORKSTANDS	• REFURBISH INJECTION STAGES BETWEEN FLIGHTS • CHECKOUT INJECTION STAGES AND SDIE PRIOR TO INTEGRATION • INTEGRATE AND CHECKOUT O.T.S. PRIOR TO INTEGRATION WITH SDCLV IN VAB	1 (NEW DESIGN)	

Figure 7.2.2-1. Additional Dedicated STS Support Facilities Required at KSC

deluge system is used to reduce the acoustic pressure levels to which the launch vehicles are subjected during liftoff. Two launch complexes are required to support the reference concept for space disposal.

<u>Mobile Launch Platform</u>. The mobile launch platform (MLP) supports all elements of the launch vehicle during buildup and checkout in the vertical assembly building (VAB) through transfer to the launch pad up until the actual liftoff. The MLP is a box section structure fabricated from steel plate. It contains flame holes to pass the plumes from the liquid rocket boosters and the space shuttle main engines and two 20-ft-tall tail service masts (TSM) which carry umbilical plates that supply propellant and other fluid services to the uprated space shuttle orbiter and the propulsion and avionics pod of the shuttle-derived cargo launch vehicle. Four MLP's are required to support the reference concept flight rate. A single platform configuration would be common to both types of launch vehicles used.

<u>Vertical Assembly Building</u>. The VAB is a 160m-tall structure that is 218m long and 158m wide. It contains four checkout cells used for buildup and checkout of both uprated shuttle and shuttle-derived cargo launch vehicles. The four checkout cells are grouped

around a transfer aisle which provides for transfer of components during the launch vehicle buildup and checkout process. Multilevel workstands at each checkout cell provide extensive access to the launch vehicle during the buildup and checkout process. Four 139m-high doors provide for launch vehicle exit after the buildup and checkout process is completed. Immediately adjacent and connected to the VAB is the launch control center which contains two launch processing systems used to control the checkout and launch of the space disposal launch systems. The VAB is also used for payload integration of the shuttle-derived cargo launch vehicle which is accomplished in the VAB rather than at the pad. A single VAB is adequate to support the 35-mission-per-year flight rate of the reference space concept. One of the four checkout cells would be available at this flight rate to provide for later increases in the number of flights per year.

Orbiter Processing Facility (OPF). The OPF is used for refurbishment of orbiters between flights of the uprated space shuttle system and for refurbishment of the propulsion and avionics modules used for the shuttle-derived cargo launch vehicle. The OPF is also used for removal of the flight support system from the uprated shuttle following each flight. Two orbiter processing facilities, each containing two checkout bays, are adequate to support the reference flight rate and will leave one spare checkout bay to accommodate potential flight rate increases.

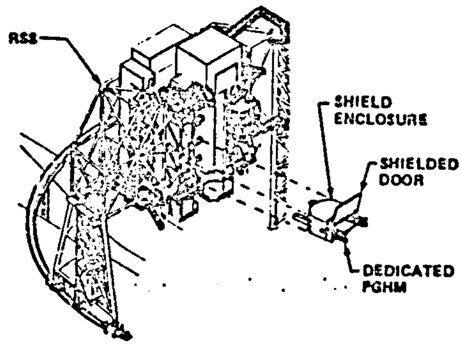
LRB Processing Facility. The LRB processing facility consists of a high bay, support shops, and workstands to refurbish the liquid rocket boosters between flights and to check out the liquid rocket boosters prior to transfer to the VAB for integration with the rest of the launch system. LRB's are checked out in a horizontal position in the high bay. The LRB processing facility would be a new design and it is assumed that one LRB processing facility would be adequate to support the reference concept flight rate.

7.2.3 Specialized Nuclear Waste Payload Cargo Integration Facilities

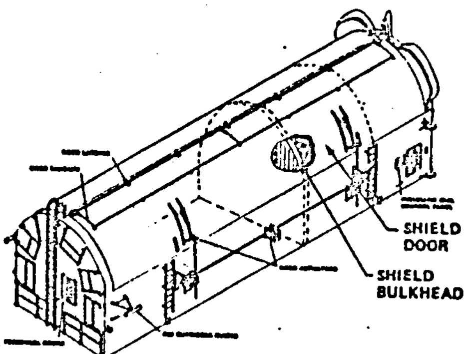
Modifications to the existing designs for the rotating service structure and multi-mission support equipment (MMSE) payload canister allow integration of the waste payload cargo element in the uprated STS cargo bay at the launch pad, while minimizing restrictions on STS launch preparation activities due to waste payload radiation.

Shielded Payload Canister. The shielded payload canister (shown in Fig. 7.2.3-1) is used to transport the waste payload cargo element from the NPPF to the RSS. A standard MMSE payload canister is modified by the addition of radiation shielding material to limit external radiation to the limits specified in the system safety guidelines document for the shipping cask (par. 2.2.3.5). Additional modifications allow the waste payload cargo

ORIGINAL PAGE IS
OF POOR QUALITY



- SHIELD
  - CONTAINS WASTE PAYLOAD CARGO ELEMENT AFTER REMOVAL FROM SHIELDED PAYLOAD CANISTER
  - ALLOWS NORMAL SHUTTLE PREPARATION ACTIVITIES BY PROVIDING SAFE DOSE RATE IN RSS
- DEDICATED PGHM
  - DEDICATED TO SHIELDING AND INTEGRATION OF WASTE PAYLOAD CARGO ELEMENT
  - PROVIDES FOR REMOTE TRANSFER OF WASTE PAYLOAD CARGO ELEMENT FROM PAYLOAD CONTAINER TO SHIELD
  - PROVIDES FOR REMOTE INTEGRATION OF WASTE PAYLOAD CARGO ELEMENT TO ORBITER
- RSS
  - USE AS IS



- PAYLOAD CANISTER
  - ADD SHIELDING TO EXISTING CONFIGURATION TO CUT EXTERNAL DOSE RATE TO < 10 MREM HR WITH WASTE PAYLOAD CARGO ELEMENT INSTALLED
  - HANDLE AS NORMAL PAYLOAD CANISTER
  - RETAIN EXTERNAL CONFIGURATION TO PROVIDE COMMONALITY WITH RSS, ORBITER INTERFACES
  - ACCOMMODATE REMOTE INSTALLATION AND REMOVAL OF WASTE PAYLOAD CARGO ELEMENT

Figure 7.2.3-1. Specialized Waste Payload Cargo Integration Facilities From Modification of Existing STS Support Equipment

element to be installed in the canister and remotely latched into position. The added shielding allows the loaded payload canister to be handled using the same facilities as used for nonradioactive payloads. An unmodified MMSE canister tranporter is used to carry the canister from NPPF to RSS.

RSS. The RSS is modified by replacing the general-purpose payload ground-handling mechanism (PGHM), used for integration of the wide variety of cargos carried by the STS, with a dedicated PGHM optimized for integration of the nuclear waste cargo element (Fig. 7.2.3-1).

The dedicated PGHM is permanently installed at the correct station for installation of the waste cargo. It is capable of single-axis translation only. The payload attach points on the PGHM engage the payload STS interface trunnions and are designed for remote engagement and disengagement.

The dedicated PGHM is mounted inside a shielded container with a door. After the payload canister is positioned in the RSS airlock, the RSS interior is cleared of personnel. The canister doors are opened, and the dedicated PGHM is extended and remotely latched to the waste payload cargo element trunnions. The latches securing the waste payload to the canister longerons are then remotely released, and the PGHM and waste cargo are retracted into the shielded container. The container door is then closed.

Personnel can reoccupy the RSS at this time and proceed with normal STS launch preparation activities. Personnel are cleared once again at about T-20 hours to allow waste payload cargo element installation in the orbiter cargo bay in a reversal of the operations that removed it from the canister. Securing of the cargo in the bay and mating of interfaces is accomplished remotely. Payload bay door closure is accomplished remotely at T-10 hours. Shadow shields are positioned around the closed payload bay doors to allow personnel to reoccupy the RSS for further launch preparation activities. Following retraction of the RSS at T-2 hours, distance alone keeps the radiation exposure below allowable limits.

Contingency Access. Use of the specialized waste payload cargo integration facilities described allows nominal integration operations to be carried out without exposure of personnel to any radiation from the waste payload. The relatively low level of radiation from the waste payloads (less than 1 rem/hr at 1m from the waste payload surface) would allow access to the area around the payload to deal with contingencies without exceeding the normal operations exposure limits for individuals in controlled areas. An individual could work on the FSS with payload installed for up to 3 hours without exceeding the 3-rem dose limit for a calendar quarter. This factor provides a powerful means of coping with "glitches" or minor problems encountered during waste payload cargo integration.

7.3 LAUNCH SYSTEM

The launch system transports all elements of the space disposal system from the launch site to low Earth orbit. The reference launch system uses two launch vehicles which offer an attractive combination of low risk and low cost. The waste payload is launched on an uprated version of the space shuttle which uses liquid rocket boosters. This vehicle minimizes risk by offering a number of intact recovery modes in the event of a launch abort. The orbit transfer system is launched using a shuttle-derived cargo launch vehicle, which offers almost twice the payload of the uprated STS at slightly lower cost but which has no intact abort capability. Combining these vehicles in a dual launch mission scenario allows taking advantage of the low risk of the winged orbiter while using the more cost-effective SDV to boost the heavier but less critical orbit transfer system.

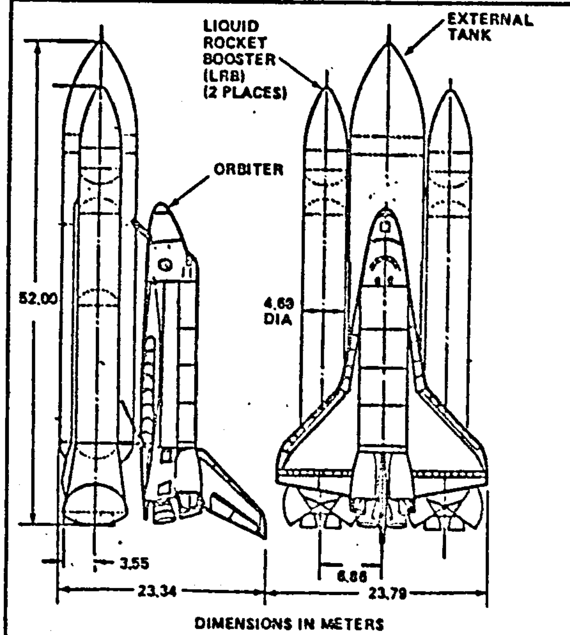
7.3.1 Waste Payload Launch Vehicle

The uprated STS is used to transport the waste payload from the KSC launch site to a circular low Earth orbit at an altitude of 370 km and an inclination of 38 deg.

7.3.1.1 General Arrangement and Key Elements

The uprated STS general arrangement and key dimensions are shown with key vehicle characteristics in Figure 7.3.1-1. Two liquid rocket boosters flank the external tank, which contains propellant for the three orbiter uprated SSME's. The orbiter is attached by struts to the upper side of the ET. Key elements include the ET, LRB's, and orbiter.

GENERAL ARRANGEMENT



KEY CHARACTERISTICS	
APPLICATION	WASTE PAYLOAD TRANSPORT TO LEO
VEHICLE CHARACTERISTICS *INCLUDES E.T., ORBITER, PAYLOAD	GLOW = 1.857×10^6 kg BLOW = 0.980×10^6 kg W_{p1} = 0.841×10^6 kg ULOW = 0.877×10^6 kg W_{p2} = 0.705×10^6 kg MASS @ MECO* = 0.171×10^6 kg
PAYLOAD TO 370 km ORBIT @ 38°	47,000 kg
CARGO ACCOMMODATION	CARGO BAY IS 4.57m DIA x 18.3m LONG. INTERFACES PER JSC 07700 VOL XIV
ABORT MODES	• RETURN TO LAUNCH SITE (RTLS) • ABORT ONCE AROUND (AOA) • ABORT TO ORBIT (ATO) • ABORT TO WATER LANDING
MODIFICATIONS	• FULL GROSS WEIGHT LANDING
SAFETY PROVISIONS	• FLOTATION SYSTEM

Figure 7.3.1-1. Uprated STS General Arrangement

External Tank. The external tank is identical to the lightweight version developed for the existing STS. It contains the liquid hydrogen fuel and liquid oxygen oxidizer and supplies them under pressure to the three main engines in the orbiter during liftoff and ascent. When the main engines are shut down, the external tank is jettisoned, enters the Earth's atmosphere, breaks up, and impacts in a remote ocean area. It is not recovered.

Major components of the ET are the monocoque LOX tank, the semimonocoque LH_2 tank, and the unpressurized intertank which contains most of the ET electronics. All components are fabricated from aluminum alloy. The ET is 97m long, 8.38m in diameter, and weighs approximately 33,503 kg when empty.

The entire external tank is covered with a 1.27-cm cork-epoxy composition sprayed or premolded to withstand localized high heating during boost. It is then covered with a 2.54- to 5-cm spray-on foam insulation. The LH_2 tank insulation also precludes liquid air formation on the external surface.

The external tank is attached to the orbiter at one point forward attachment and two points aft. In the aft attachment area, there are also umbilicals which carry fluids, gases, electrical signals, and electrical power between the tank and the orbiter. Electrical signals and controls between the orbiter and the two solid rocket boosters also are routed through those umbilicals.

Liquid Rocket Boosters. The two liquid rocket boosters are self-contained systems containing propellants and figures. In operation they boost the ET for the first 2 min of flight. At the end of the boost period, the two LRB's are separated. After separation, RCS thrusters reorient the booster to an aft end forward attitude and maintain that attitude through atmospheric reentry. Clamshell doors close over the engines and seal off the booster engine compartment, providing thermal protection for reentry and water protection on landing. Parachutes are deployed to place the stage in a horizontal attitude and to decelerate the booster to about a 25 m/sec velocity. Solid rocket motors provide final deceleration to a soft landing on the water. After landing, the LRB's are recovered by ships for refurbishment and reuse.

Major LRB components include the basic structure and the main propulsion and recovery systems.

The basic structure consists of three propellant tanks, an intertank structure, and an engine compartment. RP-1 is contained in the nose tank with LH_2 and LO_2 in an integral tank in the aft body, separated by a common bulkhead. Propellant lines from the RP-1 and LH_2 tanks are carried to the engine compartment in a tunnel along the outside of the body. The engine compartment section is made of integrally stiffened skin supported by frames and the engine thrust structure. The conical shape provides the required clearance for the engines and the necessary base diameter for the hinged clamshell doors. During ascent, the doors are positioned at either side of the engine compartment. After booster separation, the doors are pivoted to the closed position and sealed.

The main propulsion system consists of four uprated space shuttle booster engines which burn LOX and RP-1 and are cooled by LH_2. The engines have a sea level thrust of 2.39×10^6N each and operate at a specific impulse of 331 sec (sea level). Vacuum thrust and specific impulse are 2.61×10^6N and 361 sec, respectively.

The recovery system includes separation rockets, drogue and main parachutes, terminal deceleration rockets, and airbags to cushion impact. Radio and optical beacons act as location aids to help recovery.

Overall length of the LRB is 46.94m and tank diameter is 4.69m. The engine compartment maximum diameter at the flare is 7.1m. Dry mass of the LRB is 66,971 kg.

Orbiter. The orbiter is the crew- and payload-carrying unit of the shuttle system. It is 37m long, has a wingspan of 24m, and weighs approximately 75,000 kg without fuel. The orbiter is fully reusable. It carries cargo in a payload bay 18.3m long and 4.6m in diameter. The orbiter's three main liquid rocket engines each have a thrust of 2.1×10^6N.

Orbiter modifications required for the space disposal mission are discussed in later sections. Complete details on the orbiter's large number of systems are available in a variety of NASA publications.

7.3.1.2 Launch Vehicle Characteristics

Basic characteristics of the launch vehicle are also shown in Figure 7.3.1-1. Gross liftoff weight (GLOW) is 1.857×10^6 kg. This is composed of a booster liftoff weight of 0.98×10^6 kg (for two liquid rocket boosters) and an upper stage (in this vehicle, the orbiter and ET) liftoff weight (ULOW) of 0.377×10^6 kg. The booster propellant load of LOX, RP-1, and LH_2 (W_p1) is 0.841×10^6 kg. LOX and LH_2 in the ET (W_p2) equal 0.705×10^6 kg. Vehicle mass at MECO, when all propellants have been expended, is 0.171×10^6 kg.

7.3.1.3 Performance

The uprated shuttle delivers 47,000 kg of payload to the reference 370-km altitude orbit at 38-deg inclination.

7.3.1.4 Cargo Accommodation

The orbiter cargo bay is 18.3m long and 4.1m in diameter. Mechanical interfaces for the uprated shuttle are assumed to be identical to those of the existing orbiter specified in the space shuttle system payload accommodations (JSC 07700 vol. XIV). The flight support system for the waste payload described in section 7.5 has been designed to conform to these standards.

7.3.1.5 Abort Modes

The uprated shuttle has three intact abort alternatives, depending on when abort becomes necessary. These are to return to launch site (RTLS), abort once around (AOA), and to abort to orbit (ATO).

In addition, an abort to water landing is possible for many failure modes. While not an intact abort, a water landing followed by actuation of the orbiter flotation system (OFS) described in section 7.3.1.7 would pose a relatively low risk to the waste payload. Most mission contingencies would result in one of the three intact abort modes illustrated in Figure 7.3.1-2. The following paragraphs briefly describe the abort modes illustrated.

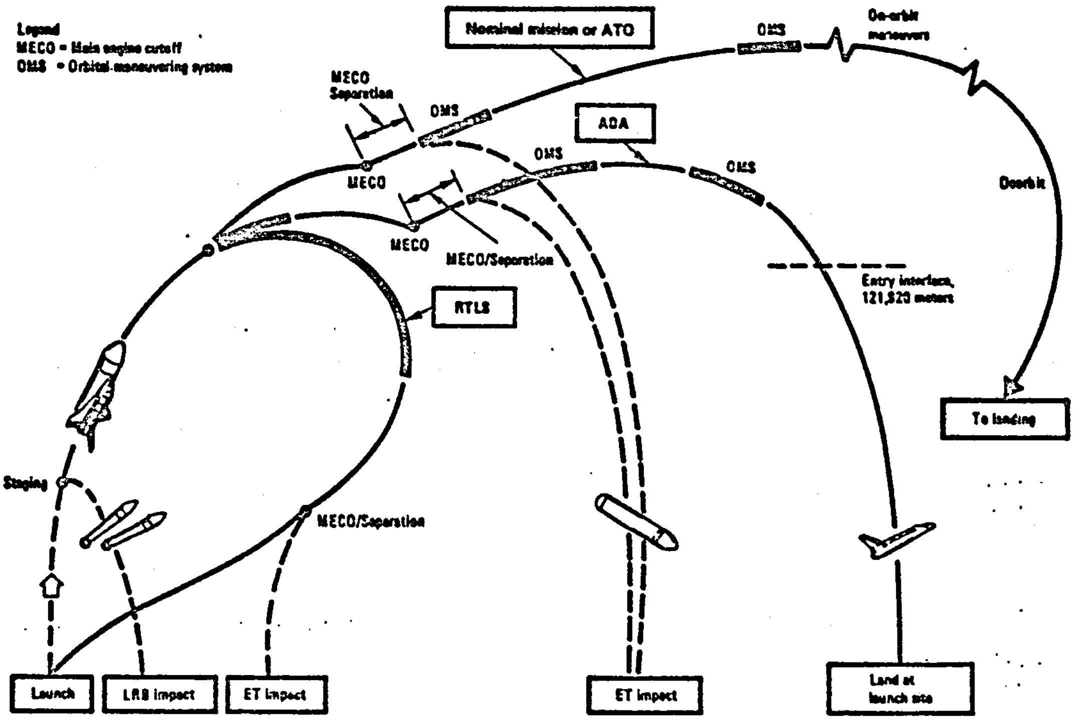


Figure 7.3.1-2. Uprated STS Intact Abort Options

<u>Return to Launch Site</u>. This mode will be used in the event of a main engine failure between liftoff and the point at which the next abort mode (AOA) is available.

The space shuttle (orbiter and external tank) continues to thrust downrange, with the two remaining main engines, the two OMS, and the four aft +X RCS thrusters firing, until the remaining propellant for the main engines equals the amount required to reverse the direction of flight.

A pitch-around (plus pitch) maneuver is then performed at approximately 5 deg/sec, which places the orbiter and external tank in a heads-up attitude, pointing back toward the launch site. Main engine cutoff is commanded when altitude, attitude, flightpath angle, heading, weight, and velocity/range conditions combine for acceptable orbiter-external tank separation (tank impact no closer than 24 n. mi. from the U.S. coast) and orbiter glides to the launch site runway.

Abort Once Around. This mode will be used from approximately 2 min after normal booster separation to the point at which the abort-to-orbit mode becomes available. Again, this abort would occur in the event of a main engine failure.

The space shuttle vehicle continues to thrust with the remaining main engines and the OMS and aft RCS +X thrusters. The OMS and RCS thrusting periods terminate when the amount of propellant remaining in these two systems will support two OMS thrusting periods after MECO.

Main engine cutoff is followed by jettisoning of the external tank. The OMS thrusters are fired after jettisoning the external tank to obtain an apogee of an intermediate orbit. The second firing of the OMS places the spacecraft into a suborbital coast phase and "free return" orbit for the desired entry interface. The flight conditions—range, flightpath angle, headings, and velocity—at entry resulting from this orbit will be selected to enable the orbiter to glide to a suitable landing site runway.

Abort to Orbit. This mode begins after the AOA point is passed and also would occur in the event of a main engine failure. The space shuttle continues to thrust with the remaining main engines to main engine cutoff and external tank jettison. The OMS thrusters fire twice, to insert the orbiter into orbit and then to circularize the orbit. The orbit coast time altitude and the coast time before the deorbit maneuver depend on when the abort was initiated. The deorbit, entry, and landing would be similar to a normal mission except for carrying the waste payload. For some failures very close to MECO, normal orbital operations could be accomplished.

7.3.1.6 Modifications for Space Disposal Mission

The only modification required is strengthening of the structure and landing gear of the orbiter to enable landing after abort while carrying the full 47,000-kg payload represented by the nuclear waste payload.

7.3.1.7 Dedicated Waste Payload Safety Provisions

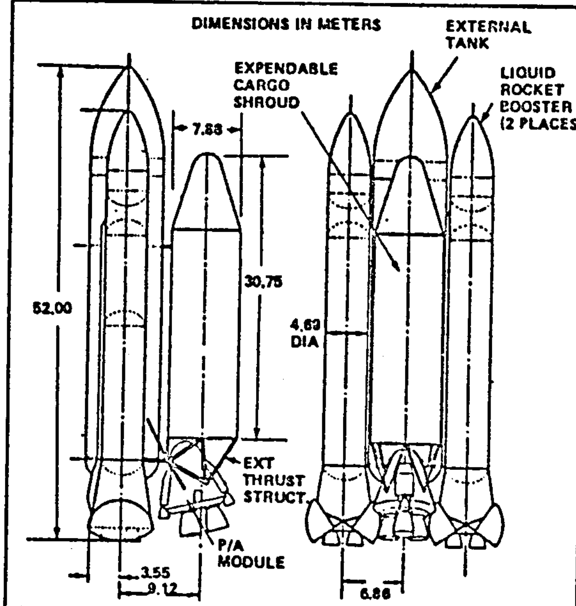
The only specific system added to decrease risk to the waste payload is the orbiter flotation system. The OFS will keep the orbiter and waste payload afloat in the event of a water landing to aid location and recovery.

The system is an inflatable ruggedized Kevlar bladder inflated by redundant gas generators which, when inflated, occupy empty space in the cargo bay, forward of the FSS. The inflated bladder is 4.57m in diameter and 7m long, providing an inflated volume of 119.82 m^3. It is secured to the orbiter payload bay longerons and the flight support system by steel cables and will support a combined orbiter and payload mass of .13,400 kg in a nose-up attitude. The orbiter already carries location beacons and will have a large radar cross section in the floated condition. The mass of the flotation system is estimated at 3,400 kg.

7.3.2 Orbit Transfer System Launch Vehicle

The SDV is used to transport the 80t orbit transfer system from KSC to the reference circular orbit at 370-km altitude and an inclination of 38 deg. The general arrangement, key dimensions, and characteristics of the SDV are illustrated in Figure 7.3.2-1.

GENERAL ARRANGEMENT



NOTE: P/A MODULE, EXT. THRUST STRUCTURE, SHROUD = CARGO ORBITER

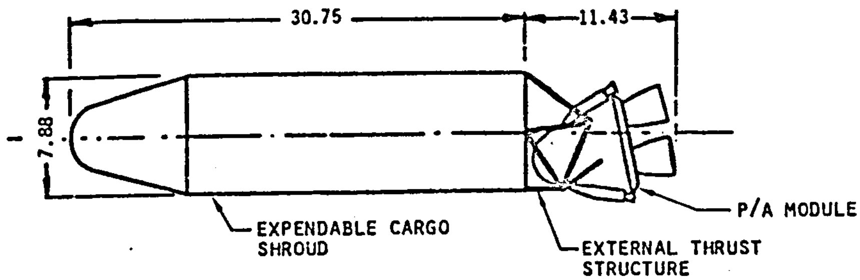
APPLICATION	ORBIT TRANSFER SYSTEM TRANSPORT TO LEO
PERFORMANCE CHARACTERISTICS *INCLUDES E.T., CARGO ORBITER, PAYLOAD	GLOW = 1.857×10^6 kg BLOW = 0.980×10^5 kg W_{P1} = 0.841×10^6 kg ULOW = 0.877×10^6 kg W_{P2} = 0.705×10^6 kg MASS @ MECO* = 0.171×10^6 kg
PAYLOAD TO 370 km ORBIT @ 33°	83,700 kg
CARGO BAY DIMENSIONS	7m DIA x 30m LONG
ABORT MODES	NONE DEFINED
MODIFICATIONS	SHROUD EXTENDED 6.86m IN LENGTH
SAFETY PROVISIONS	NONE DEFINED

Figure 7.3.2-1. Shuttle-Derived Vehicle General Arrangement

7.3.2.1 General Arrangement and Key Elements

The SDV uses the same ET and LRB's as the uprated shuttle. The two liquid rocket boosters flank the ET in an arrangement identical to the uprated shuttle, but the shuttle orbiter is replaced by a cargo orbiter which uses the same ET mechanical interfaces. The ET and LRB's are described in section 7.3.1.1. The remainder of this section describes the cargo orbiter.

Cargo Orbiter. The cargo orbiter is shown in Figure 7.3.2-2. It is a three-part vehicle designed to be directly interchangeable with the orbiter in terms of mechanical interfaces and main propulsion. Its larger volume for cargo and reduced inert mass compared to the shuttle orbiter allow transportation of cargo larger and heavier than can be carried by the shuttle. The cargo orbiter is composed of three major elements: the



P/A MODULE	28,381
CARGO SHROUD + EXT. THRUST STRUCT.	9,370
CARGO ORBITER INERT MASS	37,751

Figure 7.3.2-2. Cargo Orbiter General Arrangement

expendable cargo shroud, the external thrust structure, and the recoverable propulsion/avionics (P/A) module.

Expendable Cargo Shroud. The cargo shroud is an expendable passive cargo container. Its construction is aluminum honeycomb with intermediate support frames. Major frames are located at the forward ET attach point and at the aft end where the shroud is connected to the P/A module. The portion of the shroud aft of the forward ET attach point is carried to orbit. The remaining portion, including the conical nose cap is jettisoned during ascent as soon as Q-loads permit. This facilitates payload release on orbit and maximizes payload. Payload release on orbit is accomplished by translating the orbit transfer system forward until it clears the remaining portion of the shroud. A rail system guides the separation. The shroud is backed away by the P/A module after sufficient initial clearance with the orbit transfer system has been achieved.

External Thrust Structure. Structural connection of the cargo shroud to P/A module is accomplished with an external thrust structure of AS 3502 graphite-epoxy struts attached to the cargo shroud at four points. The aft end of the thrust structure attaches to the P/A module skin at four points. Release mechanisms at these points provide for separation of the P/A module. The thrust structure attaches to the ET with two fittings that duplicate those of the orbiter.

Following insertion of the payload into the prescribed orbit, the P/A module performs a retroburn to insert the cargo shroud into an orbit with a perigee low enough to ensure destruction during shroud reentry. The P/A module then separates from the cargo shroud and thrust structure and reinserts itself into circular orbit to wait for reentry and landing at the launch site.

P/A Module. The P/A module is designed to incorporate the main propulsion and propellant delivery systems from the STS orbiter along with the necessary avionics, OMS, and other systems in a ballistically recoverable capsule. Its shape is dictated primarily by aerodynamic balance requirements at entry. The P/A module shown in Figure 7.3.2-2 is also configured to be compatible with the orbiter TSM. The module's exterior geometry was determined by the location of the shuttle orbiter umbilical panels. The width of the orbiter body at the forward edge of the panel determined the maximum allowable diameter of the P/A module at that point. A retractable cover provides protection for the umbilical panel disconnects after separation from the TSM.

The capsule structure is an all-aluminum shell structure using skin/stringers with frames for stiffening. The thrust structure consists of titanium/graphite-epoxy beams which directly mount the three SSME's. The beams terminate at four points which provide mechanical attachments to the external thrust structure. The design of these attachments is similar to that of the orbiter/ET attachments. Design of the thrust structure accommodates passage of propellant feedlines.

The three SSME's of the P/A module are in the same relative location as in the current orbiter. The routing of the propellant feedlines has been revised to accommodate the relocated ET/orbiter disconnect valves at the module surface. The helium purge bottles and hydraulic power units for thrust vector control are also rearranged from the orbiter configurations to fit the module contour.

Auxiliary OMS propulsion for the P/A module is provided by two STS OMS engines with 26,688N of thrust each. The OMS engines are each mounted in a removable pod to enable servicing in the hypergolic maintenance facility. The required propellant is less than that of the shuttle orbiter because of reduced impulse requirements for the SDV mission, and this aids in the P/A module packaging. A thermal control system similar to that in the orbiter is required to prevent propellant freezing. Placement of the two OMS pods is above and below the SSME cluster on the P/A module vertical centerline.

Approximately 24 hr after launch, the P/A module deorbits and returns to a soft landing near the launch site. Reentry thermal protection is provided by shuttle-type reusable surface insulation (RSI). Initial aerodynamic deceleration is provided by drogue and main parachutes. Final deceleration for a soft landing is achieved by solid rocket motors (SRM) mounted on the base heat shield. The P/A module lands on three telescoping landing feet mounted on the aft end. The landing feet provide impact attenuation to accommodate deceleration system errors.

The P/A module avionics system is derived from the STS system by (1) eliminating all display electronics and voice communications, (2) eliminating pilot landing and navigation aids, (3) eliminating the aerosurface actuation system, (4) modifying the deceleration and landing system, and (5) reducing the number of computers from five to three. Onboard performance monitoring and recording equipment is incorporated to support post-flight refurbishment, maintenance, and other operations. Additional avionics are added to control the descent and recovery phases.

7.3.2.2 Launch Vehicle Characteristics

Launch vehicle basic characteristics are shown in Figure 7.3.2-1 and are identical to those of the uprated STS. The increased payload of the SDV is provided by the decreased inert weight of the cargo orbiter (37,751 kg) compared to the shuttle orbiter (75,000 kg) as the injection mass at MECO is constrained to be equal.

7.3.2.3 Performance

The SDV delivers 84,000 kg of payload to the reference 370-km-altitude circular orbit at 38-deg inclination.

7.3.2.4 Cargo Accommodation

The SDV cargo shroud, as defined for the space disposal mission, provides a clear volume, 6m in diameter by 25.5m long, topped by a truncated cone 6m in diameter at the base, 4.2m long, and 2m in diameter at the top surface. The orbit transfer system is supported by a 4.57m-diameter support ring at the base of the injection stage body shell, which takes X-, Y-, and Z-loads, and by three fittings at the station of the forward ET pickup point, which accommodate X- and Y-loads only. Separation is accomplished by a linear-shaped charge at the base interface and separation nuts at the forward fittings. Guidance during separation is provided by a rail system which guides rollers mounted on the injection stage.

7.3.2.5 Abort Modes

Abort modes for the SDV are undefined. In general, the payload could not be expected to survive any abort except an abort to orbit. The P/A module, by itself, could probably accomplish something similar to the RTLS, AOA, and ATO abort modes available to the orbiter.

7.3.2.6 Modifications for Space Disposal Mission

The only modification to the SDV configuration described in reference 3 was to extend the payload shroud length by about 7m to accommodate the orbit transfer system stack. The performance impact of this change is negligible for two reasons: (1) the extension is on a portion of the shroud subjected to aerodynamic loads only, and (2) the entire portion of the shroud forward of the ET interface ring, which includes the added length, is jettisoned early in the ascent trajectory.

7.3.2.7 Dedicated Waste Payload Safety Provisions

N/A.

7.4 ORBIT TRANSFER SYSTEM

The orbit transfer system is composed of a delivery orbit transfer system, which transfers the waste payload system from low Earth orbit to the final destination, and a rescue system which ensures delivery of the waste payload to the planned destination in the event of delivery mission failures. The rescue system is based on using the vehicles developed for the delivery mission, modified by the modular addition or "kitting" of specialized rescue systems.

This section is divided into two parts. In the first, major elements and subsystems used in the nominal delivery mission are described. In the second, additional components used in the rescue mission are described, along with the way they are integrated with the delivery mission elements to accomplish the rescue mission.

7.4.1 Delivery Mission Orbit Transfer System

The delivery mission orbit transfer system is used to transfer the waste payload from a 370-km altitude Earth orbit to the destination circular heliocentric orbit at 0.85 AU. The reference orbit transfer system is a two-stage vehicle that burns liquid oxygen and liquid hydrogen. The first stage, or injection stage, is recovered for reuse; the second stage, or SOIS, is expended. The orbit transfer system general arrangement and key characteristics are shown in Figure 7.4.1-1.

The four primary orbit transfer system elements are also shown in Figure 7.4.1-1: (1) The injection stage places the SOIS, payload adapter, and payload into heliocentric transfer orbit and returns for reuse. (2) The interstage supports the SOIS and payload adapter during ascent to low orbit and, during the injection stage burn, also supports the mass of the waste payload. It is jettisoned after SOIS separation. (3) The SOIS spends 165 days in transfer orbit and then performs a 12-min burn to insert the payload into the

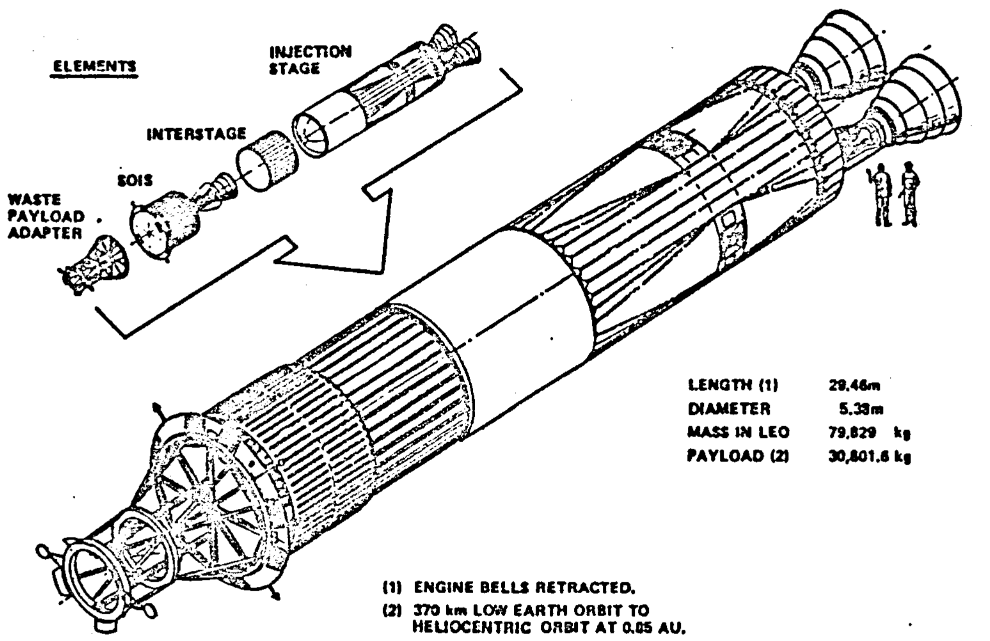


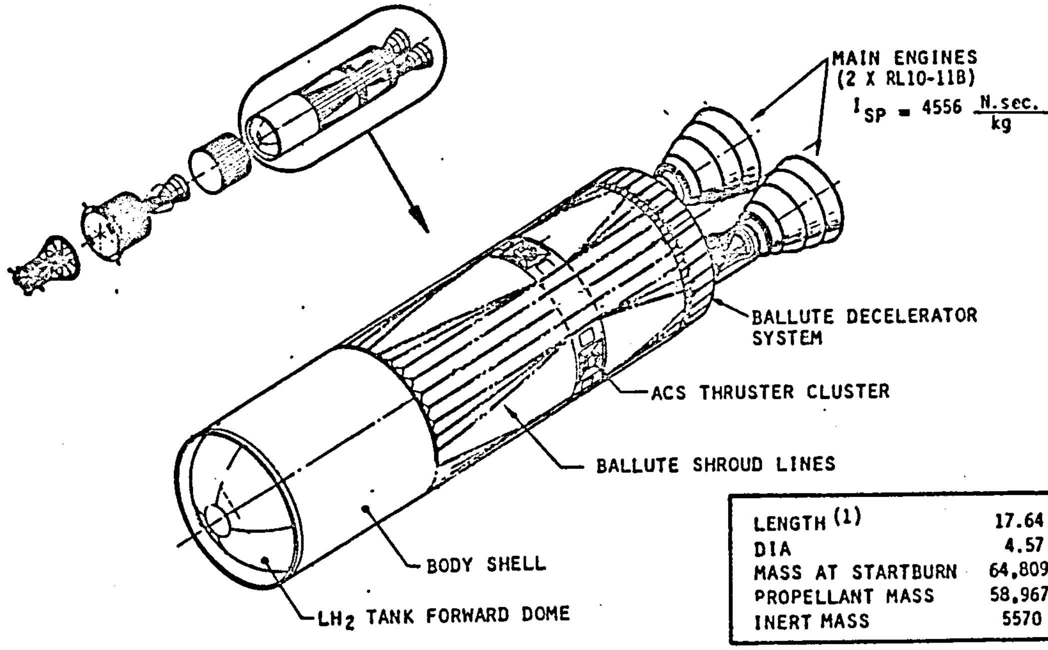
Figure 7.4.1-1. Orbit Transfer System General Arrangement

destination orbit. (4) The payload adapter allows the orbit transfer system to dock with the uprated space shuttle in orbit for transfer of the waste payload and provides for waste payload transfer and structural support.

7.4.1.1 Injection Stage

The injection stage is an LO_2/LH_2 propellant OTV that uses aerobraking to accomplish the reduction in velocity to circularize upon return to LEO. The general arrangement and key configuration features of the vehicle are illustrated in Figure 7.4.1-2, with top-level mass properties. A detailed mass statement for a closely related point design vehicle is presented along with mass trending data in Appendix C. The following sections briefly describe the vehicle and its systems. Primary systems include structure and airborne support equipment, aerobraking, thermal control, avionics, power supply and distribution, propulsion, attitude control, and interstage assembly.

Structure. All of the vehicle external body shell except for the avionics ring is fabricated from graphite-epoxy. The main propellant tanks are fabricated from 2219 aluminum 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 engines. Trunnion fittings made of titanium are used to interface with the launch vehicle. Pyrotechnically actuated payload release mechanisms



LENGTH (1)	17.64 m
DIA	4.57 m
MASS AT STARTBURN	64,809 kg
PROPELLANT MASS	58,967 kg
INERT MASS	5570 kg

(1) ENGINE BELLS RETRACTED

Figure 7.4.1-2. Injection Stage General Arrangement

are provided for interstage separation. The vehicle side of the vehicle-ASE interface has a total of 36 receptacle fittings for the vehicle latch and release mechanisms mounted on the ASE. Most of the electrical power, avionics, and attitude control systems (ACS) are mounted on the aluminum avionics and equipment ring.

Aerobraking. The aerobraking capability is achieved by the modular installation of a ballute subsystem on the aft body shell. It consists of the ballute, 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 reentry. A global positioning system (GPS) receiver and 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 passively cooled avionics are mounted on the 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 provided by an active cooling loop with a radiator mounted on the LO_2 tank support body shell. Freon 11 is used for the working fluid. Heaters are used on the ACS storage tanks, feedlines, and thrusters and for batteries and the fuel cell product water dump line. The cryogenic propellant tanks are covered with blankets composed of 23 layers of double-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. The avionics subsystem is identical to that of the Phase A OTV configuration (ref. 4) and performs 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 is a dual-string system which includes two computers and is communications compatible with both STDN and TDRS. Two GPS receivers are used to provide precise navigation for the aerobraking return maneuver.

Power Supply and Distribution. The electrical power supply and distribution subsystem, designed for 28V operation, is powered by redundant, low-pressure, modified orbiter H_2/O_2 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 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. Main propulsion is provided by two Pratt & Whitney RL10-IIB engines, which have a stowed length of 1.778m to facilitate stowage in the STS orbiter for recovery and provide a total of 66,720N of mainstage thrust. The main propellant tanks have usable capacities of 8423 and 50,543 kg of liquid hydrogen and oxygen, respectively. The propellant delivery system uses 0.057m delivery lines, tank sump-mounted prevalves, and 0.144m 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. A schematic diagram of the complete propulsion design is shown in Figure 7.4.1-3.

Attitude Control. The ACS uses hydrazine monopropellant with pressure blowdown positive expulsion, 12 IUS reaction engine modules (REM) for a total of 24 thrusters, and 6 propellant storage tank assemblies. Each of the six 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. The thrusters provide 133N of thrust with 2620 kPa inlet pressure and 36N at 690 kPa inlet pressure. Specific impulse is 235 and 230 sec at the 133N and 36N thrust levels, respectively. Propellant tanks, REM's, and all plumbing are mounted on the avionics ring section.

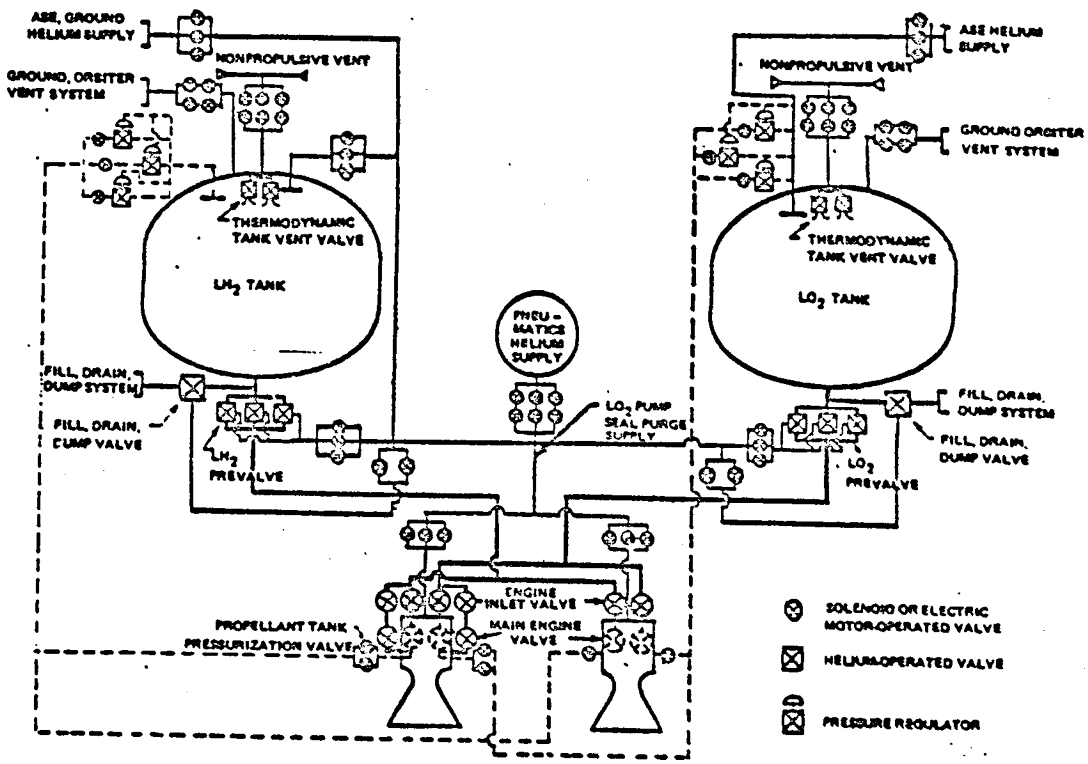


Figure 7.4.1-3. Main Engine Propulsion System Schematic

7.4.1.2 Interstage

The interstage assembly, which supports the SOIS during launch and ascent and both SOIS and waste payload during injection, consists of structure, separation, and wiring systems. The general arrangement and key characteristics of the interstage are shown in Figure 7.4.1-4.

The structure subsystem consists of a 4.57m-diameter graphite-epoxy honeycomb shell with rings at the forward and aft ends and longitudinal stiffeners also fabricated from graphite-epoxy composite. The 3.34m length of the interstage provides clearance between the retracted bell of the SOIS engine and the forward multilayer insulation (MLI) blanket of the injection stage LH_2 tank.

Eight separation fittings on each of the forward and aft rings accommodate explosive bolts used for OTV and SOIS attachment and separation. The wiring system links SOIS and injection stages.

Total mass of the interstage is 254 kg. A summary mass statement for the interstage assembly is contained in Appendix C.

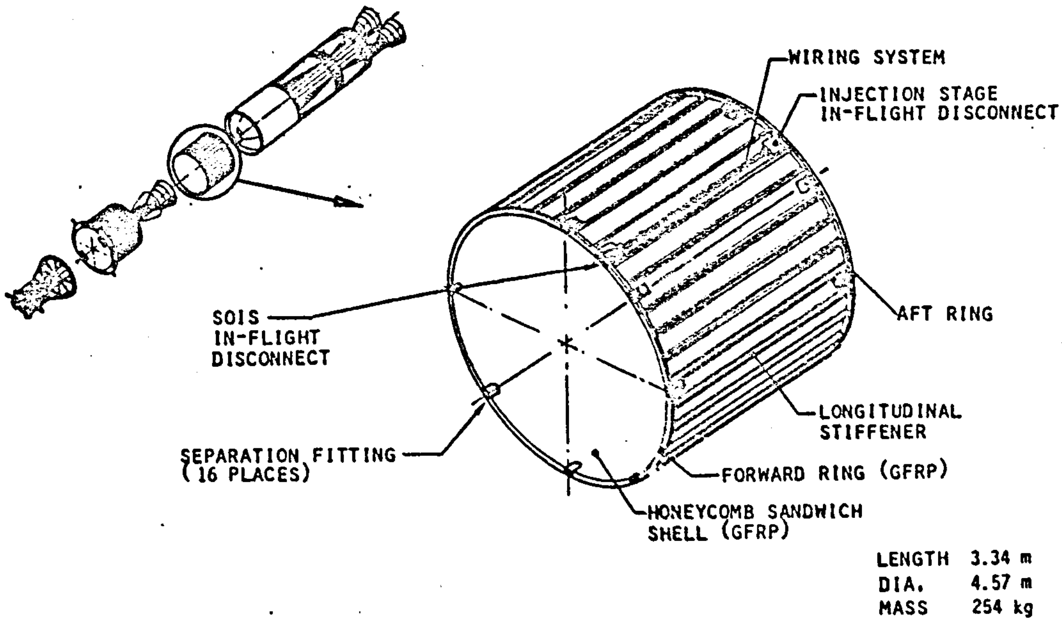


Figure 7.4.1-4. Interstage Assembly General Arrangement

7.4.1.3 Placement Stage (SOIS)

The 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 reached. The general arrangement and key features of the stage are illustrated in Figure 7.4.1-5.

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

<u>Structure</u>. Major SOIS structural elements include the body shell, avionics compartments, sunshield, and propellant tanks.

The cylindrical body shell provides structural support and meteoroid protection for the propellant tanks and avionics compartment. It interfaces with the orbit transfer system interstage at the rear and with the payload adapter at the front. The body shell is a honeycomb sandwich structure fabricated from 0.25-mm graphite-epoxy facesheets and nomex core. Forward and aft graphite-epoxy rings accommodate the interstage and payload adapter interface fittings. Intermediate rings provide mounting interfaces for the propellant tanks. Meteoroid shields of 0.25-mm aluminum are spaced 60 mm from the outer honeycomb facesheet to act as meteor bumpers for the propellant tanks. During the

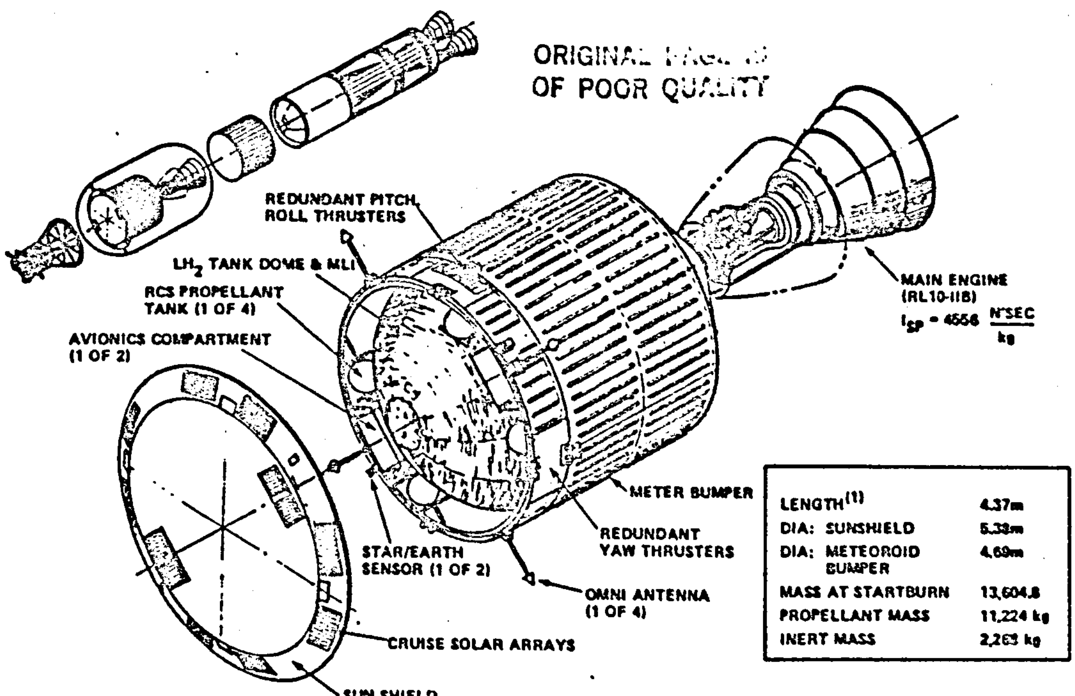


Figure 7.4.1-5. Solar Orbit Insertion Stage (SOIS) General Arrangement

165-day cruise, meteoroid protection for the oxygen tank is provided by a dedicated conical aluminum shield aft of the body shell.

Avionics are housed in two thermally isolated avionics compartments mounted to the inside of the body shell forward of the LH_2 tank. The compartment provides correct operating temperature for the avionics components without transferring heat into the propellant tanks. Each compartment is exposed to the solar flux at the forward end through penetrations in the sunshade and to space on the outboard side through a cutout in the body shell. The correct thermal balance is maintained using passive-thermal coatings, MLI, and heaters. MLI is used to isolate the relatively warm avionics compartments from the propellant tanks.

The sunshield protects the forward end of the SOIS from direct solar flux during the 165-day cruise phase. It overhangs the SOIS diameter sufficiently to prevent direct illumination of any other portion of the vehicle during the limit cycle excursions in pitch and yaw imposed by the attitude control system. The shield is a conical honeycomb sandwich structure fabricated from 0.1-mm glass-reinforced epoxy facesheet and nomex core. Its outside is covered with FOSR, and it is shielded by an MLI blanket from the rest of the SOIS.

The main propellant tanks are fabricated from 2219 aluminum. They are of all-welded construction and are sized for a single-mission service life. Fiberglass struts are used to support the liquid hydrogen tanks, and graphite-epoxy struts support the liquid oxygen tank and the main engine.

Thermal Control. Configuration changes in the thermal control subsystem used on the OTV are required because of the 165-day coast from 1 AU to 0.85 AU. The stage remains oriented during coast in a head-on attitude to the Sun. The sunshield mounted on the front end of the vehicle reduces the incident heat flux. The number of MLI layers has been increased from 23 to 40 to prevent excessive boiloff. Avionics thermal control is provided passively by the thermally isolated avionics compartment. No active thermal control is required because solar array and batteries, rather than fuel cells, provide electrical power.

Avionics. This subsystem accomplishes all guidance and control functions and, in cooperation with the ground, all navigation functions. It provides communications with the ground and handles interfaces with the orbiter-mounted flight support system during waste payload transfer and with the injection stage and SDCLV.

The avionics system was derived from systems defined for the third stage of the NASA inertial upper stage (IUS) studied by BAC, with modifications to provide for longer mission duration and increased reliability. A basic diagram of the system is shown in Figure 7.4.1-6. The avionics system is composed of a guidance and navigation sensor group, data management group, communications group, rendezvous support group, and power control group.

Guidance and Navigation Sensor Group. Redundant sensors are provided for Sun, stars/Earth, and linear acceleration. One of each type of sensor is mounted in each avionics compartment in locations that satisfy FOV requirements. Each sensor interfaces with all three command units.

Data Management Group. The data management group consists of three command units (CU) and three signal interface units (SIU). Each CU consists of a digital processor, memory, timer, an integral power supply, and power switching and distribution relays. Driving circuits for the RCS valve drivers are integral with the unit. Each CU is capable of executing all vehicle control functions independently. In normal operation, one unit is prime and two are dormant. Redundancy management is accomplished by ground command.

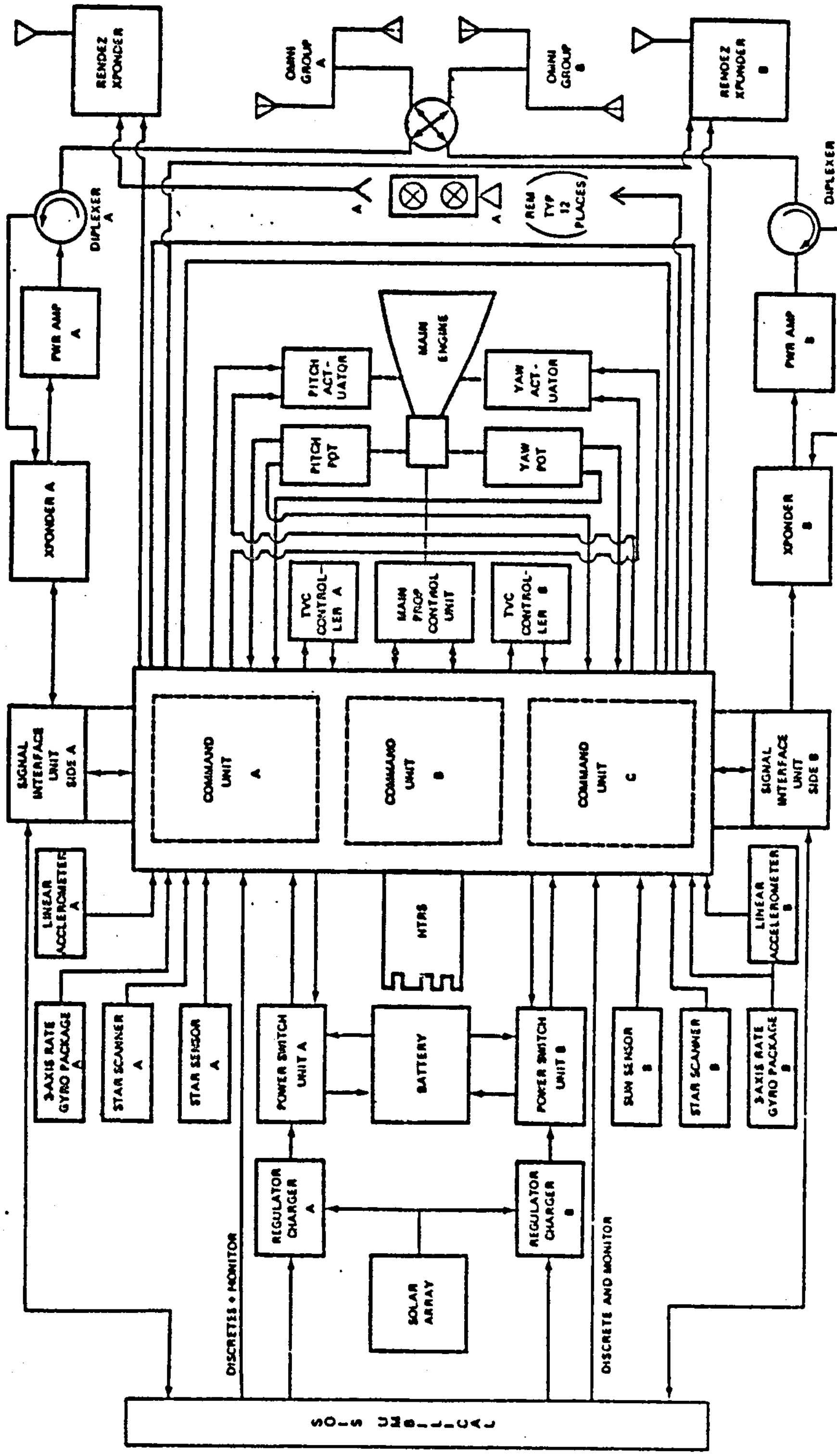


Figure 7.4.1-6. SOIS Avionics Block Diagram

The three SIU's interface the CU with the communications group transponders and with the hardwired signals transmitted throughout the SOIS umbilical. Each SIU interfaces with a single CU and with both communications groups and the SOIS umbilical.

Communications Group. The communications group provides redundant communication between the SOIS command units and ground control. It consists of two transceivers, an RF switch, and four omni antennas. Each transceiver consists of an IUS transponder and power amplifier and a diplexer. The SIU's of the data management group interface with the transponder. Both diplexers are connected to the RF switch which allows either transceiver to talk through either of two omni antenna pairs. Each pair of omni's provides a 4-pi steradian FOV for SOIS communications. Space disposal control Earth station will have sufficient ERP to talk to the omni's during all stages of the nominal disposal mission. Use of the NASA Deep-Space Network (DSN) on a contingency basis would allow communications and tracking at the maximum range (about 2 AU) reached during rescue missions.

Rendezvous Support Group. Redundant rendezvous transponders are provided to aid waste payload transfer and rescue operations (if required). The transponders are mounted on the payload adapter to satisfy FOV requirements. Either transonder can be switched by any of the three CU's.

Power Control Group. The power control group drives the redundant main facing thrust vector controllers and handles the main engine control functions. The redundant thrust vector controllers interface with the three CU's and the pitch and yaw actuators on the main engine. Each thrust vector control (TVC) controller can drive both actuators. The main engine controller is a dedicated unit which sequences the integral solenoid valves on the RL10-IIB engine to run the engine through its start (and shutdown) sequences after being initiated by either of the three CU's.

Power Supply and Distribution. Figure 7.4.1-6 also shows a block diagram of the electrical power system. Cruise 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 coverglass, and 2-mil substrate. Two 89-Ah batteries provide redundant power storage. Dedicated regulator charger and power switching units are associated with each of the three CU's to provide high reliability. Batteries, regulator charger, and switching units are mounted in the avionics compartments. The solar array is mounted on the sunshield outer skirt.

Propulsion. Main propulsion is provided by a single Pratt & Whitney 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 1663 and 9561 kg of liquid hydrogen and oxygen, respectively. Figure 7.4.1-7 shows a schematic of the main propulsion system. The

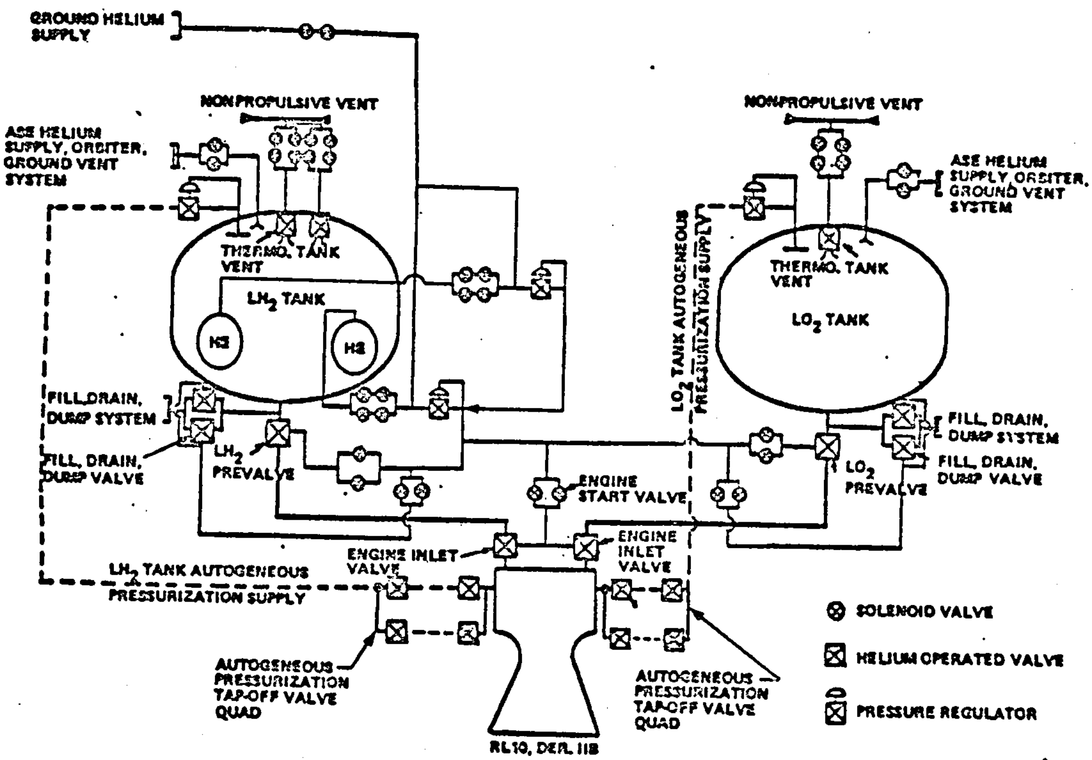


Figure 7.4.1-7. SOIS Main Propulsion System Schematic

propellant delivery system uses 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 vent systems are provided. Reliability has been augmented for the 165-day cruise mission by providing a redundant helium supply and regulators, LH_2 thermodynamic vent valves, and autogenous pressurization tapoff valves.

Attitude Control. The ACS uses hydrazine monopropellant with pressure blowdown positive expulsion. The ACS uses 12 IUS REM's with two thrusters each and four propellant storage tank assemblies. Two clusters with eight thrusters each located on both the z-axis and the y-axis provide redundant attitude control and three-axis translation capabilities.

Each of the four 0.533m-diameter titanium propellant 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. The thrusters provide 133N of thrust with 2620 kPa inlet pressure and 36N at 690 kPa inlet pressure. Specific impulse is 235 and 230 sec at the 133N and 36N thrust levels, respectively. Propellant tanks, REM's, and all plumbing are mounted on the inside of the body shell forward of the LH_2 tank (Fig. 7.4.1-5).

7.4.1.4 Payload Adapter

The payload adapter mounts to the front of the SOIS and has four primary functions: (1) It provides a passive docking collar which allows the OTV to dock with the orbiter in LEO to allow waste payload transfer. (2) It provides guide rollers to aid waste payload transfer from the orbiter-mounted FSS. (3) It provides structural support for the waste payload during injection, transfer orbit cruise, and placement operations. (4) It provides two mechanical latches and a trunnion socket to reversibly secure the waste payload in position.

Primary elements of the payload adapter designed to perform these functions are shown in Figure 7.4.1-8 and include the docking ring assembly, waste payload transfer

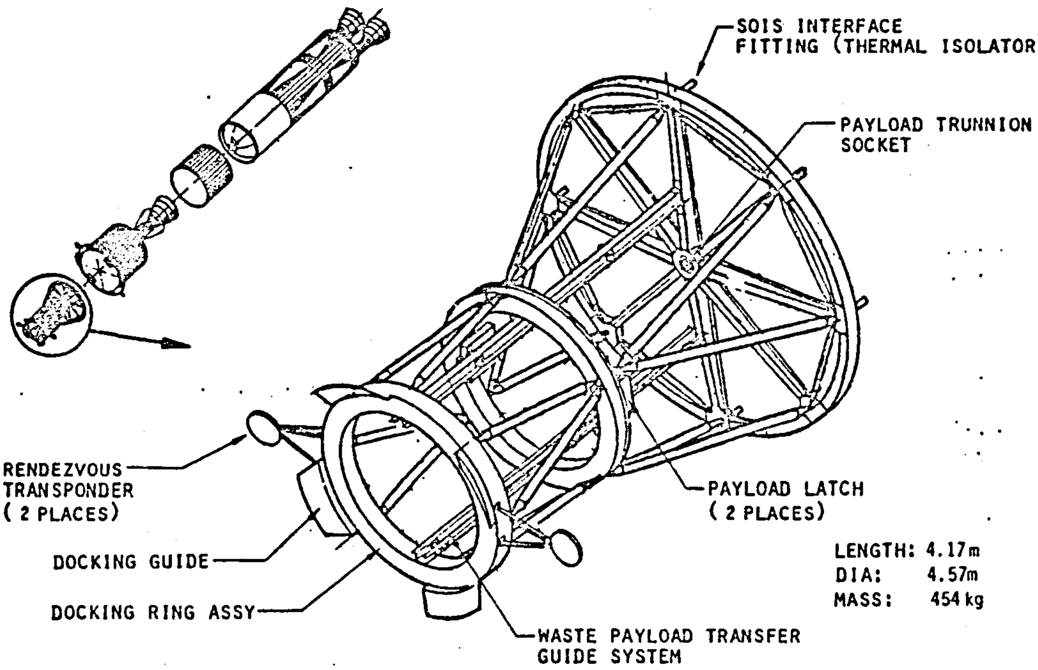


Figure 7.4.1-8. Payload Adapter Primary Elements

guide system, reversible waste payload latch system, structure, thermal control, and the SOIS interface fittings.

The docking ring assembly includes docking guides and capture and hard-docking latch strikers. No active components are included. Components are integrated with the box section docking ring which provides structural support for all components used in docking. The ring also supports the two rendezvous transponders.

Two rails equipped with eight rollers each engage the waste payload system guide rails to guide the waste payload during normal and rescue mission waste payload transfer operations. The rails are supported by the docking and payload interface rings. The rollers are dry lubricated to ensure operation after extended exposure to the space environment.

The reversible waste payload latch system is designed to provide passive, fail operational latching of the waste payload during normal operation and to allow mechanically operated unlatching during rescue operations. The normal latching operation is shown in Figure 7.4.1-9. Two spring-loaded jaws capture the waste payload lateral

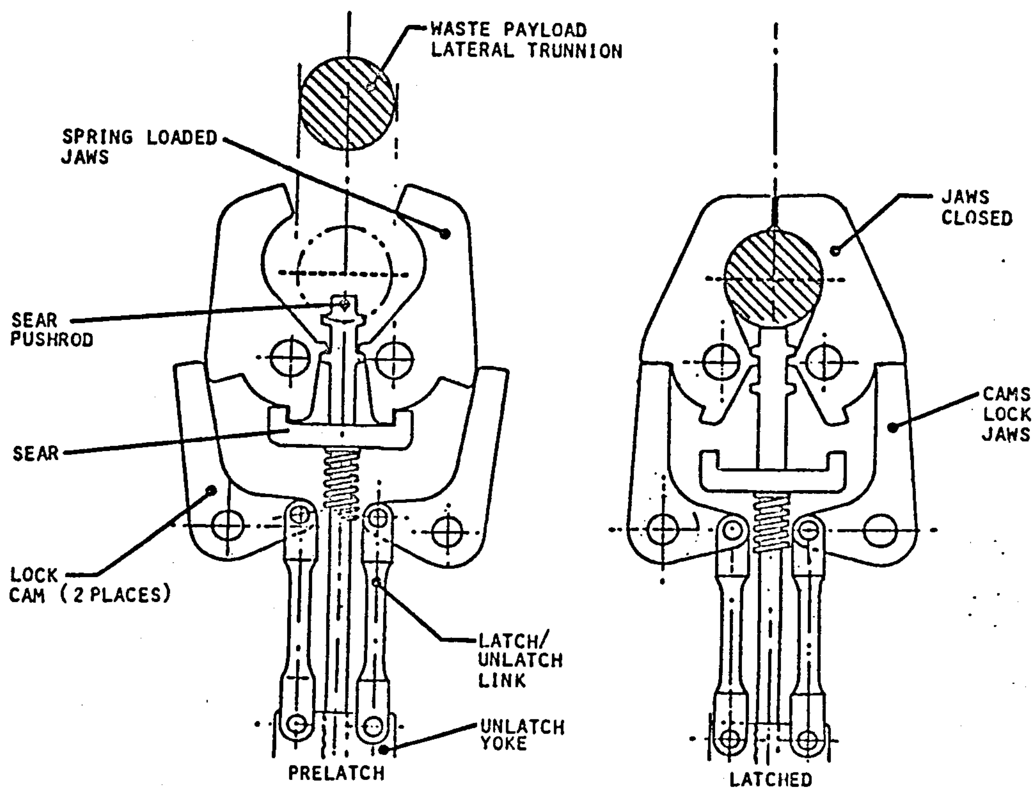


Figure 7.4.1-9. Payload Adapter Reversible Waste Payload Latch System

trunnion when the trunnion displaces the sear which holds the jaws open. Two cams linked to the displacement bar rotate into positions which lock the jaws in the closed position as the trunnion moves into capture position. Jaws are unlocked only if required by a rescue. To prevent inadvertent unlatching, the locking cams can be retracted only by a shaft-driven mechanism. Because the SOIS does not carry a drive for the shaft, unlatching can only be accomplished by a dedicated drive mechanism carried on the docking ring of the rescue SOIS.

The two latches are mounted on the payload interface ring and carry all waste-payload-induced X- and Z-loads from the waste payload lateral trunnions into the ring and adapter structure. Y-loads are reacted by the trunnion of the aft waste payload which fits into a trunnion fitting, located in the center of the SOIS interface structure. Reaction loads in the X-direction, caused by the offset between the waste payload trunnion and center of gravity, are reacted by the latch fittings.

Structural support for docking, payload guide, and latch mechanisms is provided by the space frame adapter structure. Primary structural elements include the docking and payload interface rings, the SOIS interface ring assembly, and the struts which make up the fully triangulated space frame connecting them. All components except the docking ring, which is fabricated from aluminum, are graphite-reinforced epoxy. Eight fittings on the periphery of the SOIS interface ring interface with thermal isolators to mount the adapter to the forward ring of the SOIS body shell.

Payload adapter thermal control is required to control structure interface dimensions and to minimize heat transferred into the SOIS. Dimensional control is established by limiting thermal excursions using passive coatings and selective application of MLI blankets. Thermal input to the SOIS is minimized by adjusting the adapter thermal balance to establish a low equilibrium temperature and by using glass-fiber-reinforced epoxy thermal isolators to connect the adapter SOIS interface ring to the SOIS body shell.

7.4.2 Rescue Mission Orbit Transfer System

The rescue orbit transfer system is a derivative of the delivery mission orbit transfer system modified to accommodate the increased duration and more complex operations of the rescue mission. It consists of a delivery mission orbit transfer system plus a standard SOIS modified by addition of a rescue kit. These elements are assembled into three distinct configurations used in executing the mission: (1) the rescue vehicle, which provides mission control after insertion for all phases and provides for rendezvous with the failed vehicle, transfer of the waste payload, and final solar orbit insertion; (2) the pursuit configuration which carries the rescue vehicle to the target after injection; and (3) an injection configuration which injects the pursuit configuration into its initial transfer orbit.

7.4.2.1 Rescue Vehicle

Rescue Vehicle Requirements. Rescue mission requirements which differ from the requirements of the standard delivery mission are summarized in Figure 7.4.2-1. Navigation to within terminal acquisition range of the target vehicle involves two propulsive

RESCUE VEHICLES

- PLACE RESCUE SYSTEM AT FAILED VEHICLE
  - 2 PHASE TRANSFER
  - TWO 1.2 KM/SEC VELOCITY IMPULSES
  - UP TO 308 DAYS LAUNCH TO RENDEZVOUS
- NAVIGATE TO WITHIN TERMINAL ACQUISITION RANGE OF TARGET VEHICLE
- ACCOMPLISH AUTONOMOUS 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
- INITIATE SOIS AND JETTISON RESCUE PECULIAR HARDWARE
- SOIS COMPLETES PLACEMENT AT 0.85 AU

FAILED SOIS

- BEACON TRANSPONDER SURVIVAL UNTIL RESCUE
- MAINTAIN STABLE ATTITUDE DURING TERMINAL RENDEZVOUS AND DOCKING

Figure 7.4.2-1. Rescue System Top-Level Requirements

maneuvers accomplished in the pursuit configuration using the deep-space network to track the onboard beacon transponder. Target vehicle and rescue vehicle relative positions are monitored and the maneuvers required for closing are calculated on the ground and uplinked to the rescue vehicle. The initial navigation phase is completed when the rescue vehicle arrives within 1000 km of the vehicle to be rescued.

Requirements peculiar to the rescue vehicle include rendezvous and docking with the failed SOIS, waste payload transfer to the rescue vehicle, and orientation of the rescue vehicle to the placement burn orientation, 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 these operations is approximately 18 hr.

This sequence dictates 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 requires 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 placement operations.

Rescue Vehicle Configuration. The rescue vehicle design uses a standard SOIS for the insertion maneuver. Additional capabilities required for rescue are provided by equipping the vehicle with a rescue kit. Elements of the rescue kit are illustrated in Figure 7.4.2-2 and include the rescue avionics ring, the active docking adapter, and the aft sunshade.

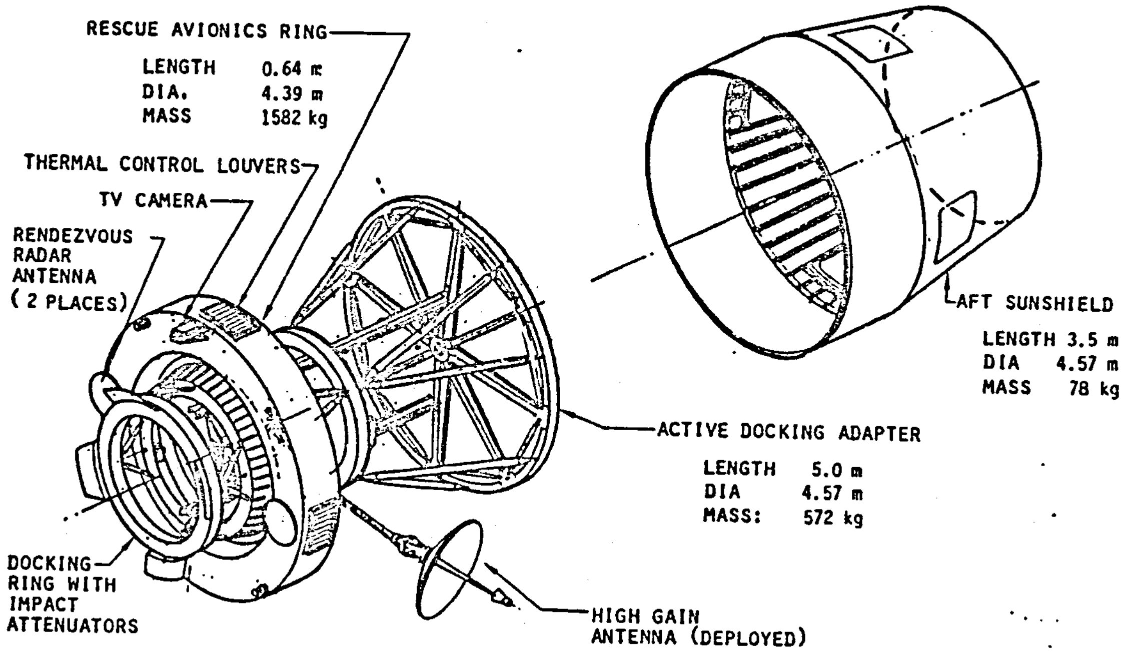


Figure 7.4.2-2. SOIS Rescue Kit

Rescue Avionics Ring. The rescue avionics ring internal arrangement, equipment complement, and summary mass properties are illustrated in Figure 7.4.2-3. Primary components include the reaction control system, propellant tankage, communication 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 used by the 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 active docking adapter.

- **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.	378

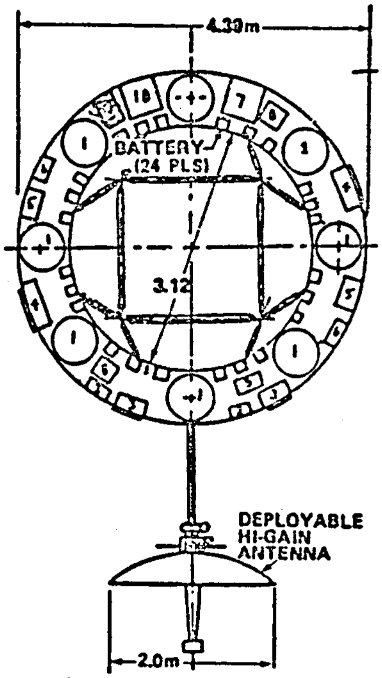


Figure 7.4.2-3. Rescue Avionics Ring

Active Docking Adapter. The active docking adapter, in addition to providing the waste payload support and waste payload transfer functions of the waste payload adapter, has its docking ring mounted on extendable impact absorbers to allow the rescue vehicle to perform an active role in docking. Motorized actuators draw the docking ring into a hard-docked configuration after the initial shock-absorbed contact. In the hard-docked condition, electrical and mechanical interfaces are completed with the docking ring of the failed SOIS. The mechanical interface, along with a drive motor on the active docking adapter, allows driving the shaft which unlatches the failed vehicle's waste payload adapter payload retention latches. An electric-motor-driven cable and winch system then draws the waste payload into the rescue-vehicle-mounted active docking adapter.

The adapter configuration with the avionics ring installed is shown in Figure 7.4.2-3. Its ring and space frame construction and SOIS interface features are identical to the normal waste payload adapter. Addition of the active docking ring and its associated support hardware adds 95 kg to the 454 kg of the base adapter; the waste payload transfer mechanization adds a further 23 kg for a total of 372 kg.

Aft Sunshield. The aft sunshield provides thermal control for the vehicle cryogenic propellant during the Sun-oriented coast portion of the pursuit mission when the negative x-axis of the rescue vehicle is pointed at the Sun. The sunshield also mounts a solar array and redundant Sun sensors used for vehicle power and pointing during the cruise portion of the mission. The sunshield is of honeycomb sandwich construction using glass-fiber-reinforced epoxy face sheets and nomex core. The conical portion of the shield is stiffened. Thermal control is provided by passive thermal coatings on the outside and extensive use of multilayer insulation blankets. Total mass of structure, thermal control, supplementary power, and Sun sensors is 78 kg.

Gross weight of the rescue kit is about 2232 kg. The kit dry mass including avionics ring, adapter, and sunshield is about 1821 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 Assembly. Assembly of the rescue kit and SOIS into a rescue vehicle is illustrated in Figure 7.4.2-4. The rescue kit is strut mounted to the active docking adapter. At the conclusion of rescue operations, the avionics ring and sunshield are

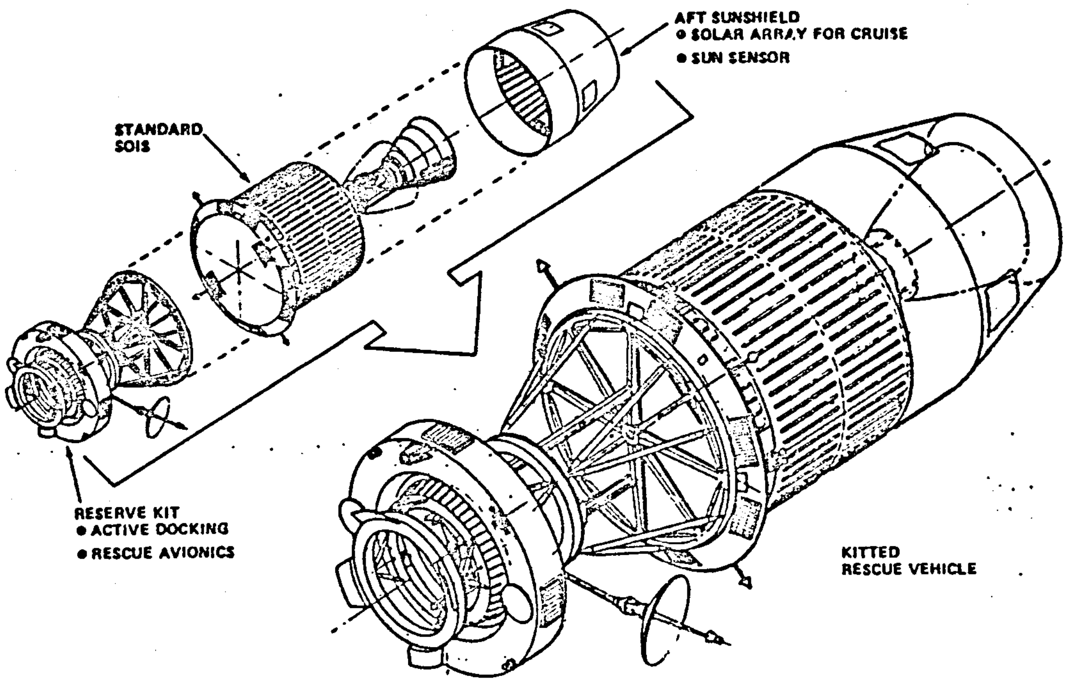


Figure 7.4.2-4. Rescue Vehicle Assembly

jettisoned, converting the rescue vehicle back to a standard SOIS.

Key features of the assembled rescue vehicle, including a summary mass statement, are shown in Figure 7.4.2-5.

The aft sunshield encloses the retracted engine bell to provide the thermal control needed to minimize boiloff of cryogenic propellant during the cruise and pursuit phases of

STANDARD SOIS 13605 kg

- -MAIN PROPULSION
- -ACS THRUSTERS
- -STAR SENSORS

WASTE PAYLOAD ADAPTER 454 kg

-PLUS-

RESCUE KIT TOTAL: 1773 kg

AVIONICS RING 1582 kg

- -TARGET ACQUISTION
- -RENDEZVOUS AND DOCKING
- -NAVIGATION
- -GROUND MONITORING
- -ADDED RCS PROPELLANT
- -ADDED ELECTRICAL POWER
- -JETTISONABLE

ACTIVE DOCKING RING 95 kg

- -IMPACT ATTENUATION

PAYLOAD TRANSFER MECH. 23 kg

AFT SUNSHADE 78 kg

- -THERMAL CONTROL
- -SUPPLEMENTARY POWER
- -SUN SENSORS

RESCUE VEHICLE GROSS 15837 kg

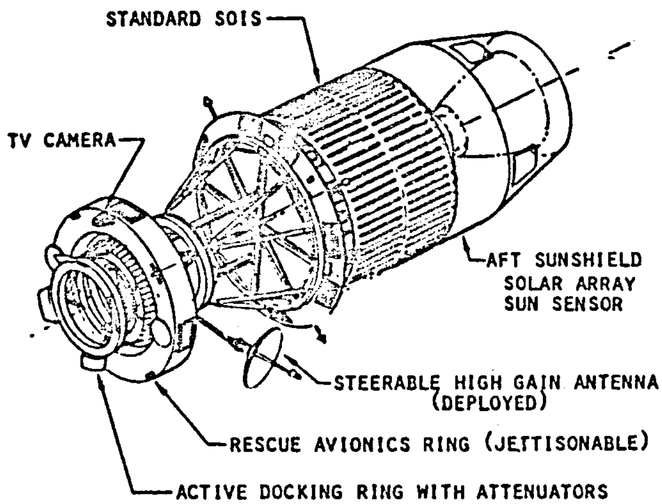


Figure 7.4.2-5. Rescue Vehicle Key Features

the mission when the sunshield is kept pointed at the Sun. The steerable high gain antenna of the avionics ring, shown in the deployed position, is used for the high data rate downlink required for closed-circuit television monitoring of the rescue operations. The active docking ring is shown in deployed condition. The similarity of the overall configuration to the standard SOIS illustrates the straightforward nature of the conversion.

The mass of the standard SOIS is increased from approximately 13,605 kg to about 15,837 kg by the addition of rescue provisions. The largest increment is provided by the 1582-kg avionics ring. The active docking adapter adds 572 kg and the aft sunshade adds 78 kg.

7.4.2.2 Pursuit Configuration

Pursuit Configuration Requirements. The basic mission profile for rendezvous with the failed SOIS in heliocentric orbit is described in detail in section 6.2.2 and illustrated in Figure 6.2-2. The reference trajectory requires two velocity changes of 1.19 km/sec each. The first velocity impulse at injection plus 154 days places the rescue vehicle in the pursuit trajectory which phases it for target vehicle intercept at 308 days. A second velocity impulse at this point matches velocity with the target vehicle, leaving the rescue vehicle in the same orbit as the target vehicle in preparation for rendezvous.

C-2

Pursuit Vehicle Configuration. The pursuit vehicle configuration uses a standard SOIS docked to the rescue vehicle, described in section 7.4.2.1, to provide the pursuit and rendezvous velocity impulses. At the conclusion of the rendezvous insertion burn, the expended standard SOIS is separated, leaving the rescue vehicle (now in the configuration shown in Fig. 7.4.2-5) ready for rendezvous and waste payload transfer.

The pursuit configuration described is illustrated in Figure 7.4.2-6 and consists of the rescue vehicle docked to the waste payload adapter of the standard SOIS. After

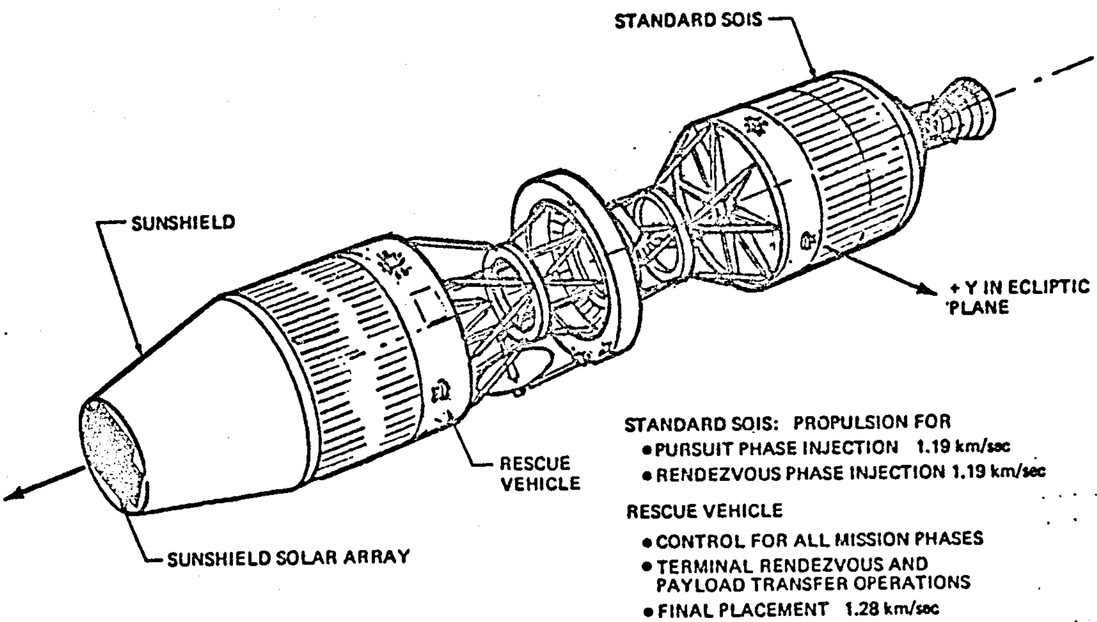


Figure 7.4.2-6. Pursuit Vehicle Configuration

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 capability 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 6.2-1.

7.4.2.3 Injection Configuration

Injection Configuration Requirements. The injection configuration is used to inject the pursuit configuration into the rescue transfer orbit. The mission profile is virtually identical to the standard delivery mission. The slightly increased injection delta-V (3.5 km/sec versus 3.27 km/sec) is more than offset by the substantial decrease in injected mass (20,150 kg versus 45,114 kg), allowing a substantial performance reserve.

Injection Configuration Description. The injection configuration of the rescue orbit transfer system is illustrated in Figure 7.4.2-7. The injection configuration is assembled on orbit from a 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 orbit transfer system, the rescue vehicle is deployed from the orbiter and in the first exercise of its

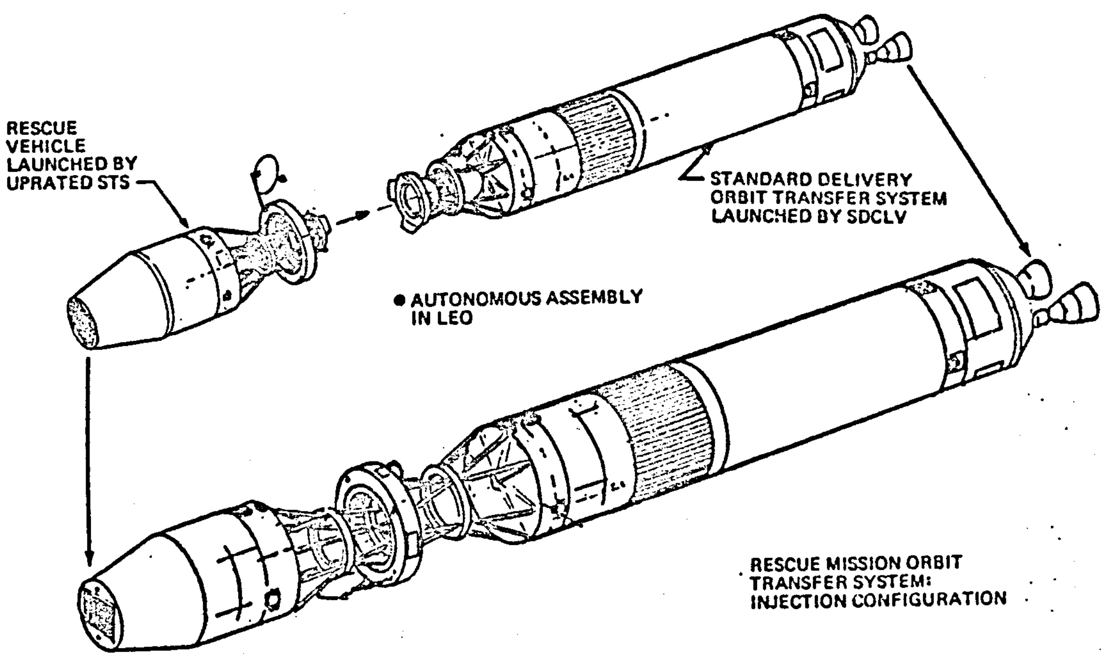


Figure 7.4.2-7. Injection Configuration of Rescue Orbit Transfer System

functions, which serves as a final checkout, rendezvouses and docks with the standard delivery orbit transfer system. Injection of the pursuit configuration to its initial transfer orbit is then accomplished by the injection stage which uses an aerobraking maneuver to return to low orbit for recovery by the orbiter.

7.5 FLIGHT SUPPORT SYSTEM

The waste payload flight support system serves three primary functions in association with the interpayload support structure described in section 7.1. 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 7.5-1. Two built-up titanium T-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.

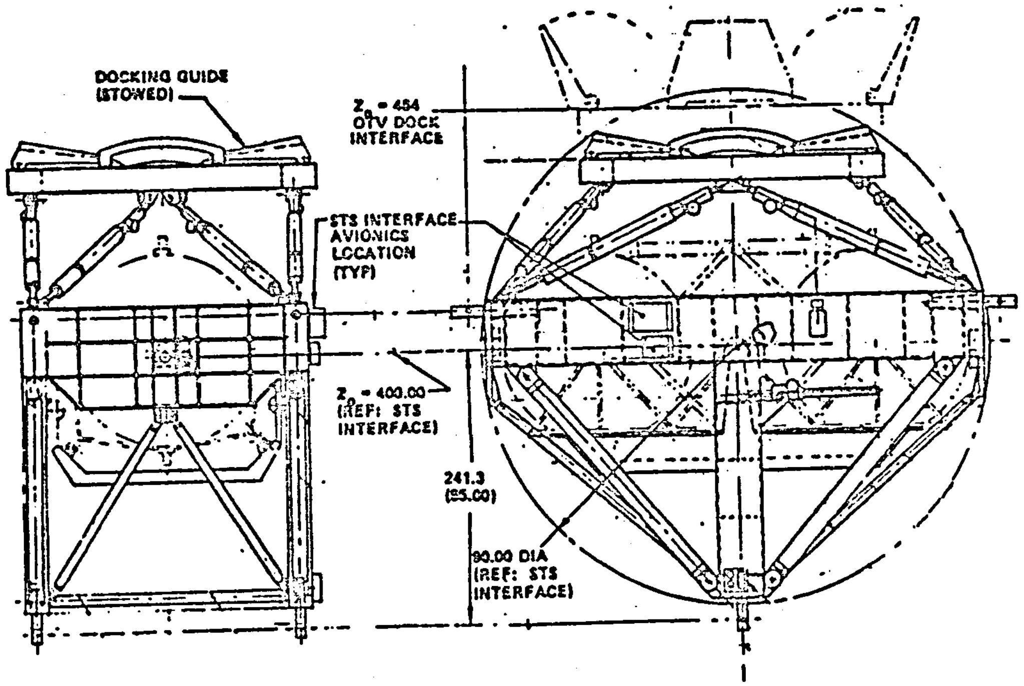


Figure 7.5-1. Flight Support System General Arrangement

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

A tilt table, driven by two linear actuators, rotates 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 T-frame.

Operation of the flight support system is illustrated in Figure 7.5-2. During operation, the orbit transfer system docks to the extended docking collar. The STS orbiter performs the active role in this docking sequence. Waste payload is transferred by rotating the transfer cradle 90 deg, using the linear actuators, allowing the waste payload

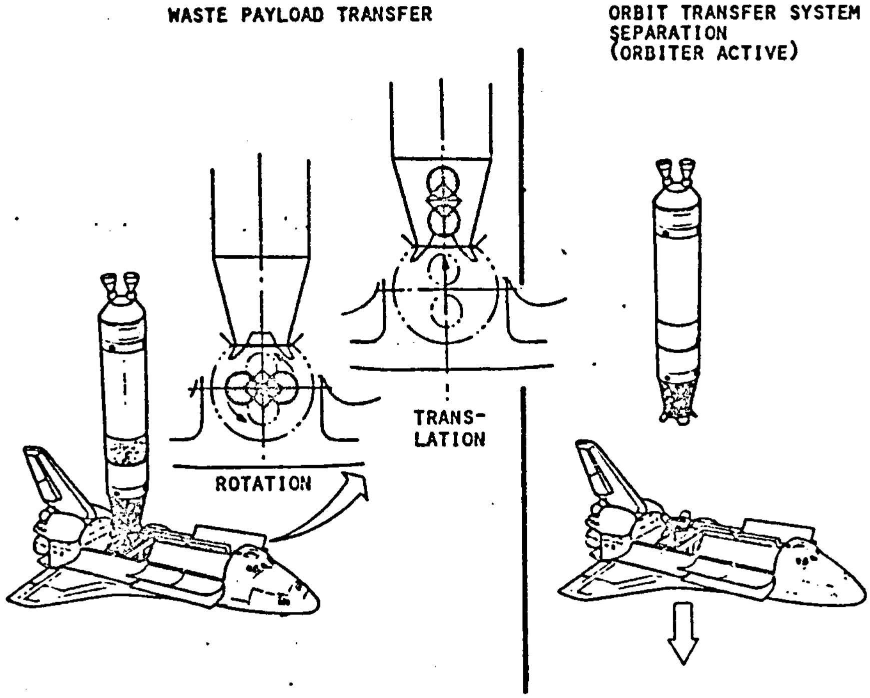


Figure 7.5-2. Flight Support System Operation Detail

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 transfer to the destination.

8.0 REFERENCE SPACE SYSTEMS OPERATIONS DEFINITION

The objective of this effort was to define operations to the level required in order to:

1. Determine the impact of assembly operations on the design of the waste payload system.
2. Define the major ground support facilities required at the launch site.
3. Define the complete timeline for all flight operations involving the waste payload in the nominal delivery mission.
4. Define those operations and sequences involved in conducting the nominal rescue mission.

The approach was to base the waste payload assembly operations on manufacturing sequences for the reference waste payload defined in section 4.0. The 1980 MSFC study of space disposal was used as the basis for setting launch site facility requirements. Timelines for the nominal delivery mission were adapted from actual timelines for the STS and IUS, modified for the space disposal trajectories. The sequence of rescue operations was derived from those developed in the 1980 MSFC space systems study.

8.1 WASTE PAYLOAD FABRICATION AND ASSEMBLY OPERATIONS

The sequence of operations involved in fabrication and assembly of the waste payload is illustrated in Figure 8.1-1. Discrete operations are enclosed in the individual boxes. Bulleted headings below the boxes describe fixtures and equipment used in the operation; bulleted headings above each box describe the facility at which the operation takes place.

Operations begin with fabrication of core and shield components at the shield fabrication vendor's facility. Shield tiles have been assembled to the Inconel shield shell and the assembled halves are transported to the waste payload fabrication and assembly facility, which is assumed to be colocated with the facility where the waste form billets are fabricated.

The first step in waste payload assembly is the loading of the waste form billets into the core, followed by the sealing of each bore. These operations are accomplished using a billet loading machine at the core load station. Following billet loading, the loaded core is transferred to the shield assembly station where the core is installed into the lower half of the shield assembly. The upper shield half assembly is then dropped into place and the

ORIGINAL PAGE IS
OF POOR QUALITY

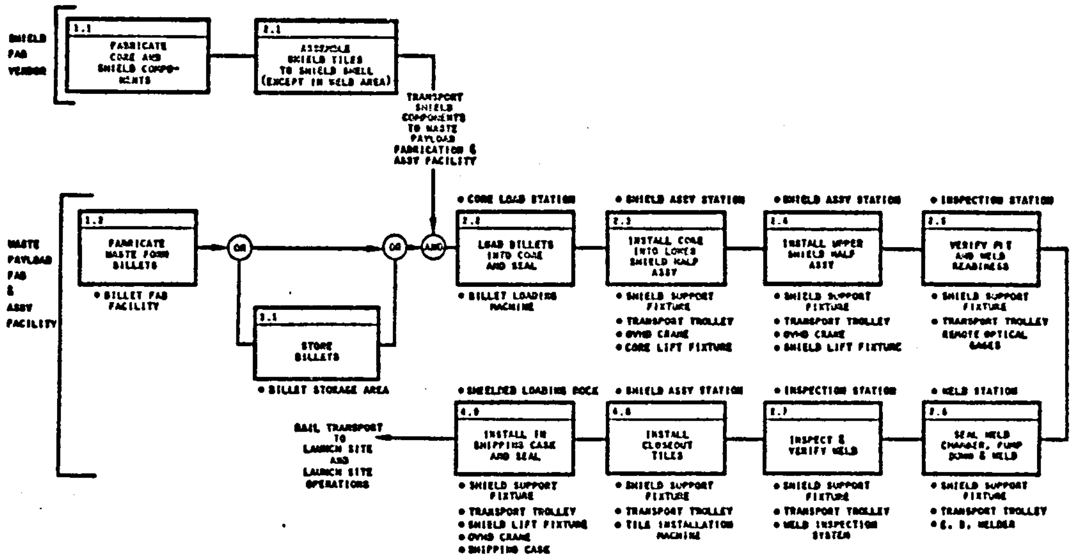


Figure 8.1-1. Waste Payload Fabrication and Assembly Flow Diagram

waste payload is transferred to the inspection station where the fit of the shield halves is verified, clearing the waste payload for welding operations. Following transfer to the weld station, the electron beam welder vacuum chamber is pumped down and the waste payload is rotated as the electron beam welds the two shield halves together. From the weld station, the welded waste payload is transferred back to the inspection station where the electron beam weld is inspected and verified as ready for flight. Following inspection, the waste payload is returned to the shield assembly station where the final row of closeout tiles, left uninstalled to allow access for the electron beam during the welding operation, is now installed. Closeout tile installation is followed by transfer of the waste payload to the shielded loading dock where it is installed in the shipping cask for shipment to the launch facility by rail.

Figure 8.1-2 is a schematic of a conceptual waste payload assembly facility. No attempt has been made to illustrate the billet fabrication or storage facilities. The path of the waste payload core is illustrated, beginning at the core load station to the shield assembly station, inspection station, and weld station. Following the weld, the waste payload moves through the stations in reverse order and is loaded into the shipping cask in the shielded loading dock. Waste payload transfer is accomplished by rail system and a shielded overhead crane. An adjacent unshielded area provides for receiving and inspection of the shield and core components and for control of the waste payload assembly operations.

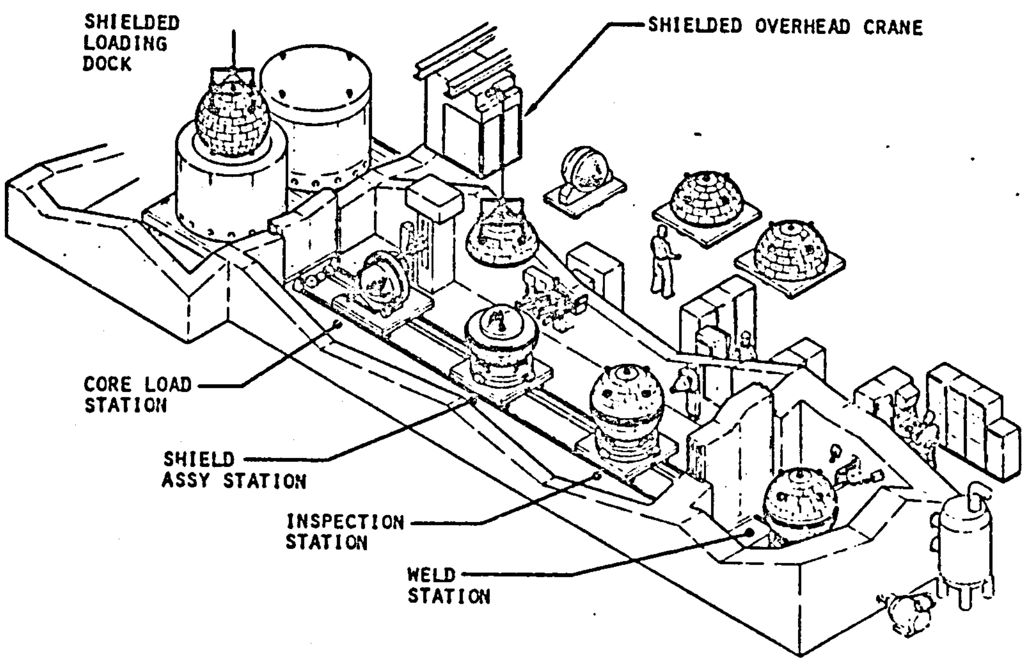


Figure 8.1-2. Conceptual Waste Payload Assembly Facility Schematic

8.2 LAUNCH SITE OPERATIONS

Launch site operations are divided into three primary phases. The first is conducted in the nuclear payload processing facility, where the individual waste payload assemblies are joined by the interpayload support structure to form complete waste payload systems. The second phase is mating of the integrated waste payload system to the orbiter component of the uprated space transportation system. The final phase, prelaunch operations, is conducted immediately prior to launch vehicle ascent.

Operations in the nuclear payload processing facility are illustrated in Figure 8.2-1. Waste payload assembly operations begin with the unloading of the shipping cask. After unloading, the individual waste payload assemblies may be placed in the storage canyon for later buildup or transferred to the assembly canyon for the beginning of waste payload assembly operations. In the assembly canyon, the first waste payload assembly is installed on the buildup fixture using a shielded overhead crane. The same crane is used for the installation of the interpayload support structure on top of the first waste payload. The second waste payload is placed on top of the interpayload support structure, and captive connecting bolts are tightened using a remote operated impact wrench to complete the assembly. The assembled waste payload system is lifted by the shielded overhead crane and installed in the flight support system which has its payload transfer cradle rotated to

ORIGINAL PAGE IS
OF POOR QUALITY

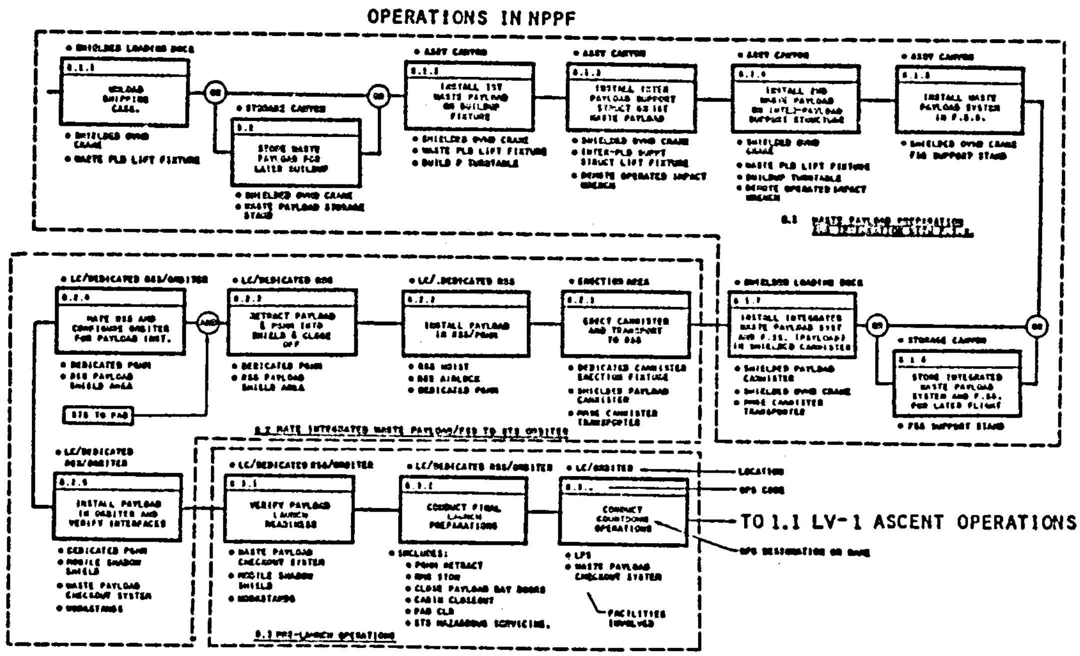


Figure 8.2-1. Launch Site Operations Flow Diagram

the vertical position. The flight support system payload transfer cradle is rotated to horizontal and latched remotely into its flight position. The integrated waste payload system and flight support system may now be either removed to the storage canyon for holding for a later flight or transferred to the shielded loading dock for installation in the shielded payload cannister.

Mating of the integrated waste payload cargo element, comprising the flight support and waste payload systems, to the uprated STS orbiter begins with transport of the shielded payload canister to the erection area by a standard MMSE canister transporter. In the erection area, the canister is tilted upright and reinstalled on the canister transporter, which takes it to the rotating service structure at the launch pad. Using the crane located in the rotating service structure, the payload is hoisted up inside it and installed in the dedicated PGHM. After installation into the payload ground handling mechanism, the waste payload is retracted inside a shielded container and a shielded door is closed to allow personnel access to the interior of the rotating service structure required for orbiter prelaunch operations. At this point, the uprated STS is transported to the pad from the VAB on the mobile launch platform. The RSS is rotated into position and mated to the orbiter and the orbiter configured for installation of the waste payload cargo element. Installation of the payload is followed by verification of payload interfaces.

Personnel access to the interior of the RSS is provided at this point by mobile shadow shields, which move into place around the installed waste payload cargo element.

Prelaunch operations begin with verification of payload launch readiness, using a dedicated waste payload checkout system. This step is followed by the final launch preparations, including closing of the payload bay doors, loading of the crew followed by cabin closeout, clearing of the launch pad, hazardous servicing of the launch system, and retraction of the rotating service structure. Following this step, countdown operations using the launch processing system and the waste payload checkout system are conducted leading to ascent and the beginning of flight operations.

8.3 FLIGHT OPERATIONS FOR NOMINAL DELIVERY MISSION

Flight operations for the reference nominal space disposal delivery mission are divided into orbiter ascent operations, low Earth orbit operations, orbit transfer system injection operations, and solar orbit insertion stage operations. Figure 8.3-1 shows the entire sequence of operations.

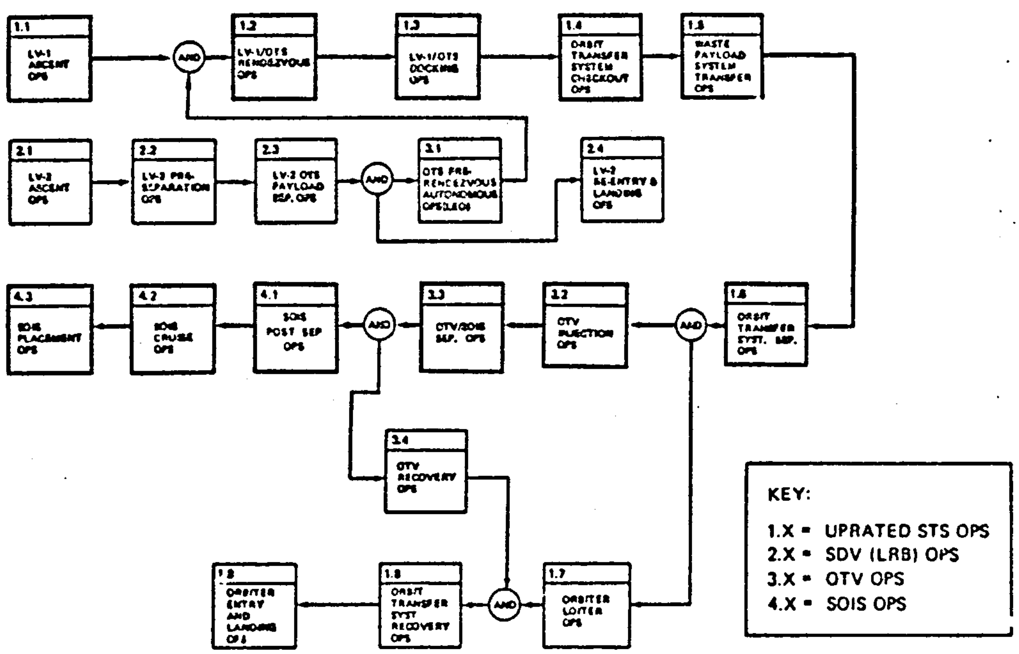


Figure 8.3-1. Nominal Waste Payload Delivery Mission Flow Diagram

8.3.1 Orbiter Ascent Operations

The sequence of operations involved in orbiter ascent is illustrated in Figure 8.3-2. Operations begin with ignition of the space shuttle main engines and the liquid rocket

ORIGINAL PAGE IS
OF POOR QUALITY

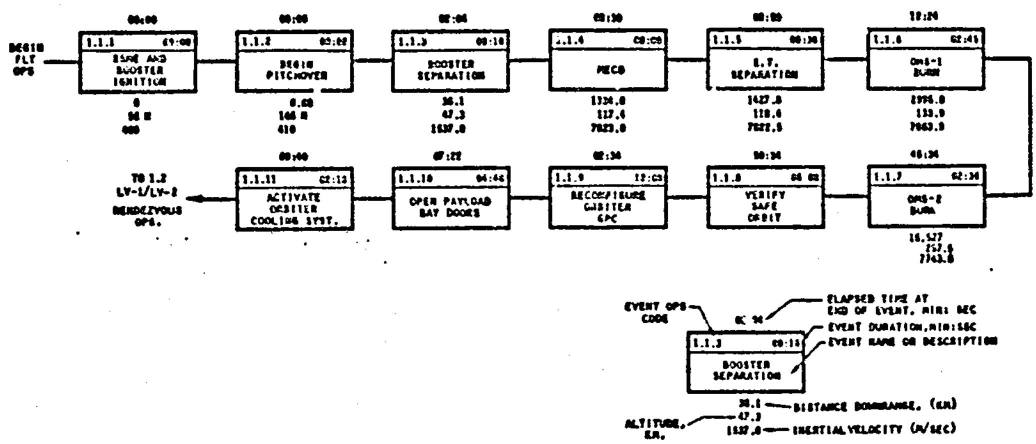


Figure 8.3-2. Orbiter Ascent Operations

boosters. Approximately 9 sec into the flight, a 2-sec pitchover maneuver is initiated, followed by booster separation at 2 min 4 sec and cutoff of the space shuttle main engines at about 6 min 38 sec. After another 30 sec, the external tank is separated; and approximately 12 min into the flight, the first burn of the orbiting maneuvering system engines inserts the vehicle into an elliptical transfer orbit with an apoapsis altitude of approximately 270 km. At 45 min 30 sec into the flight, the second orbiting maneuvering system burn circularizes the orbit at an altitude of 270 km. Five minutes are required for verification of a safe orbit, followed by a 12-min operation which reconfigures the orbiter general-purpose computers for on-orbit operations. By 67 min into the flight, the payload bay doors are opened. By 69 min 40 sec into the ascent, the orbiter cooling system has been activated and the orbiter is ready to begin low Earth orbit operations.

8.3.2 Orbiter LEO Operations

Operations performed by the orbiter in low Earth orbit include rendezvous and docking with the orbit transfer system, checkout of the orbit transfer system, waste payload system transfer operations, and separation. The entire sequence of LEO operations is illustrated in Figure 8.3-3. Each box contains the name of an operation, number assigned to the operation, and the elapsed time for the operation. The larger dotted boxes correspond to the individual events on the top-level operations flow shown in Figure 8.3-1. Each dotted box contains the elapsed time for the entire sequence of operations enclosed.

Rendezvous operations begin at the completion of ascent operations and are illustrated in Figure 8.3-4. The first operation is orientation of the orbiter and

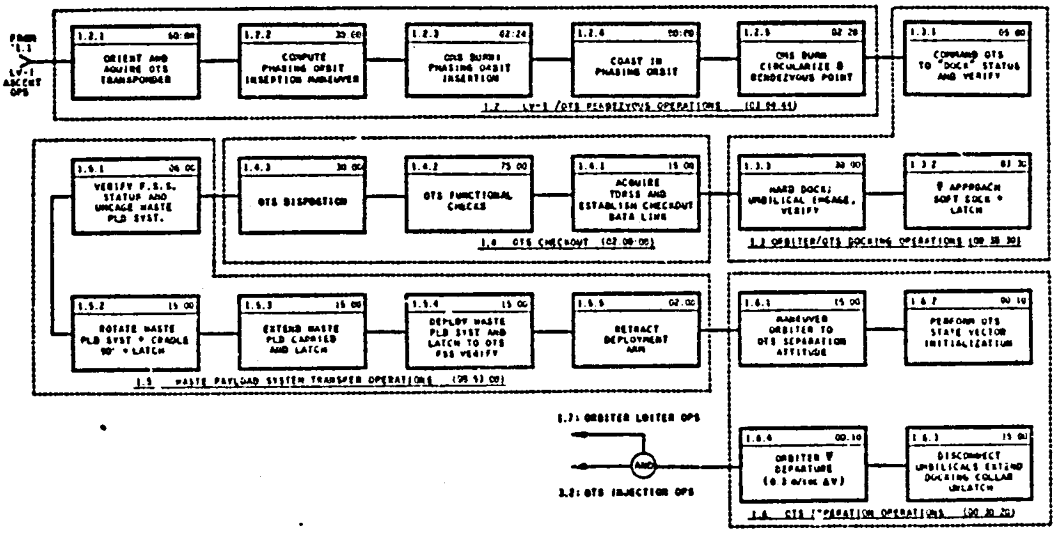


Figure 8.3-3. Orbiter Leo Operations

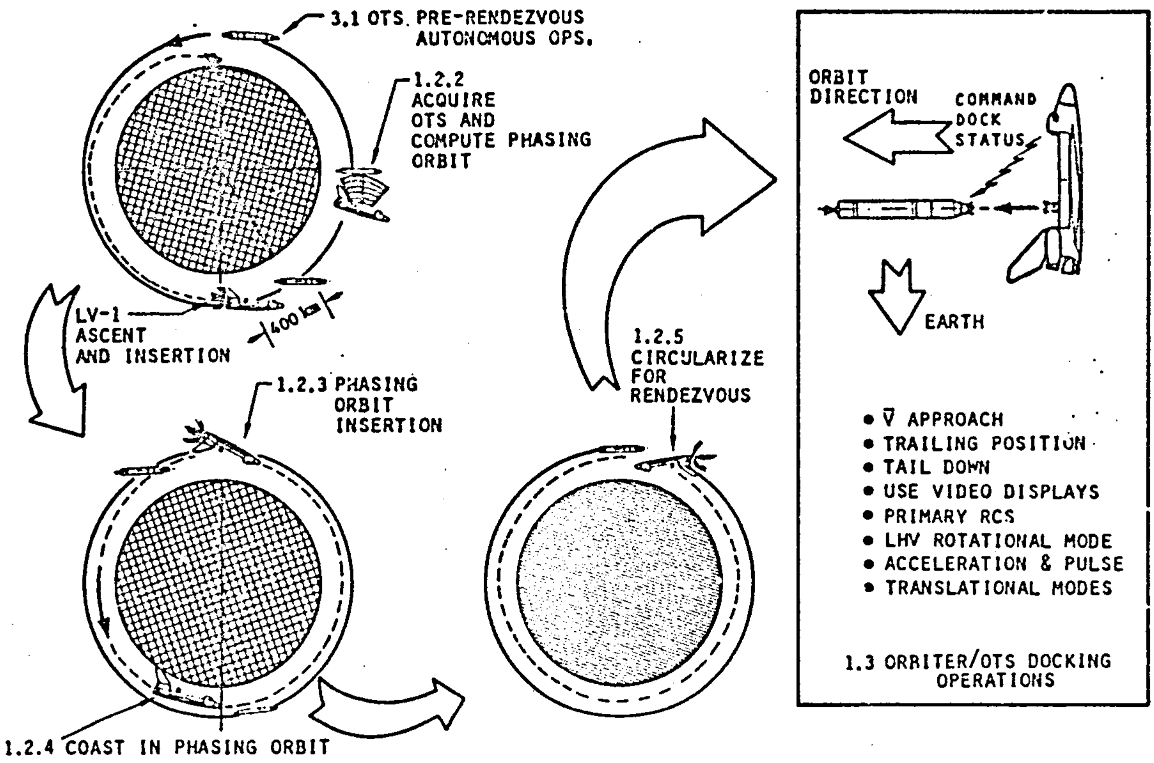


Figure 8.3-4. Leo Rendezvous and Docking Operations

acquisition of the orbit transfer system transponder, using one of the orbiter's two K-band radar/communications systems. Acquisition is followed by computation of the phasing orbit insertion maneuver using the orbiter general-purpose computer. A single orbiting maneuver system burn places the orbiter into the elliptical phasing orbit. A 90-min coast in the phasing orbit is followed by a second orbital maneuvering system burn, which places the orbiter in the same orbit as the orbit transfer system, approximately 10 km behind it

Orbiter and orbit transfer system docking operations are also illustrated in Figure 8.3-3. The first step is to command the orbit transfer system to docking status with its reaction control system shut down using a low-power command link from the orbiter. Following verification of orbit transfer system achievement of docking status, the orbiter approaches along the velocity vector in a tail-down orientation until the docking collars on the waste payload adapter and the orbiter-mounted flight support system contact and soft latch. After the initial latchup, the flight support system actuators pull the docking collars back into the hard engagement position. Electrical and mechanical umbilical connections between the orbit transfer system and the flight support system are made at this time.

Verification of hard docking and engagement of the umbilicals is followed by orbit transfer system checkout which is accomplished by means of a data link with the ground-mounted checkout facilities. Orbit transfer system functional checks are followed by disposition of the vehicle as to suitability for flight. Waste payload system transfer operations follow the verification of the orbit transfer system for flight.

Transfer operations are illustrated in Figure 8.3-5. The initial step is to rotate the waste payload system 90 deg within the flight support system using the transfer cradle. After latching in this position, the waste payload carrier is extended to translate the waste payload system from the flight support system into the waste payload adaptor of the orbit transfer system. After latching to the waste payload adaptor, the payload carrier is retracted back into the flight support system leaving the waste payload firmly mounted to the orbit transfer system.

Waste payload transfer is followed by separation from the orbit transfer system. After maneuvering the orbiter to the proper attitude for orbit transfer system separation, the orbiter general-purpose computer is used to update the injection stage guidance system. This is followed by disconnection of the orbit transfer system umbilicals, extension of the docking collar, and unlatching of the waste payload adapter docking ring. The orbiter then backs away from the orbit transfer system along the velocity vector at about 0.3 m/sec. At this point, the orbit transfer system begins the sequence of injection operations.

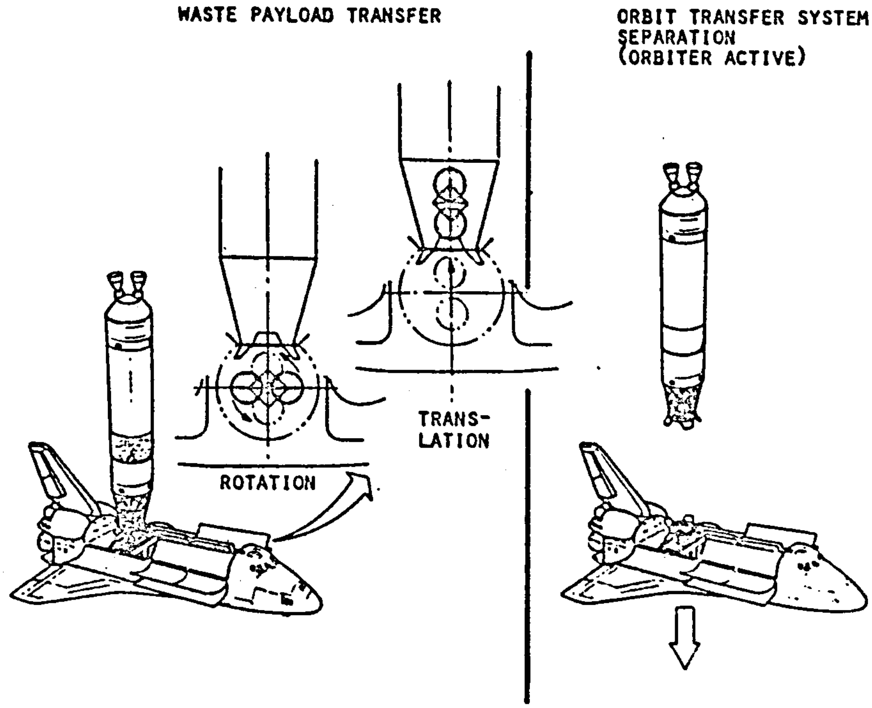


Figure 8.3-5. Waste Payload Transfer Operation Detail

8.3.3 Orbit Transfer System Injection Operations

The sequence of operations involved in injection of the solar orbit insertion stage and waste payload into heliocentric transfer orbit is illustrated in Figure 8.3-6. Injection

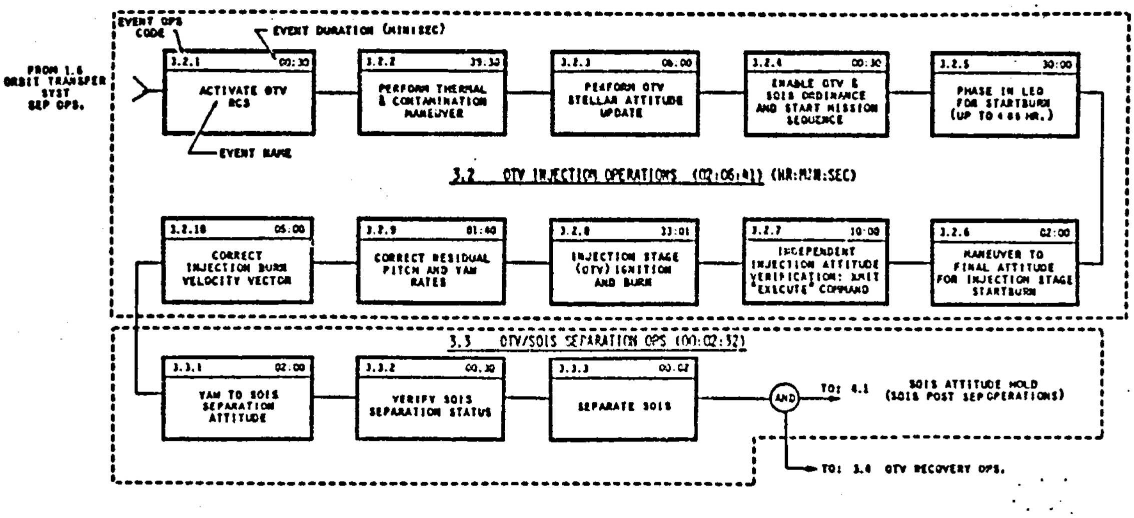


Figure 8.3-6. Nominal Delivery Mission OTV Injection Operations

operations begin 30 sec after separation from the orbiter with activation of the injection stage reaction control system. A thermal and contamination maneuver is followed by an OTV stellar attitude update to initialize the vehicle inertial platform. OTV ordnance is enabled and the mission sequence is started 30 min after separation from the orbiter. A nominal 30-min wait in low Earth orbit phases the vehicle for injection. At the conclusion of the phasing period, the orbit transfer system is maneuvered to the final attitude for injection. This maneuver is followed by independent attitude verification conducted from the orbiter or the ground. Verification of correct injection attitude is followed by transmission of an execute command. Without transmission of the execute command, the injection sequence would be halted at this point. If the execute command is transmitted, the injection stage ignites at the scheduled time for the 33-min injection burn. Following main engine cutoff, the reaction control system is used to correct residual pitch and yaw rates. An axial trim burn is then conducted by the reaction control system to correct injection burn velocity vector errors.

OTV solar orbit insertion stage separation operations begin as the orbit transfer system yaws to the correct attitude for solar orbit insertion stage separation. A final check is run by the OTV to verify that the solar orbit insertion stage is in the correct status for separation. This operation is followed by SOIS separation, which is accomplished by firing the pyrotechnic separation nuts which secure the solar orbit insertion stage to the interstage. Preloaded separation springs provide necessary separation impulse. Following separation, the solar orbit insertion stage holds attitude in the beginning of its own autonomous operations and the OTV begins the sequence of operations which will lead to its recovery for reuse by the orbiter in low Earth orbit.

8.3.4 SOIS Operations

Solar orbit insertion stage operations are illustrated in Figure 8.3-7. Operations begin with the post-separation phase. The stage holds attitude for 1 min and then yaws to allow its Sun sensors to acquire the Sun, which serves as the primary attitude reference during cruise. Following Sun acquisition, the pitch and yaw limit cycle which will govern the vehicle attitude during the cruise phase is initiated. The first 12 hr of the cruise are used to verify by telemetry to the ground that the pitch and yaw limit cycle is being executed within the limits required to ensure that propellant consumption limits are not exceeded.

With post-separation operations completed, the SOIS enters the cruise phase of its mission, which lasts for 3899 hr. During this period, the vehicle keeps its x-axis pointed

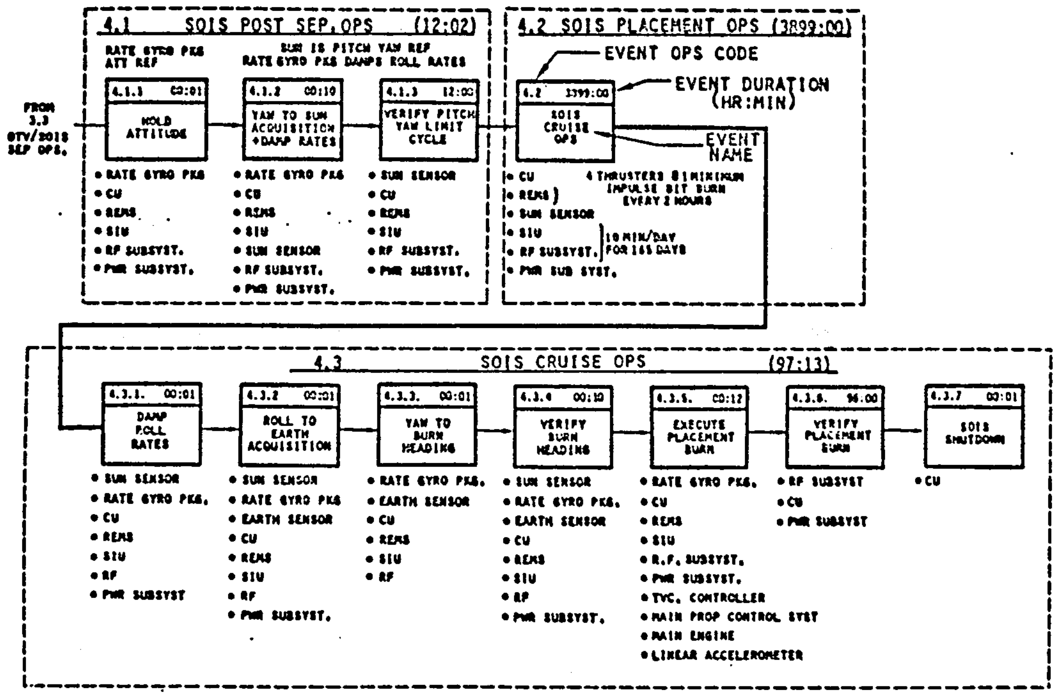


Figure 8.3-7. Nominal Delivery Mission SOIS Operations

at the Sun, using limit cycle attitude stabilization about the y- and z-axes with no stabilization about the x-axis.

The end-of-cruise operation is signalled by time-out of the SOIS cruise timer. The initial step in placement operations is damping of roll rates which is accomplished using a rate gyro package as the primary roll reference. Damping of roll rates is followed by vehicle roll about the x-axis until its Earth sensor acquires the Earth. Following Earth acquisition, the vehicle yaws to the correct burn heading for the placement burn, which is followed by a 10-min period for verification from Earth that the burn heading is correct. Telemetered Earth and Sun sensor readings allow remote verification of the correct burn heading. If the heading is correct, no signal is sent to the SOIS and the vehicle continues in the placement sequence. If an anomaly in the position is discovered, an override signal from the ground control center allows control of the vehicle to be taken over directly by the ground.

Following verification of the burn heading, the 12-min placement burn is executed by the main engine, placing the vehicle in the destination heliocentric orbit. The vehicle beacon transponders are left on for 96 hr after the placement maneuver to allow ground verification of the placement burn. Placement burn verification is followed by permanent shutdown of the SOIS, leaving the vehicle and the waste payload in the destination heliocentric orbit.

8.4 DEEP-SPACE RESCUE OPERATION SEQUENCE

The sequence of operations involved in a nominal deep-space rescue mission is illustrated in Figure 8.4-1. The sequence begins with the insertion of the rescue system and the orbit transfer system into low Earth orbit by the same launch system used in the

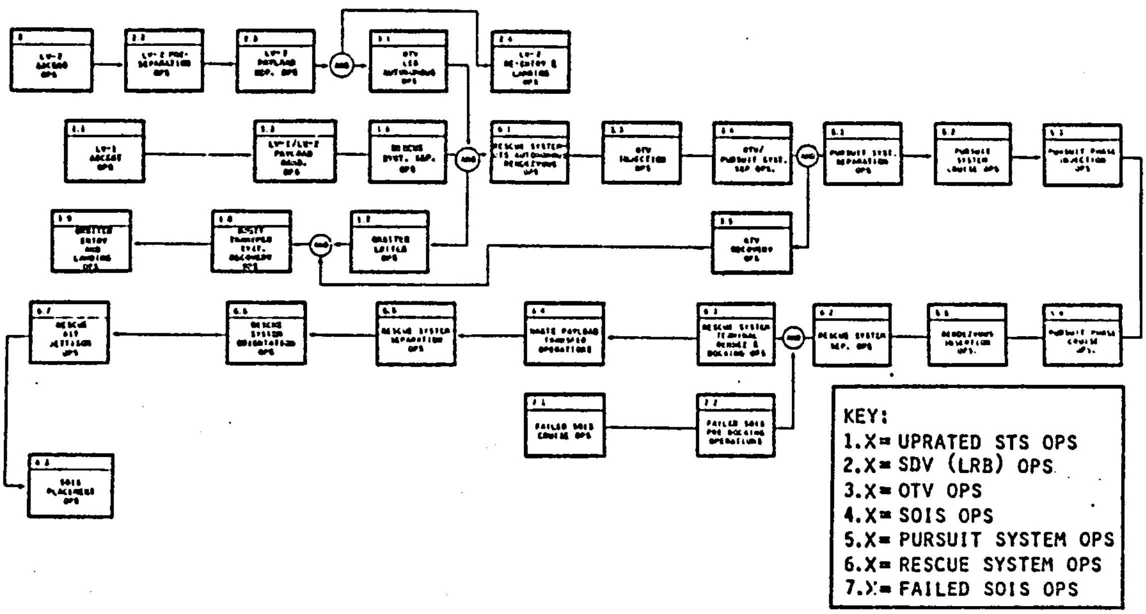


Figure 8.4-1. Nominal Deep-Space Rescue Mission Operations Flow Diagram

nominal delivery mission. Following an autonomous rendezvous of the rescue system with the orbit transfer system in low Earth orbit, the rescue system pursuit configuration is injected into the rescue transfer orbit by the injection stage, which returns to low Earth orbit for eventual reuse. Following separation from the injection stage, the rescue system, now in the pursuit configuration, coasts in the rescue transfer orbit to perihelion where the pursuit phase injection operations leave it in the pursuit phase cruise orbit. At the termination of pursuit phase, the rescue system is inserted into the rendezvous orbit and rendezvouses and docks with the failed SOIS. Following waste payload transfer from the failed stage to the rescue system, the rescue system separates and orients itself for the final placement burn. After orientation and initiation of SOIS autonomous operations, the rescue kit is jettisoned and the rescue SOIS executes a normal placement burn, leaving itself and the waste payload in the destination heliocentric orbit.

9.0 CONCLUSIONS

This section summarizes some of the conclusions reached as a result of this study.

1. Waste form parameters for the reference cermet waste form are available only by analogy. Detail design of the waste payload would require determination of actual waste form properties.
2. The billet configuration constraints for the cermet waste form limit the packing efficiency to slightly under 75% net volume. The effect of this packing inefficiency in reducing the net waste form per waste payload can be seen in Figure 4.1-5. The cermet waste form mass per unit mass of waste payload is lower than that of the iodine waste form even though the cermet has a higher density (6.5 versus 5.5). This is because the lead iodide is cast achieving almost 100% efficiency in packing. This inefficiency in the packing of the cermet results in a 20% increase in number of flights which increases both cost and risk.
3. Alternative systems for waste mixes requiring low flight rates (technetium 99, iodine 129) can make effective use of the existing 65K STS in either single- or dual-launch scenarios.
4. A comprehensive trade study would be required to select the optimum orbit transfer system for low-launch-rate systems. This study was not conducted as part of the present effort due to selection of the cermet waste form as the reference for the study. Several candidates look attractive for both single- and dual-launch systems (see sec. 4.4), but due to the relatively small number of missions, a comprehensive comparison of life cycle costs including DDT&E would be required to select the best system.
5. The reference system described in sections 5.0, 6.0, 7.0, and 8.0 offers the best combination of cost, risk, and alignment with ongoing NASA technology development efforts for disposal of the reference cermet waste form.

10.0 RECOMMENDATIONS

The reference space system selected for this study is virtually identical to system DL-2 described in the 1980 MSFC study. Accordingly, the recommendations from this study are not specific to this effort and should be considered an amplification of those from the 1980 study. Because of the very preliminary level of definition of the space system, the following recommendations address generic issues and are not specific to the reference system described in sections 7.0 and 8.0.

1. Further 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 waste payload accident-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.
2. Because of the influence of waste form packing efficiency on the mass of the waste payload, research should be directed at relaxing the fabrication constraints on the cermet waste form in the interest of achieving better packing efficiency. Up to a 20% to 25% reduction in the total number of missions for disposal of a given mass of cermet could be achieved.
3. A preliminary study of the contingency rescue mission in more detail than reported in past studies is required to identify concepts and define areas more specifically for further study. The goals of this effort should be to:

    a. Establish the quantitative risk benefits of maintaining the contingency rescue capability, as opposed to maintaining the nominal rescue mission capability only.
    b. Establish the fundamental technical viability of contingency rescue in deep space.
    c. Estimate cost for implementation (particularly DDT&E costs).

    These tasks will determine whether contingency rescue is an enabling capability for space disposal and, if it is, will provide the basis for decisions on the level of emphasis to be applied.

4. While the 0.85-AU heliocentric orbit destination was selected as a reference for this study, further analysis should be conducted of space disposal destinations in the geolunar system. If the stability of such destinations could be verified to the same level as that of the reference 0.85-AU destination, substantial cost and risk benefits could be realized, including the following.

   a. Geolunar destinations could be important if further studies of the contingency rescue mission find it infeasible or impractically expensive due to acquisition, tracking, or rendezvous/docking problems. Rescue can be accomplished witnin weeks from any location in the geolunar system, and the restricted ranges involved make passive tracking of failed vehicles possible.
   b. By selecting a geolunar destination, the possibility of high-energy reentries in the far future due to loss of a payload in deep space could be eliminated. This would minimize the long-term risk of loss of containment posed by high-energy reentry.
   c. Use of geolunar system destinations would allow elimination of the placement stage with complete reuse of the injection stage, which could be an unmodified version of the OTV planned by NASA for operation in the 1990's. The resulting reduction in DDT&E and production costs should be evaluated.

Efforts should be aimed at defining the best geolunar destination and validating its stability to the same level as the reference 0.85-AU heliocentric orbit destination. Validation would allow realization of the cost and risk benefits of the geolunar destination.

APPENDIX A

ACRONYMS

ACRONYMS

ABOTV	aerobraked OTV
ACS	attitude control system
AOA	abort once around
ASE	airborne support equipment
ATO	abort to orbit
AU	astronomical unit
BAC	Boeing Aerospace Company
BCL	Battelle Columbus Laboratories
CU	command unit
DDT&E	design, development, test, and evaluation
DOE	Department of Energy
DSN	Deep-Space Network
EB	electron beam
ERP	effective isotropic radiated power
ET	external tank
FOSR	flexible optical solar reflector
FOV	field of view
FSS	flight support system
GLOW	gross liftoff weight
GPS	global positioning system
HLW	high-level waste
IPSS	interpayload support structure
IUS	inertial upper stage
KSC	Kennedy Space Center
LCC	life cycle cost
LEO	low Earth orbit
LLOTV	long-life OTV
LRB	liquid rocket booster
MECO	main engine cutoff
MLI	multilayer insulation
MLP	mobile launch platform
MMSE	multimission support equipment
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration

NPPF	nuclear payload processing facility
OFS	orbiter flotation system
OMS	orbital maneuvering system
ONI	Office of Nuclear Waste Terminal Storage Integration
ONWI	Office of Nuclear Waste Isolation
OPF	orbiter processing facility
ORNL	Oak Ridge National Laboratory
OTS	orbital transfer system
OTV	orbital transfer vehicle
P/A	propulsion/avionics
$Pb\,I_2^{129}$	iodine 129
PGHM	payload ground-handling mechanism
PSMC	Payload and Sequential Mass Calculation (code)
RCS	reaction control system
REM	reaction engine module
RF	radiofrequency
RSI	reusable surface insulation
RSS	rotating service structure
RTLS	return to launch site
SDCLV	shuttle-derived cargo launch vehicle
SDV	shuttle-derived vehicle
SES	solar electric stage
SIU	signal interface unit
SOIS	solar orbit insertion stage
SRM	solid rocket motor
SSME	space shuttle main engine
STS	space transportation system
Tc^{99}	technetium 99
TSM	tail service mast
TVC	thrust vector control
ULOW	upperstage liftoff weight
VAB	vertical assembly building

APPENDIX B

REFERENCES

REFERENCES

1. Reinert, R. P., "Analysis of Space Systems Study for the Space Disposal of Nuclear Waste," Vol. II, Technical Report, Boeing Aerospace Company, Contract NAS8-33847, 1981.

2. Rice, E. E. et al, "System Safety Guidelines Document for Nuclear Waste Disposal in Space," Third Draft, Battelle Columbus Laboratories, Contract NAS8-34512, December 7, 1981.

3. Diramio, H. A., "Shuttle-Derivative Vehicles Study—Operations, Systems, and Facilities," Vol. II, Final Report, Boeing Aerospace Company, Contract NAS8-32395, December 1977.

4. Anon., "Orbital Transfer Vehicle Concept Definition Study," Vol. 4, Final Report, Second Concept Definition, Boeing Aerospace Company, Contract NAS8-33532, 1980.

APPENDIX C

WASTE PAYLOAD THERMAL ANALYSIS

COORDINATION SHEET

D180-26777-2

TO	R. A. Reinert 8F-74	NO.	2-3632-0000-113
		ITEM NO.	
CC	C. L. Wilkinson 8C-05	DATE	September 23, 1981
		MODEL	

GROUP INDEX Propulsion/Mechanical/Fluid Systems-Flight Technology

SUBJECT Space Disposal of Nuclear Waste/Containment Based on Waste Form Melting

Waste form containment within the integrally shielded core, which is described in attachment 1, requires that the waste melting temperature of 1200°C not be exceeded.

Thermal modeling and analysis of the disposal vessel was performed for a 0.85 astronomical unit orbit, see attachment 2. An upper limit for the waste temperature was determined to be about 700°C, well below the 1200°C melting point.

For the shielded core thermal configuration which was addressed, it is concluded that containment is not jeopardized due to melting of the waste form.

Additional analysis showed that a waste heat generation rate of 0.022 W/cm^3 would be required to attain the 1200°C melting temperature. See Figure 3.1 of Attachment 3.

Prepared by: Byron Siegel
B. L. Siegel
2-3632 773-0362 8C-05

Approved by: T. J. Kramer
T. J. Kramer
2-3632 773-2973 8C-05

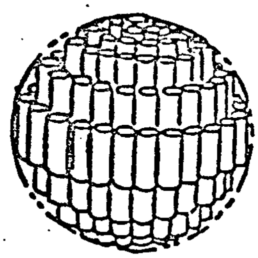
Attachments

D6 4000 1098 REV. 5/78

Attachment 1 - Waste Core/Core Shield Thermal Geometry

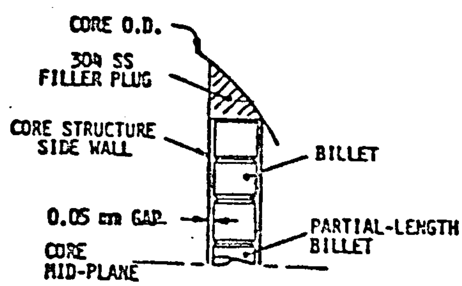
The waste core structure consists of a 304 SS sphere with 59 mm O.D. holes bored in a hexagonal close-packed pattern, with a 60 mm center to center spacing.

Figure 1.1. Waste Core Drilling Pattern



The waste form (CERMET) is fabricated into cylindrical "billets", 29.45 mm radius by 58.9 mm length, which are stacked end to end within the core structure void. Ths axial packing of the billets is enhanced in by inserting a partial-length billet so as to bring the billets flush with the surface of the core structure.

Figure 1.2. Billet Arrangement in the Core



A 0.05 mm void is assumed to separate the billets' circumference from the core side walls, and filler plugs are used to maintain billet orientation/decrease the thermal resistance to heat transfer from the core. The CERMET thermal conductivity was taken to be 9.5 W/M-K, and its volumetric heat generation rate equaled 0.0065 W/cubic cm.

The waste vessel's core is protected against physical damage by a 22 mm thick spherical shield made from 300 Series Maraging Steel. The concentric shield is separated from the core by a 1 mm gap. The thermal conductivities for the Maraging Steel and that for the 304 Stainless Steel were both conservatively taken to be equal to 10 W/M-K.

A 5 cm thick graphite radiation shield is bolted to the Maraging shield; with a contact conductance of 300 W/sq M-K. A thermal conductivity of 75 W/M-K for the graphite shield was also used.

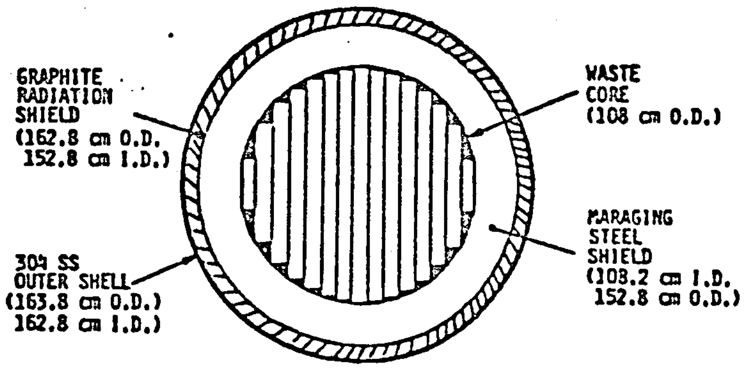
The graphite shield is protected against damage by a 0.5 cm thick shell made from 304 SS. The protective shell is attached to the outside surface of the graphite shield by a room temperature vulcanizing resin to ensure good thermal contact.

All of the waste vessel's radiative surfaces: the outer stainless-steel shell, the inside surface of the Maraging steel shield, the outside surface of the stainless-steel waste core, and the individual billets are assumed to be flamed sprayed with aluminum oxide; the oxide's radiative properties are defined by

$$\alpha_{solar}/\epsilon_{infrared} = 0.26/0.80$$

A sketch of the waste vessel configuration is given by Figure 1.3.

Figure 1.3. Cross-section of Nuclear Waste Disposal Vessel (waste billets not shown)



Attachment 2 - Thermal Modeling and Analysis

A heat balance on the waste vessel is initially performed so that the vessel's average outside surface temperature may be found. The waste heat generation rate within the core (Q_i) is evaluated by

$$Q_i = (\text{volumetric generation rate})\ (\text{total billet volume})$$
$$= 0.0065 \frac{W}{cm^3} \quad 526{,}000\ cm^3 = 3400\ W \qquad (2.1)$$

The solar heating rate (Q_s) at a 0.85 astronomical unit (AU) orbit is calculated by Equation (2.2).

$$Q_s = (\text{absorptance}) \left(\frac{\text{solar flux @ 1 A.U.}}{\text{orbital distance}^2}\right) (\text{vessel projected area})$$
$$= 0.26 \quad \frac{1400\ W/M^2}{(0.85 AU)^2} \quad \frac{\pi}{4}(1.64 m)^2 = 1010\ W \qquad (2.2)$$

The vessel surface temperature (T_0) is determined by equating the total surface heating rate to the rate at which the surface rejects heat (Q_r).

$$Q_i + Q_s = Q_r = \varepsilon\ \sigma\ T_o^4\ \pi D_o^2$$
$$4410\ W = 0.8 \quad 5.6693 \times 10^{-8} \frac{W}{M^2\text{-}K^4} \quad T_o^4 \quad \pi (1.64 m)^2 \qquad (2.3)$$

This yields an average exterior surface temperature for the vessel of 328 K.

The temperature difference between the exterior surface and the outside surface of the Maraging shield is found by first evaluating the radial thermal resistance to heat flow through: the bolted Maraging/Graphite shields interface, the graphite shield, and the outer Stainless Steel shield.

$$R_0 = \left(\frac{1}{hA}\right|_{contact} + \frac{1}{2\pi K_{Gr}}\left(\frac{1}{I.D.} - \frac{1}{O.D.}\right|_{\substack{Gr\\shield}} + \frac{1}{2\pi K_{SS}}\left(\frac{1}{I.D.} - \frac{1}{O.D.}\right|_{\substack{S.S.\\shell}}$$
$$= \frac{1}{300 \quad 2.33} + \frac{1}{2\pi\ 75}\left(\frac{1}{1.53} - \frac{1}{1.62}\right) + \frac{1}{2\pi\ 15}\left(\frac{1}{1.63} - \frac{1}{1.64}\right) \qquad (2.4)$$
$$= 0.000592 \frac{K}{W}$$

The temperature at the outer surface of the Maraging shield ($T_{m,o}$) may now be calculated.

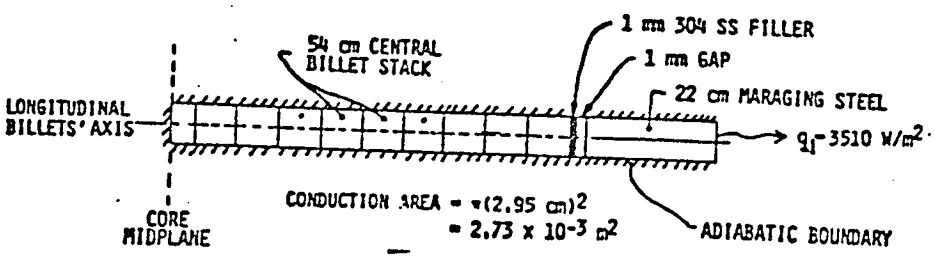
$$T_{m,o} = T_o + Q_i R_o$$

$$= 328 + (3400)(0.000592) = 330\ K \qquad (2.5)$$

Notice that to a good approximation that the temperature at the outer surface of the Maraging shield may be taken to be equal to the temperature at the vessel's exterior surface.

Thermal modeling of the waste core and of the Maraging shield is performed, to enable simplified methods for determination of an upper (waste temperature) limit. Due to the symmetry of heat generation in the waste core, the maximum waste temperature will occur at the mid-length point of the centrally located billet tube (see Figure 1.3). In order that an upper bound for the maximum waste temperature be found, the temperature gradient along the central billets' axis is maximized. From Fourier's Law of heat conduction, the temperature gradient along the billets' axis is given to be proportional to the heat transfer rate along the same axis. An upper found for the waste temperature is therefore formulated by requiring an adiabatic boundary at the billets' circumference. This model is shown in Figure 2.1.

Figure 2.1. Cross-Section Depicting the Thermal Model for the Waste Core and the Maraging Shield.



Note that due to a contract conductance at the interface between individual billets, the temperature drop along the longitudinal billets' axis is greater than that for a continuous waste cylinder of the same active length. Also shown in Figure 2.1 is the heat transfer path from the outermost billet to the outer surface of the Maraging Steel shield. The thermal resistance of this linear path is larger than that for a radial path; thermal resistance is inversely proportional to the heat flow area. It follows that the temperature rise from the outside of the Maraging steel to the outermost billet as calculated by the linear path model will be an upper bound.

By using the thermal model that has been described, an upper bound for the maximum temperature at the inside surface of the Maraging shield ($T_{m,i}$) is determined by

$$T_{m,i} = T_{m,o} + q_i \left(\frac{L}{K}\right|_{maraging}$$

$$= 330 + 3510\left(\frac{0.22}{10}\right) = 407\ K \tag{2.6}$$

where q_i is the axial heat flux from the billet stack. The waste core's outside surface temperature ($T_{c,o}$) is evaluated by Equation (2.7).

$$T_{c,o}^{4} = \frac{q_i}{\sigma}\left(\frac{1}{\varepsilon_{Al_2O_3}} + \frac{1}{\varepsilon_{Al_2O_3}} - 1\right) + T_{m,i}^{4}$$

$$= \frac{3510}{5.6693\times10^{-8}}\left(\frac{2}{0.8} - 1\right) + 407^{4} \tag{2.7}$$

The core surface temperature is calculated to be about 539 K. The thermal resistance of the 1 mm thick layer of 304 Stainless Steel is very small, and therefore the layer may be taken to be isothermal. The temperature difference at the billets' interface is determined by neglecting the conduction heat transfer mode.

$$(T_n^{+})^{4} = \frac{q_{i,n}}{\sigma}\left(\frac{1}{\varepsilon_{Al_2O_3}} + \frac{1}{\varepsilon_{Al_2O_3}} - 1\right) + (T_n^{-})^{4}$$

$$= q_{i,n} \times 2.646\times10^{7} + (T_n^{-})^{4} \tag{2.8}$$

where, T_n^+/T_n^- : the upper/lower temperature at the n^{th} billet interface (see Figure 2.2), and

$q_{i,n}$: the heat flux passing through the n^{th} interface (see Figure 2.2).

The temperature difference between the opposing ends of a billet with an adiabatic circumferential surface is determined from Equation (2.9).

$$(T_n^-) = (T_{n+1}^+) + \frac{q'''}{2K_{billet}}\left(x_{n+1}^2 - x_n^2\right) \qquad (2.9)$$

where,

T_n^-: the lower temperature at the n^{th} billet interface (see Figure 2.2),

T_{n+1}^+: the higher temperature at the n^{th} + 1 billet interface (see Figure 2.2),

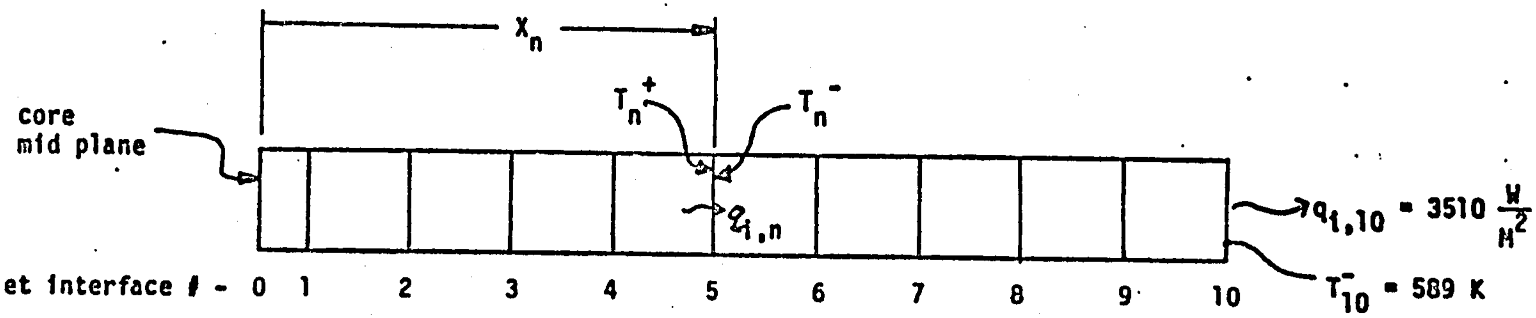
x_{n+1}/x_n: the axial distance from the core midplane to the $n^{th}/n^{th}+1$ billet interface, and

q''': the billet volumetric heat generation rate.

With both the core surface temperature ($T_{10}^- = 589K$) and the billet-stack boundary heat flux ($q_{10} = 3510\ W/m^2$) known, the upper temperature at the 10^{th} billet interface (T_{10}^+) is evaluated from Equation (2.8) to equal 680 K. Utilizing Equation (2.9) with $x_{10} = 54$ cm and $x_9 = 48.1$ cm, the lower temperature at the 9^{th} interface (T_9^-) is calculated to be 701 K. Using this bootstrap technique, as given by Equations (2.8) and (2.9), the waste temperature at the core midplane is found to equal 951 K, or 678°C.

For the given waste core and shielding thermal geometry, an upper bound on the maximum waste temperature is 700°C.

Figure 2.2. Solution for the Billets' Axial Temperature Gradient



n: billet interface # -	0	1	2	3	4	5	6	7	8	9	10
X_n (cm) -	0	0.9	6.8	12.7	18.6	24.5	30.4	36.3	42.2	48.1	54.0
$q_{i,n}$ ($\frac{W}{M^2}$) -	0	59	442	826	1209	1593	1976	2360	2743	3127	3510
T_n^- (K)	951	951	946	935	919	896	865	826	773	701	589
T_n^+ (K)		951	949	942	929	910	885	852	810	755	680

maximum waste temperature = 951 K = 678°C

C10

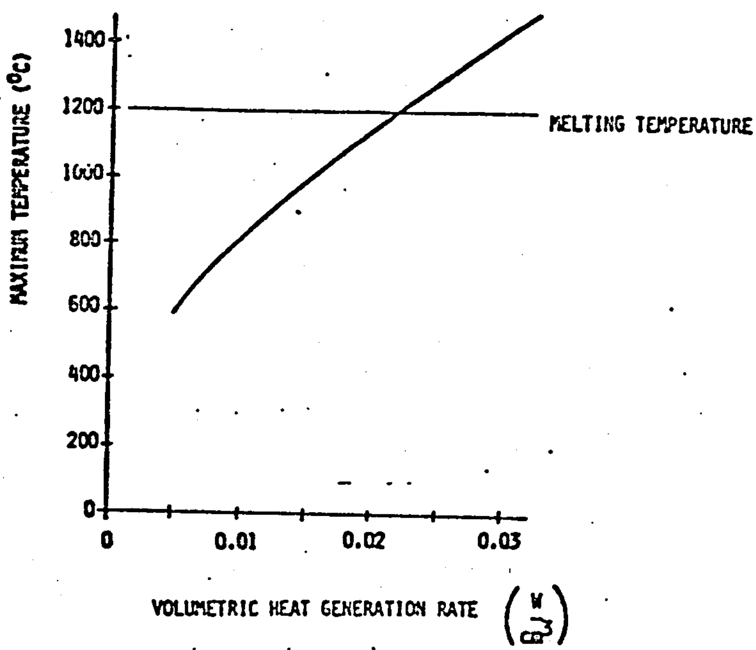
D180-26777-2

Attachment 3 - Cermet Melting as a Function of the Volumetric Heat Generation Rate

The thermal model presented in Attachment 2 is parametrically analyzed by varying the waste volumetric heat generation rate. The analysis technique used parallels that given by Equations (2.1) to (2.9)

Figure 3.1 depicts the results that were obtained. It was found that a heat generation rate of 0.022 W/cm^3 would cause melting at the center of the waste core.

Figure 3.1, Waste Containment Based on Varying Thermal Loads



APPENDIX D

SOIS RELIABILITY ANALYSIS

November 6, 1981
2-3810-0001-173

To: R. P. Reinert

cc: R. C. Schneider
M. R. Stanley

Subject: Preliminary SOLAR Orbit Insertion Stage (SOIS) Reliability Analysis

A two part analysis of the SOIS design & mission indicates the baseline SOIS system has a preliminary reliability prediction of .901 for its complete mission profile and that a reliability of .995 is feasible without excessive cost and mass penalties.

The baseline configuration analysis (Attachment 1) is based upon the provided system configuration and the defined mission profile from OTV release to system shutdown. This design incorporates some equipment redundancy for reliability and safety purposes. It has a mission reliability prediction of .901, with the major failure probability occuring during the dormant cruise phase. The failure potential is nearly equally divided among the four major subsystems (see Table 1.1).

In addition to the baseline system analysis the SOIS design was optimized for reliability from its basic functional design. This analysis indicates the SOIS mission reliability can be increased from a minimum equipment design value of .745 to .995 with approximately a 40% increase in mass. The reliability of the optimized SOIS design is limited to the .995 region by certain equipments where redundancy is not practical.

For complete resolution of the optimized design a detailed analysis would be necessary. Several assumptions are used in the optimization process which make both the exact mass penality and reliability figures dependent upon more specific analyses. However, the optimization does provide good visibility to the design potential and optimum configuration.

R. C. Hall
Roger C. Hall

Michael L. Janssen
Michael L. Janssen

ATTACHMENT I

This attachment consists of a series of charts which outline the reliability analysis of the baseline SOIS configuration and mission. A negative exponential reliability model is assumed and the future OTV technology study (CR-) is the primary source of configuration data.

The study results are summarized in Table 1. Both system and subsystem reliabilities are shown by mission phase in addition to mission totals. Most of the baseline subsystem designs contain redundancy in critical areas and no subsystem appears to have a disproportionate failure potential.

By mission phase its obvious that the cruise phase has the highest failure probability. This is due entirely to its long duration (180 days) and not the result of any design deficiency.

Figures 1 through 4 detail the SOIS subsystem analysis. These include the reliability block diagrams, mission success criteria, component failure rates, and basic reliability estimates by equipment and mission phase.

Figure 5 shows the appropriate duty cycle for each equipment; i.e., its on-off history by mission phase. In addition, it defines the phase durations and some mission phase groupings made for analysis purposes. These groupings combined phases which had similar equipment duty cycles and were made to simplify the analyses.

The power subsystem analysis, Figure 1, shows that the single battery and heater represent the majority of its failure potential and would be candidates for design refinements.

The baseline avionics subsystem, Figure 2, shows the CPU string to have the highest failure potential. This string is already redundant and no other reliability enhancements are apparent in the remainder of the avionics equipment.

The propulsion subsystem, Figure 3, has three major contributors to its failure potential. The hydrogen and oxygen systems both have a single thermal vent valve to relieve pressure build up as necessary. These valves are each single failure points. The main engine is also a single point which has a significant failure potential but redundancy is not anticipated for this equipment.

The main failure potential in the RCS subsystem, Figure 4, is in the hydrozine regulator valves and manifold system. Sufficient redundancy exists in the REM systems to provide a high reliability prediction for these elements.

ATTACHMENT II

An effective way to achieve high reliability is through the use of redundant components. Rather than introduce redundant components into a design in a random fashion, Boeing employs the Single Thread Reliability Optimization Program (STROP) computer model. Starting with a single thread (non-redundant) system, STROP sequences redundant additions to the system in accordance with the maximum incremental improvement in system reliability with respect to the minimum incremental increase in system weight ($\Delta R/\Delta W$).

The results of this analysis disclose both the weight required to meet a given reliability goal and the maximum reliability which can be achieved within a given weight constraint. It is a powerful tool which has been employed in the design of many Boeing space and weapons systems. STROP also optimizes reliability with respect to cost. The results of the optimization provide a guide to program management in the achievement of required reliability.

This program (STROP) was applied to the basic SOIS design to assess its reliability potential and associated system weight increase. To implement this process, the baseline SOIS design shown in attachment 1 was reduced to its minimum equipment functional design, Figure 2.1. The STROP program then adds redundant additions of a specific type, either standby or parallel, for the SOIS design. These redundant additions are shown in Table 2.1 with their corresponding system weights and reliabilities. The first addition is the helium regulator valve, the second addition includes the power distribution unit, command unit and signal interface unit as a single block, and so on. A plot of the system reliability vs the attendant system weight is shown in Figure 2.2.

Figure 2.2 shows the reliability potential of the SOIS to have an upper limit near .995. This limit is associated with certain equipment e.g., the main engine, where redundancy for reliability purposes is not feasible. The system reliability potential is in turn limited by the reliability of these equipments.

The results show significant reliability improvements for the first 20 to 24 equipment additions requiring approximately a 40% increase in weight. The potential reliability increase is nearly exhausted at this point and the marginal reliability increases beyond these additions appears to be negligible.

In interpreting these results, it should be noted that they represent the potential of the SOIS design to perform its mission. Several design assumptions and simplifications were used in the STROP analysis which make an exact reliability assessment dependent upon more detailed analyses similar to those in Attachment 1. A reliability vs cost optimization could also be performed. However, the reliability vs weight optimization analysis herein does provide a clear representation of the SOIS reliability potential to program design and management.

PHASES - TABLE I.1

	HOLD & SUN ACQ.	VERIFY	CRUISE	ATT. CONTROL	BURN	VER	TOTAL
POWER SUBSYSTEM	.999999	.999996	.986212	.999995	~1.0	.999991	.986156
AVIONICS SUBSYSTEM	~1.0	~1.0	.984706	.999954	.999994	.999482	.984635
PROPULSION SUBSYSTEM	.999998	.999900	.966301	.999990	.996902	—	.963198
RCS SUBSYSTEM	.999997	.999875	.963195	.999966	.999999	.999964	.963000
STRUCTURE	—	—	—	—	—	—	.999990
SOIS SYSTEM	.999994	.999771	.903865	.999905	.996895	.999937	.90065[illegible]

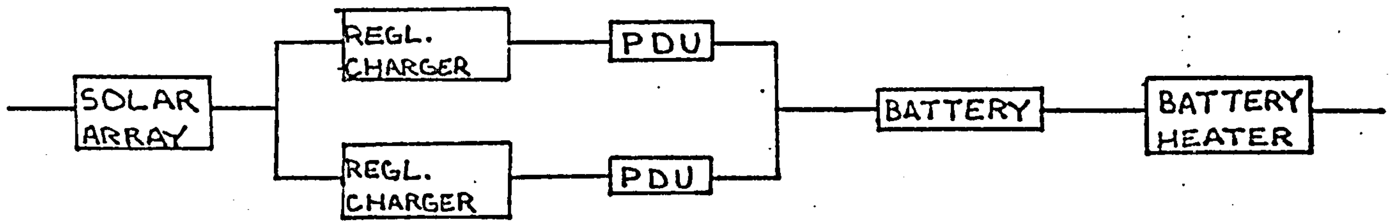
D7

D180-26777-2

ORIGINAL PAGE IS OF POOR QUALITY

POWER SUBSYSTEM - BLOCK DIAGRAM AND ANALYSIS

(FIG. 1.1)



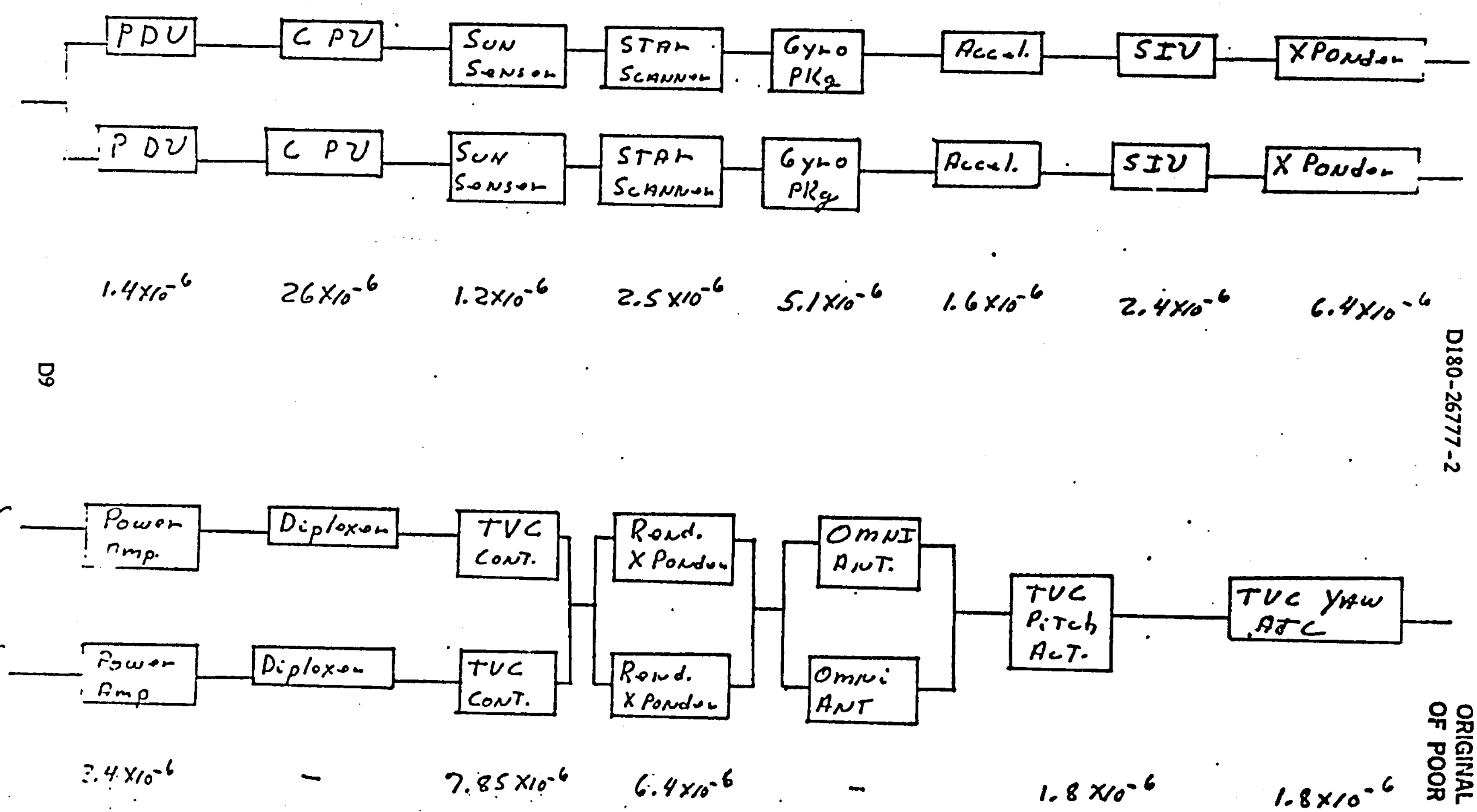
1/2

λ	— (.999999)	$1.0 \cdot 10^{-6}$ $.25 \cdot 10^{-6}$	$3 \cdot 10^{-6}$	$0.5 \cdot 10^{-6}$	TOTAL
HOLD & SUN ACQ.	—	~1.0	.999999	~1.0	.9999[illegible]
VERIFY	—	~1.0	.999964	.999994	.9999[illegible]
CRUISE	—	.999976	.988190	.998022	.9862[illegible]2
ATT. CONTROL & VERIFY	—	~1.0	.999996	.999999	.999955
BURN	—	~1.0	~1.0	~1.0	~1.0
VERIFY/ SHUTDOWN	—	~1.0	.999992	.999999	.99999[illegible]
MISSION	.999999	.999976	.988142	.998014	.986[illegible]3

ORIGINAL PAGE IS OF POOR QUALITY

D180-26777-2

Avionics Subsystem - Block Diagram
(Fig. 1.2)



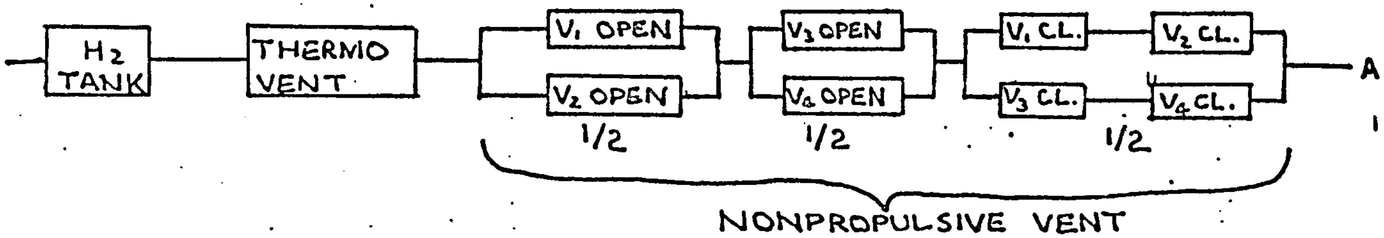
AVIONICS SUBSYSTEM - ANALYSIS

(FIG. 1.2)

	BASIC STRING	REND. EXPONDER	AINT.	TVC	TOTAL
HOLD & SUN ACQ.	~1.0	~1.0	—	~1.0	~1.0
VERIFY	~1.0	~1.0	—	~1.0	~1.0
CRUISE	.985176	.999999	—	.999524	.984706
ATT. CONTROL & VERIFY	.999954	~1.0	—	~1.0	.999954
BURN	.999994	~1.0	—	~1.0	.999994
VERIFY / SHUTDOWN	.999982	~1.0	—	~1.0	.999982
MISSION	.985110	.999999	.999999	.999524	.984639

PROPULSION SUBSYSTEM - BLOCK DIAGRAM AND ANALYSIS

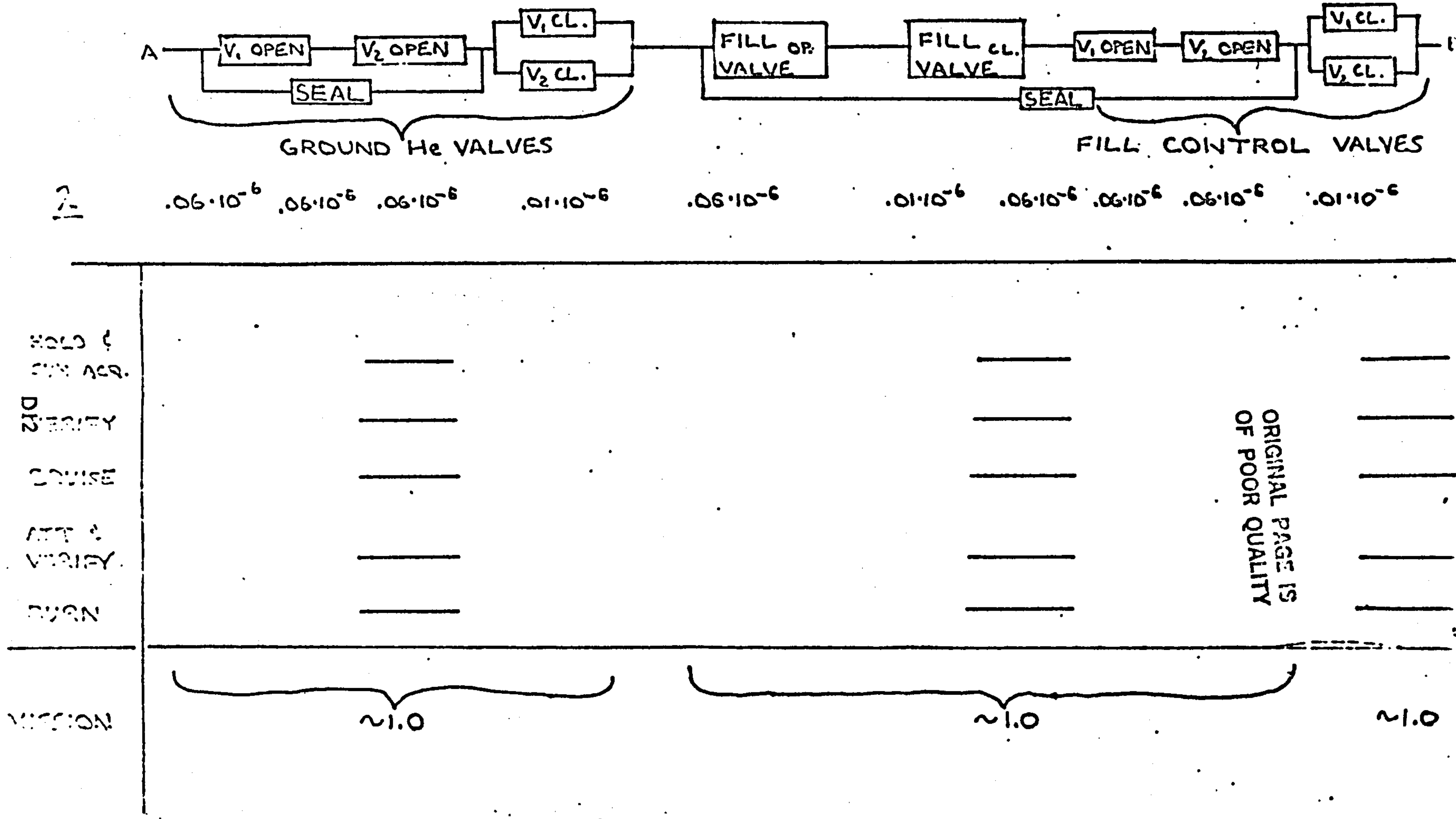
(FIG. 1.3)



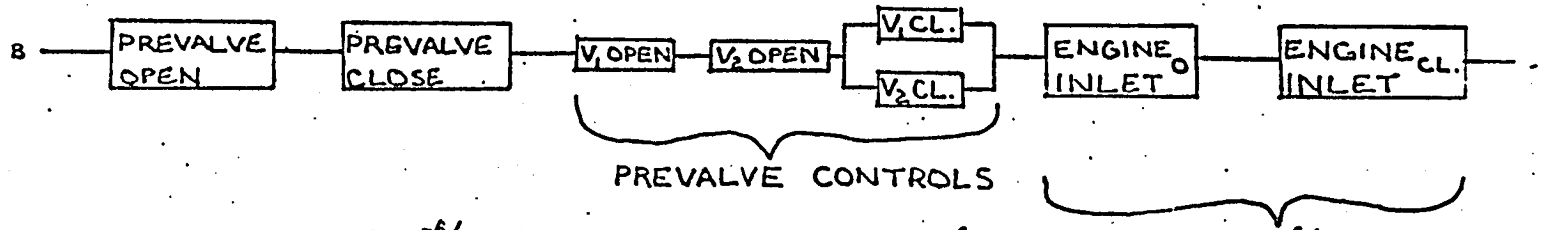
DII

λ	$.12 \cdot 10^{-6}$	$3.5 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$
HOLD & SUN ACQ.	~1.0	.999999	–	–	–	–
VERIFY	.999999	.999958	–	–	–	–
CRUISE	.999525	.986236	–	–	–	–
ATT. & VERIFY	~1.0	.999995	–	–	–	–
BURN	~1.0	~1.0	–	–	–	–
MISSION	.999524	.986189	~1.0	~1.0	~1.0	~1.0

(FIG. 1.3)


A
V1 OPEN
V2 OPEN
SEAL
V1 CL.
V2 CL.
GROUND He VALVES
FILL OP. VALVE
FILL CL. VALVE
V1 OPEN
V2 OPEN
SEAL
V1 CL.
V2 CL.
FILL CONTROL VALVES
.06·10^-6 .06·10^-6 .06·10^-6 .01·10^-6 .06·10^-6 .01·10^-6 .06·10^-6 .06·10^-6 .06·10^-6 .01·10^-6
~1.0
~1.0
~1.0

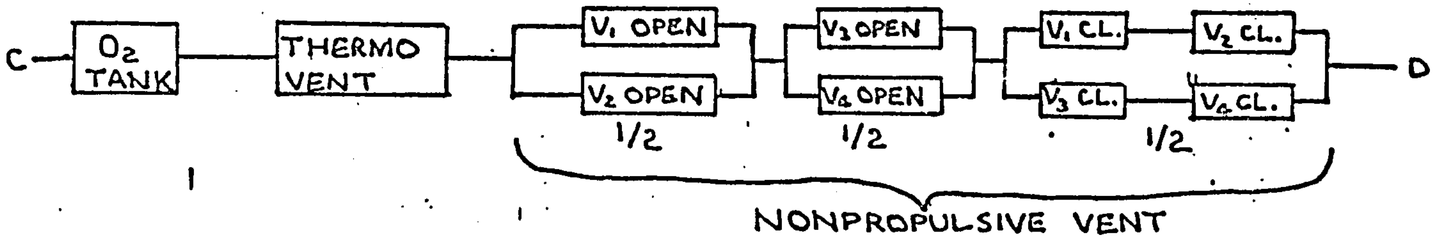
(FIG. 1.3)



	Prevalve Open	Prevalve Close	V_1 / V_2 Open	V_1 / V_2 CL.	Engine Inlet
λ	—	$125 \cdot 10^{-6}$/CYCLE		$.01 \cdot 10^{-6}$	$125 \cdot 10^{-6}$/CYCLE
HOLD & SUN ACQ.	NOT MISSION CRITICAL	—	NOT MISSION CRITICAL	—	—
VERIFY	NOT MISSION CRITICAL	—	NOT MISSION CRITICAL	—	—
CRUISE	NOT MISSION CRITICAL	—	NOT MISSION CRITICAL	—	—
ATT. & VERIFY	NOT MISSION CRITICAL	—	NOT MISSION CRITICAL	—	—
BURN	NOT MISSION CRITICAL	.999875	NOT MISSION CRITICAL	—	.999875
MISSION	—	.999875	—	~1.0	.999875

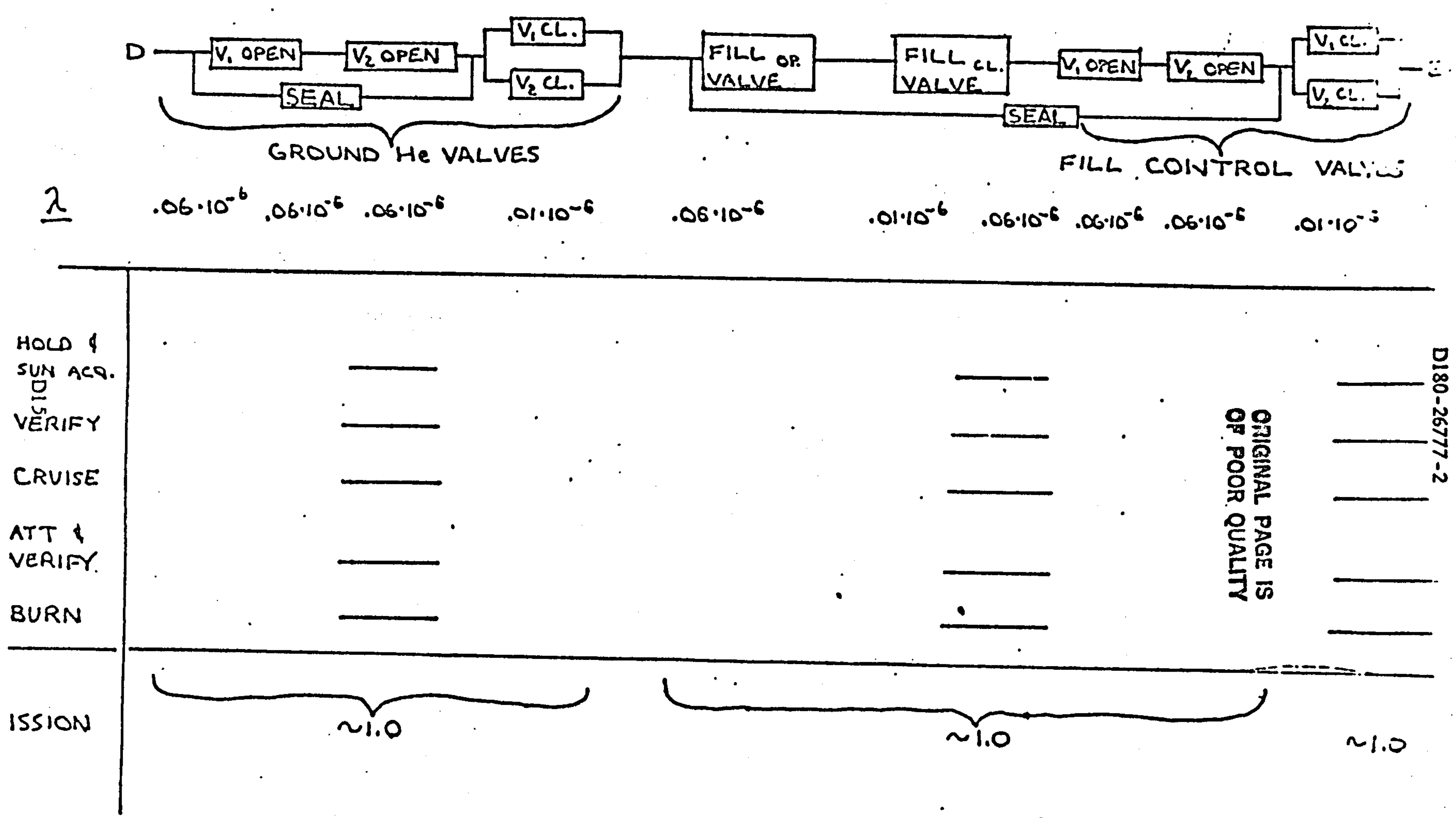
ORIGINAL PAGE IS OF POOR QUALITY

(FIG. 1.3)

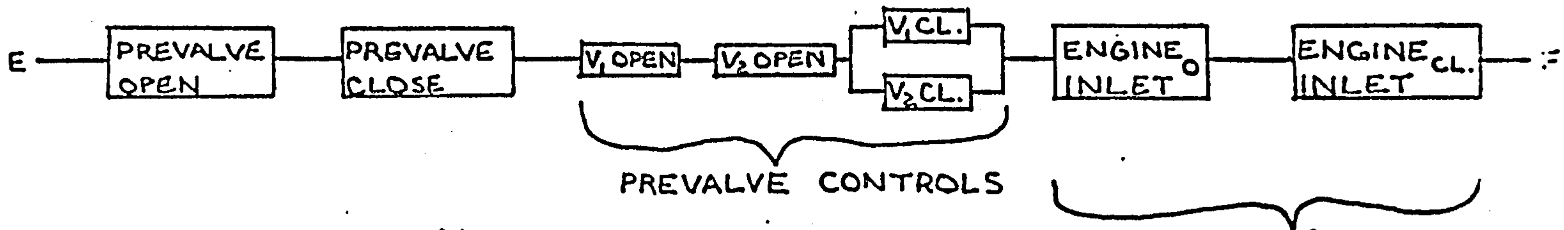


λ	$.12 \cdot 10^{-6}$	$3.5 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$
D14						
HOLD & SUN ACQ.	~1.0	.999999	–	–	–	–
VERIFY	.999999	.999958	–	–	–	–
CRUISE	.999525	.986236	–	–	–	–
ATT. & VERIFY	~1.0	.999995	–	–	–	–
BURN	~1.0	~1.0	–	–	–	–
SSION	.999524	.986189	~1.0	~1.0	~1.0	~1.0

(FIG. 1.3)

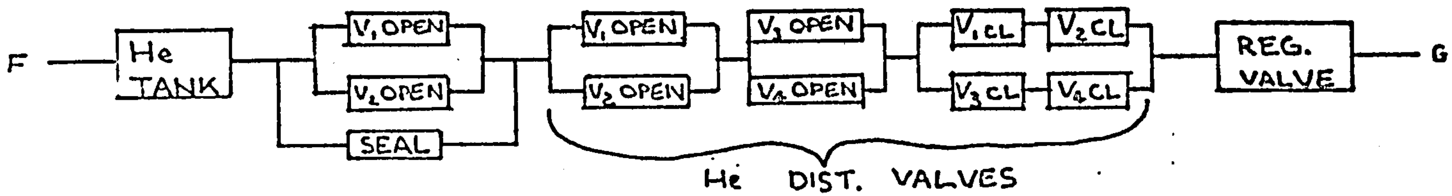

D
V1 OPEN
V2 OPEN
SEAL
V1 CL.
V2 CL.
GROUND He VALVES
FILL OP. VALVE
FILL CL. VALVE
V1 OPEN
V2 OPEN
SEAL
V1 CL.
V2 CL.
FILL CONTROL VALVES
λ
.06·10^-6 .06·10^-6 .06·10^-6 .01·10^-6 .06·10^-6 .01·10^-6 .06·10^-6 .06·10^-6 .06·10^-6 .01·10^-6
HOLD & SUN ACQ.
VERIFY
CRUISE
ATT & VERIFY
BURN
ISSION
~1.0
~1.0
~1.0

(FIG. 1.3)



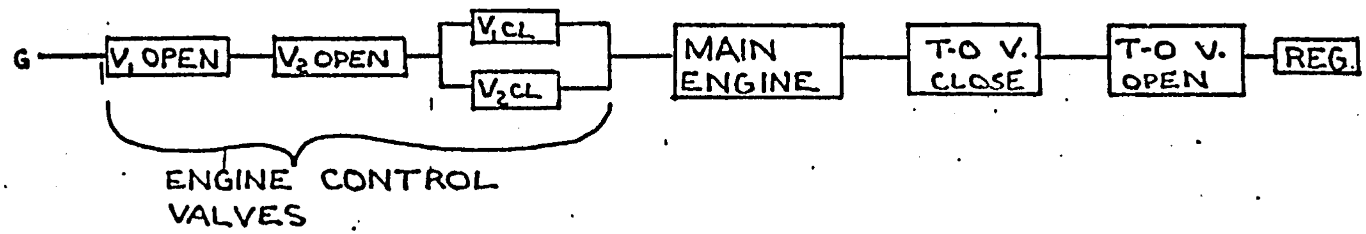
	PREVALVE OPEN	PREVALVE CLOSE	V_1 OPEN / V_2 OPEN	PREVALVE CONTROLS	ENGINE INLET
$\underline{\lambda}$	—	$125 \cdot 10^{-6}$/CYCLE		$.01 \cdot 10^{-6}$	$125 \cdot 10^{-6}$/CYCLE
HOLD & SUN ACQ.	NOT MISSION CRITICAL	—	NOT MISSION CRITICAL	—	—
VERIFY		—		—	—
CRUISE		—		—	—
ATT. & VERIFY		—		—	—
BURN		.999875		—	.999875
ISSION	—	.999875	—	~1.0	.999875

(FIG 1.3)



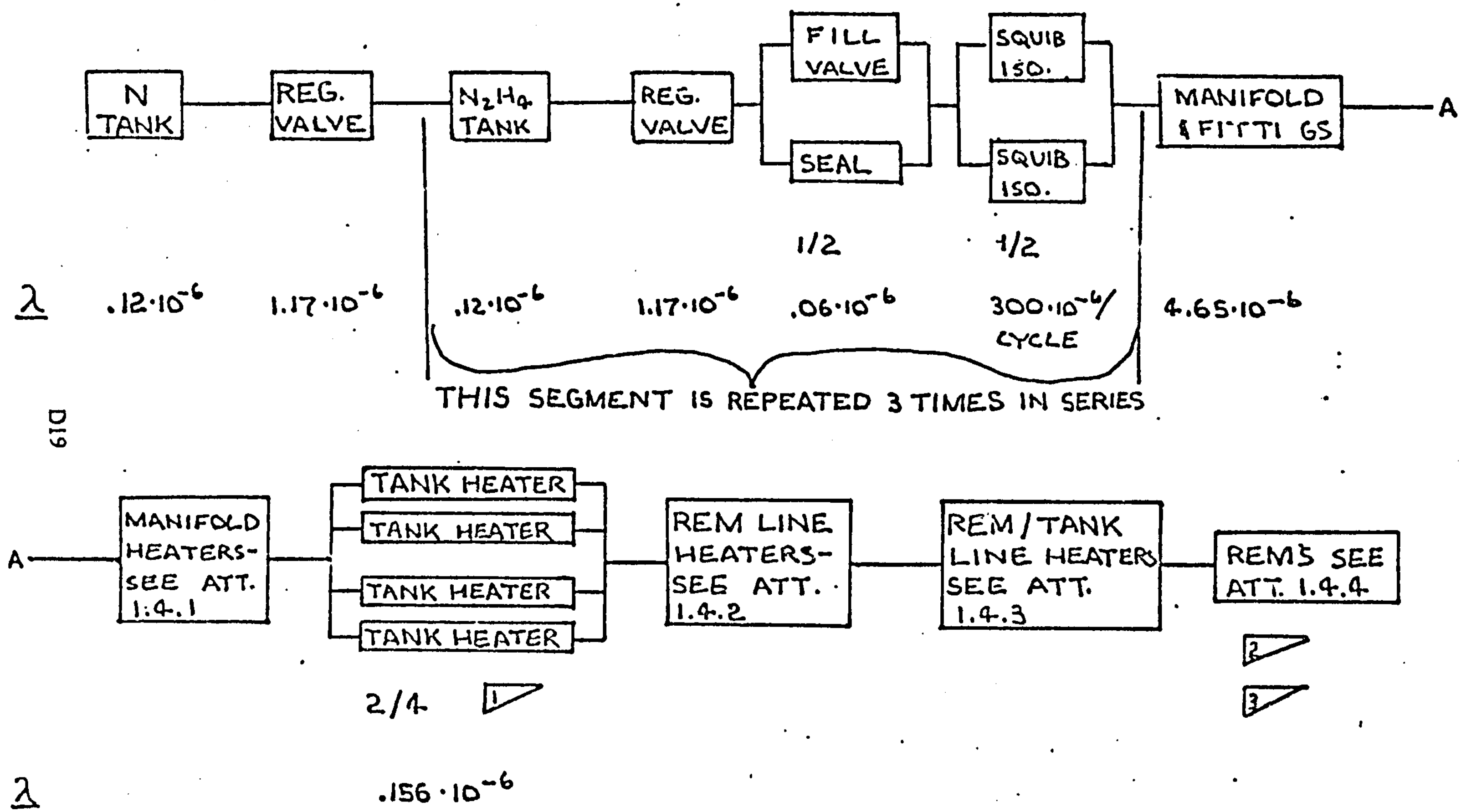
λ	$.12 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.06 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$	$.01 \cdot 10^{-6}$	$1.17 \cdot 10^{-6}$
[illegible] ACQ.	~1.0	—	—	—	—	—	~1.0
[illegible]RIFY	1.999999	—	—	—	—	—	.999986
[illegible]UISE	.999525	—	—	—	—	—	.995378
[illegible]	~1.0	—	—	—	—	—	~1.0
[illegible]	~1.0	—	—	—	—	—	~1.0
[illegible]SION	.999524	~1.0	~1.0	~1.0	~1.0	~1.0	.995364

(FIG. 1.3)



λ		$.01\cdot10^{-6}$	$2602\cdot10^{-6}$/CYCLE	$.01\cdot10^{-6}$ $.06\cdot10^{-6}$	$1.17\cdot10^{-6}$	TOTAL
HOLD & SUN ACQ.	NOT MISSION CRITICAL	—	~1.0	~1.0	~1.0	[illegible]
VERIFY		—	~1.0	~1.0	.999999	[illegible]
CRUISE		—	~1.0	.999960	.999537	[illegible]
ATT. & VERIFY		—	~1.0	~1.0	~1.0	.999990
BURN		—	.997401	~1.0	~1.0	[illegible]
MISSION	—	~1.0	.997401	.999960	.999536	[illegible]

REACTION CONTROL SUBSYSTEM - BLOCK DIAGRAM
(FIG. 1.4)

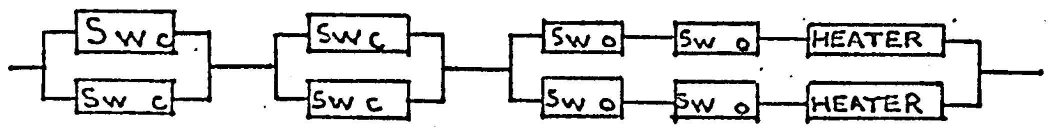


1 THE TANK HEATERS ARE REPEATED 3 TIMES IN SERIES

2 DUTY CYCLE FOR ROLL THRUSTERS - 4 REMS

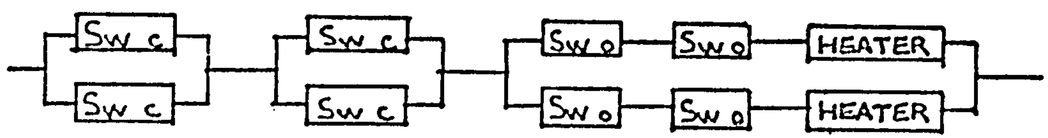
3 DUTY CYCLE FOR PITCH AND YAW THRUSTERS - 8 REMS

MANIFOLD HEATERS - FIG 1.4.1



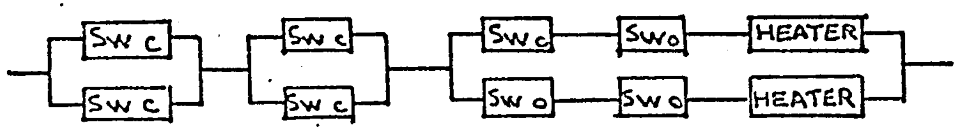
λ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.449 \cdot 10^{-6}$

REM LINE HEATERS - FIG 1.4.2



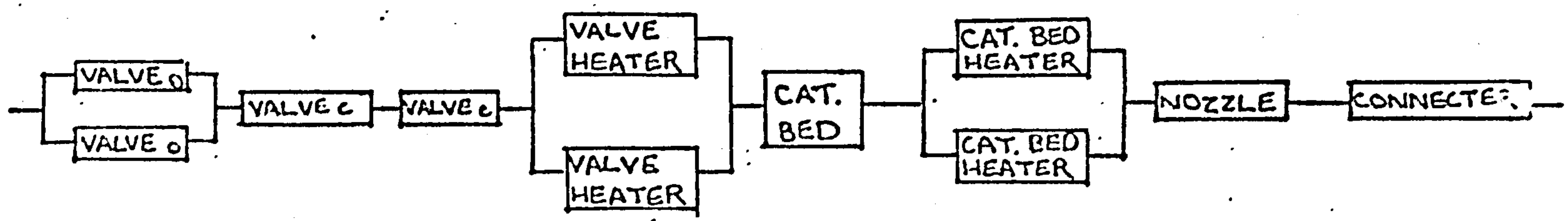
λ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.035 \cdot 10^{-6}$ $.20 \cdot 10^{-6}$

TANK LINE / REM LINE HEATERS – FIG. 1.4.3



λ $.035\cdot10^{-6}$ $.035\cdot10^{-6}$ $.035\cdot10^{-6}$ $.035\cdot10^{-6}$ $.366\cdot10^{-6}$

REMS – FIG. 1.4.4



λ $.245\cdot10^{-6}$/CYCLE $.105\cdot10^{-6}$/CYCLE $.105\cdot10^{-6}$/CYCLE $.01\cdot10^{-6}$ $.04\cdot10^{-6}$ $.01\cdot10^{-6}$ $.166\cdot10^{-6}$/CYCLE $1.41\cdot10^{-6}$

REATION CONTROL SUBSYSTEM - ANALYSIS
(FIG. 1.4)

	N & VALVE TANK	N_2H_4 & VALVE	FILL	SQUIB	MANI-FOLD	MANI HEAT	TANK HEAT	REM HEAT	REM/TA[illegible] HEATE[illegible]
HOLD & ACQ.	~1	.999999	~1	—	.999998	~1	—	~1	~1
VERIFY	.999986	.999957	~1	—	.999944	~1	—	~1	~1
CRUISE	.995299	.985962	~1	—	.981735	.999996	—	.999999	.999997
ATT. CONT.	.999998	.9999995	~1	—	.999993	~1	—	~1	~1
BURN	~1	.9999999	~1	—	~1	~1	—	~1	~1
VERIFY	.999997	.999991	~1	—	.999988	~1	—	~1	~1
MISSION	.995279	.985902	~1	~1	.981677	.999996	~1	.999999	.999[illegible]7

	REMS ①	REMS ②	TOTAL
HOLD & ACQ.	~1	~1	.999997
VERIFY	.999996	.999992	.999875
CRUISE	~1	.999772	.963195
ATT. CONT.	.999992	.999988	.999966
BURN	~1	~1	.999999
VERIFY	~1	.999988	.999964

D22

D180-26777-2

ORIGINAL PAGE IS OF POOR QUALITY

DUTY CYCLE SUBSYSTEM - FIG. 1.5

EQUIPMENT \ TIME	PRE-MISSION	HOLD*	SUN ACQ.*	VERIFY*	CRUISE	DAMP ROLL*	EARTH ACQ.*	YAW TO BURN*	VERIFY*	BURN*	VERIFY*	SHUT DOWN*	NOTES
		10 MIN.	10 MIN.	12 HR.	165 DAYS	10 MIN.	10 MIN.	10 MIN.	1 HR.	12.5 MIN.	2.5 HR.	10 MIN.	
POWER SUBSYSTEM													
SOLAR PANELS							ON						
BATTERY							ON						
BATTERY HEATERS							ON						
REGULATOR / CHARGER							ON						
DISTRIBUTION							ON						
AVIONICS SUBSYSTEM													
CPU							ON						
DISTRIBUTION							ON						
SUN SENSOR							ON						
STAR SCANNER			OFF						ON				
GYROS							ON						
LINEAR ACC					OFF					ON			

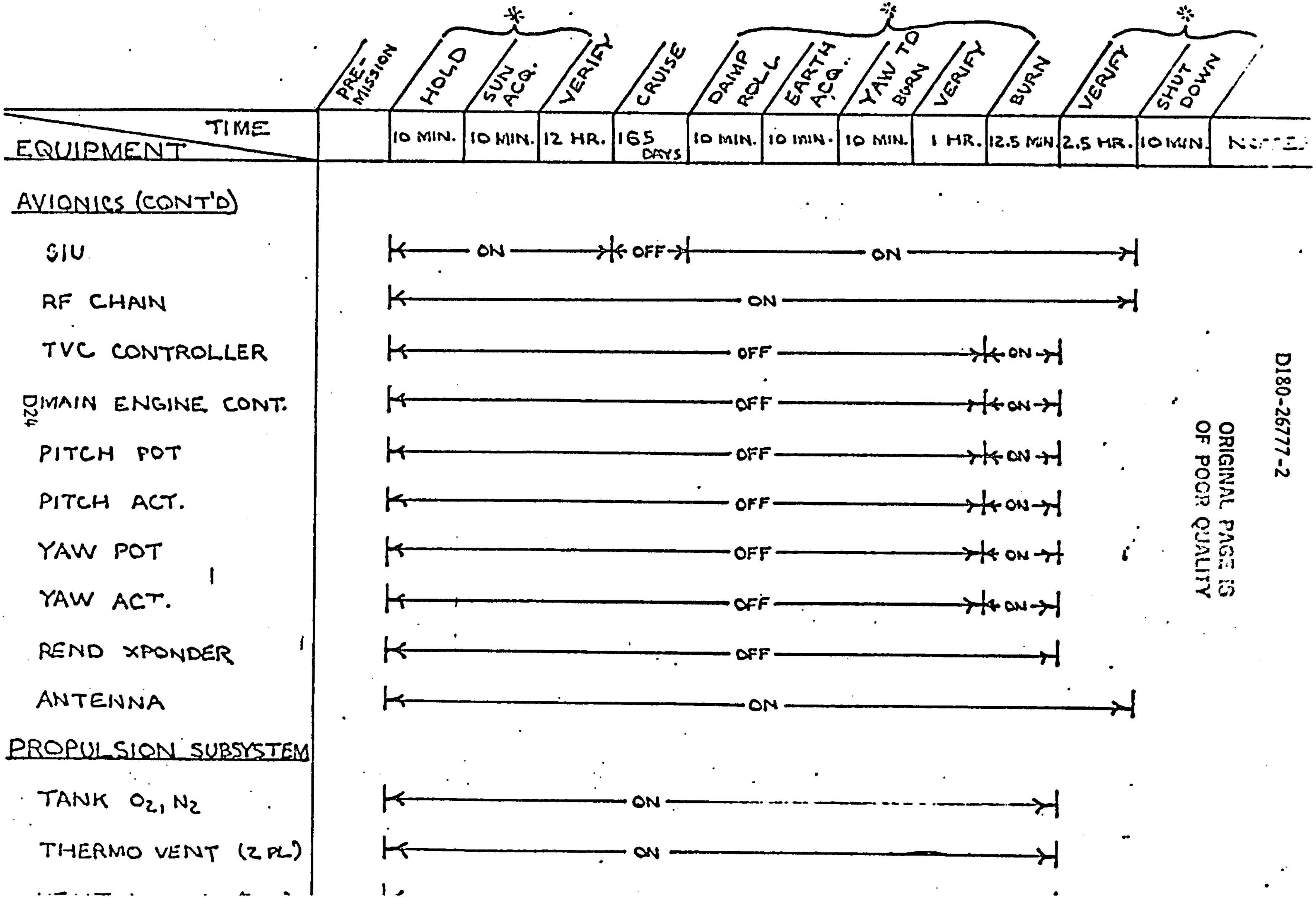
D23

D180-26777-2

ORIGINAL PAGE IS OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY

(FIG. 1.5)



(FIG. 1.5)

EQUIPMENT \ TIME	PRE-MISSION	HOLD *	SUN ACQ. *	VERIFY *	CRUISE	DAMP ROLL *	EARTH ACQ. *	YAW TO BURN *	VERIFY *	BURN *	VERIFY *	SHUT DOWN *	NOTE
		10 MIN.	10 MIN.	12 HR.	165 DAYS	10 MIN	10 MIN	10 MIN.	1 HR.	12.5 MIN.	2.5 HR.	10 MIN.	
PROPULSION (CONT'D)													
He SUPPLY					ON CLOSED								
[illegible] VENT VALVES (2 PL)					CLOSED								
FILL & DRAIN VALVES (2 PL)					CLOSED								
FILL & DRAIN He VALVES (2 PL)					CLOSED								
PREVALVE (2 PL)					CLOSED								
PREVALVE He VALVES (2 PL)					CLOSED								
He SUPPLY VALVES					CLOSED								
TAP OFF VALVE (2 PL)					OFF					ON			
TAP OFF REG. (2 PL)					OFF					ON			
He REG.					ON								
START VALVES He					CLOSED					ON			
INLET VALVES (2 PL)					CLOSED					ON			
ENGINE					OFF					ON			

(FIG. 1.5)

EQUIPMENT / TIME	PRE-MISSION	HOLD	SUN ACQ.	VERIFY	CRUISE	DAMP ROLL	EARTH ACQ.	YAW TO BURN	VERIFY	BURN	VERIFY	SHUT DOWN	NOTES
		10 MIN.	16 MIN.	12 HR.	165 DAYS	10 MIN.	10 MIN.	10 MIN.	1 HR	12.5 MIN.	2.5 HR.	10 MIN.	
RCS SUBSYSTEM													
N_2 TANK						ON							
N_2 TANK FILL						ON							
N_2H_4 TANK & FITTING (3)						ON							
D26 N_2 ISO. & REG. VALVE (3)						ON							
N_2H_4 FILL VALVE (3)						ON							
N_2H_4 ISOLATION (SQUIB) VALVE (3)						ON							
MANIFOLD						ON							
TANK HEATERS (3)						ON							
MANIFOLD HEATERS (3)						ON							
REM LINE HEATERS (4)						ON							
LINE TANK HEATERS (4)						ON							
REM VALVES (24)						ON							
VALVE HEATERS (24)						ON							

(FIG. 1.5)

EQUIPMENT \ TIME	PRE-MISSION	* HOLD	* SUN ACQ.	* VERIFY	CRUISE	* DAMP ROLL	* EARTH ACQ.	* YAW TO BURN	* VERIFY	* BURN	* VERIFY	* SHUT DOWN	
		10 MIN.	10 MIN.	12 HR.	165 DAYS	10 MIN.	10 MIN.	10 MIN.	1 HR	12.5 MIN.	2.5 HR.	10 MIN.	
RCS (CONT'D)													
CAT BED (24)		├				ON					┤		
CAT BED HEATER (24)		├				ON					┤		
CONNECTORS (24)		├				ON					┤		
NOZZLE (24)		├				ON					┤		

REM DUTY CYCLE

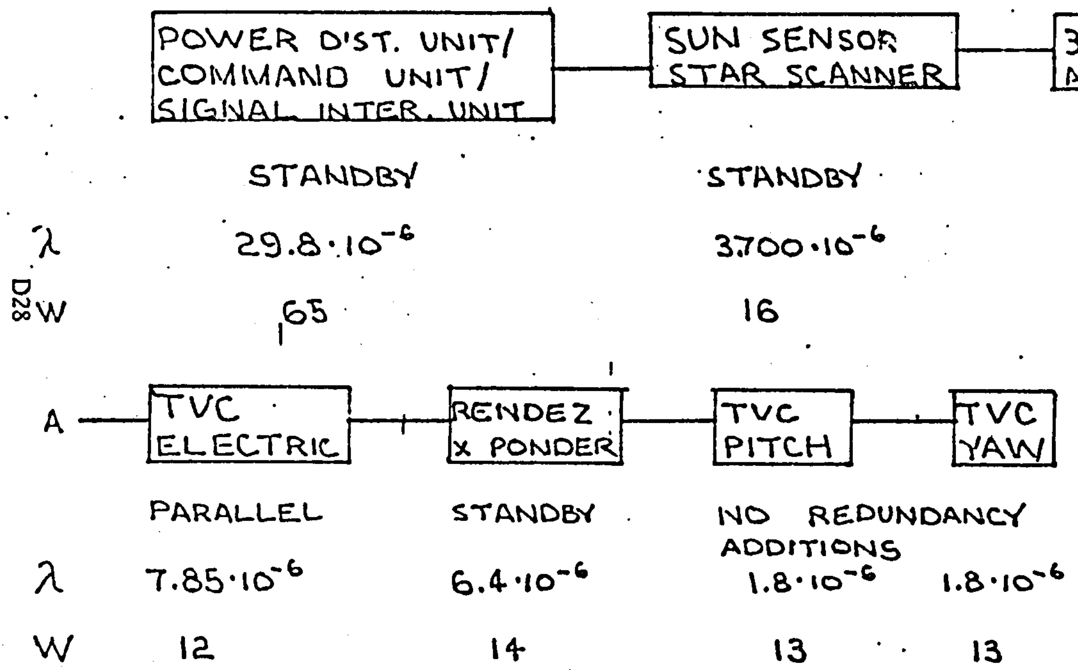
PHASE	ROLL	PITCH & YAW
HOLD	250 CYCLES/HR	250 CYCLES/HR
VERIFY	250 CYCLES/HR	250 CYCLES/HR
CRUISE	OFF	.5 CYCLES/HR
AT-CONTROL	250 CYCLES/HR	250 CYCLES/HR
BURN	250 CYCLES/HR	250 CYCLES/HR
VERIFY	250 CYCLES/HR	250 CYCLES/HR

D180-26777-2

ORIGINAL PAGE IS OF POOR QUALITY

SINGLE THREAD BLOCK DIAGRAM - FIG 2.1

AVIONIC SUBSYSTEM



D180-26777-2

ORIGINAL PAGE IS
OF POOR QUALITY

(FIG. 2.1)

POWER SUBSYSTEM

	REG. CHARGER / POWER DIST.	BATTERY / HEATER
	PARALLEL	PARALLEL
λ	$1.25 \cdot 10^{-6}$	$3.5 \cdot 10^{-6}$
W	55	192

PROPULSION SUBSYSTEM

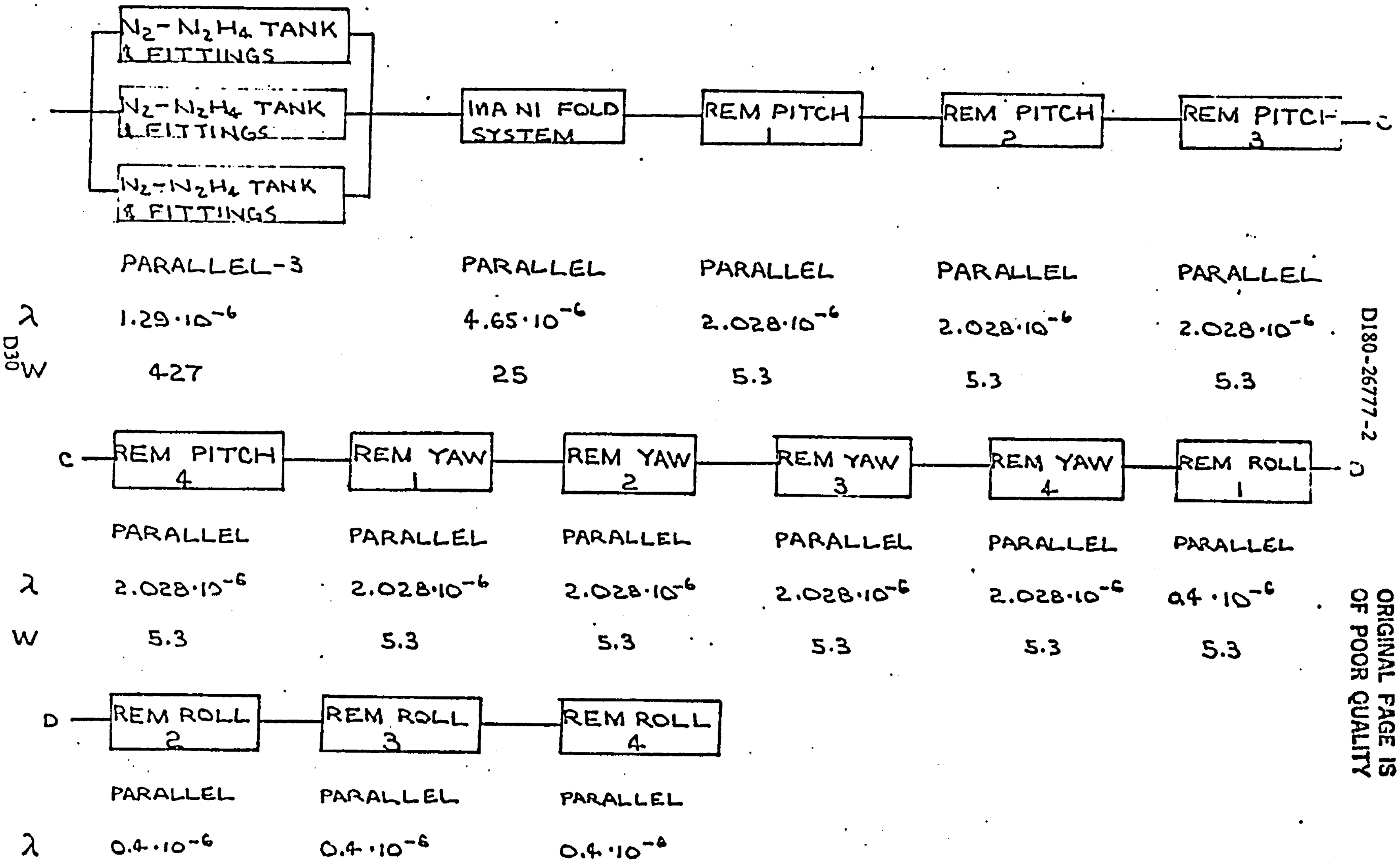
	H_2 THERMO VENT	O_2 THERMO VENT	He TANK & PROPELLANT	He REG. VALVE	T-O REQG. VALVE - H_2
	STANDBY	STANDBY	PARALLEL	PARALLEL	PARALLEL
λ	$3.5 \cdot 10^{-6}$	$3.5 \cdot 10^{-6}$	$.12 \cdot 10^{-6}$	$1.17 \cdot 10^{-6}$	$1.17 \cdot 10^{-6}$
W	30	30	6	2	5

	T-O REG. VALVE - O_2	MAIN ENGINE / H_2, O_2 TANK / PREVALVE / INLET
	PARALLEL	NO REDUNDANCY ADDITIONS
λ	$1.17 \cdot 10^{-6}$	$1.02 \cdot 10^{-6}$
W	[illegible]	[illegible]

ORIGINAL PAGE IS OF POOR QUALITY

(FIG. 2.1)

REACTION CONTROL SUBSYSTEM

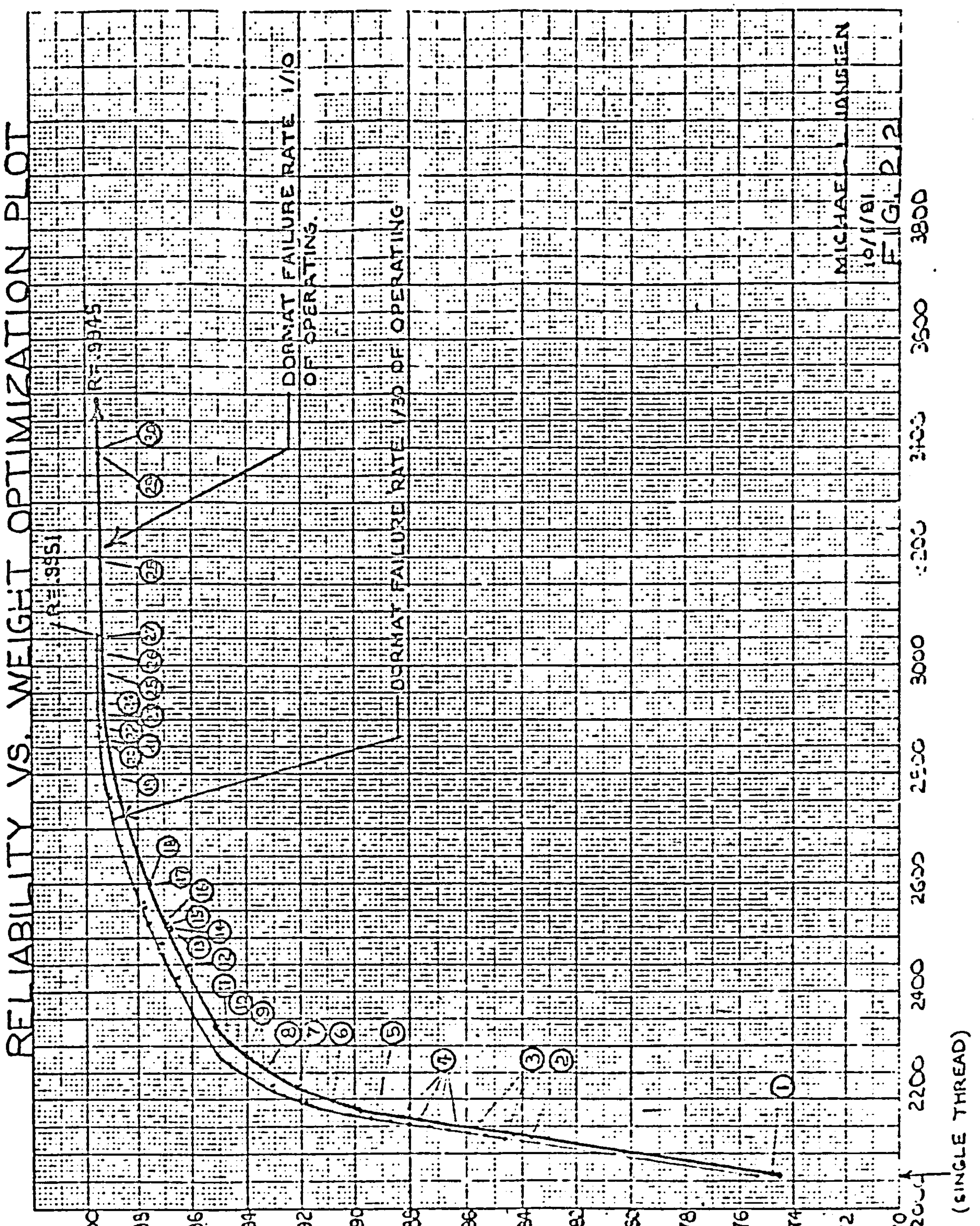


D30

D180-26777-2

ORIGINAL PAGE IS OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY



SYSTEM RELIABILITY

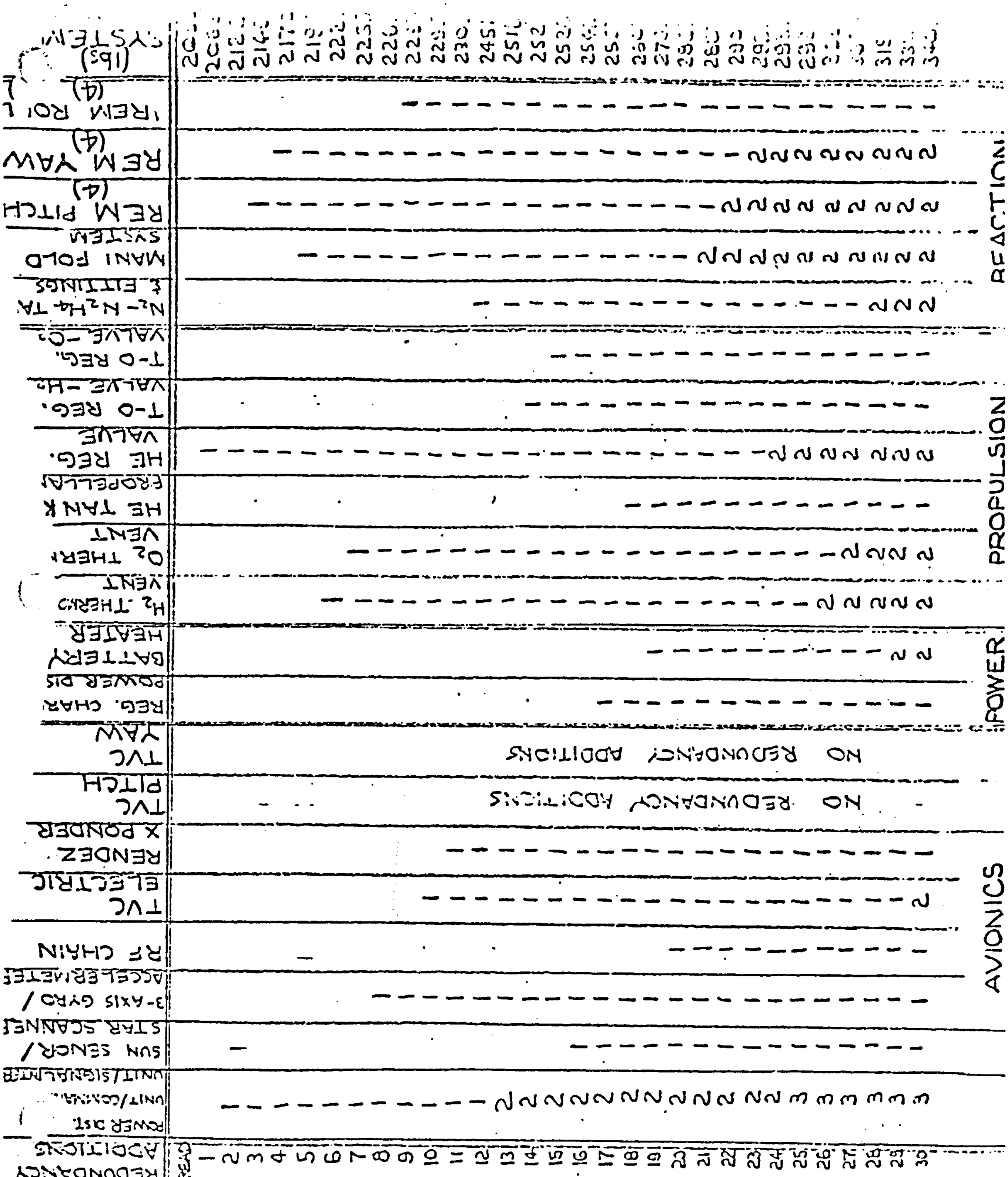


TABLE 2.1
D32 REDUNDANCY ADDITIONS

APPENDIX E

RECOMMENDED CHANGES TO SYSTEM SAFETY DOCUMENT AND CONFORMANCE TO SYSTEM SAFETY GUIDELINES, SECTION 4.0, "SYSTEM SAFETY ASPECTS FOR REFERENCE CONCEPT"

BOEING AEROSPACE COMPANY

P.O. Box 3999
Seattle, Washington 9812

A Division of The Boeing Compar

2-4032-0081-121
10/19/81

Dr. Eric E. Rice
Space Systems and Applications Section
Battelle's Columbus Laboratories
Columbus, OH 43201

Dear Eric:

Here are the results of our review of the second draft of the "System Guidelin Document for Nuclear Waste Disposal in Space". It is a comprehensive and important document and will be a valuable aid in our definition of the reference space system
The document was reviewed in two steps. In the first, recommended charges to the document itself were identified. In the second, the reference space system was reviewed for conformance to Section 4.0, "System Safety Aspects for the Reference Concept". Results of these efforts are described below.

1.0 Recommended Charges to Document - Most changes recommended are intended to bri the specification into closer conformance with the approach to waste payload protection adopted by the 1980-81 study without compromising their generality. Deletion c references to elements of the waste payload no longer used is suggested; addition of specifications for the waste form support structure, or "core" is also recommended as the core can be considered a generic element of the waste payload. An addition to Section 2.3.3, on/or near pad or ascent booster accident environments covering shuttle crash conditions is included.

Revisions are shown with a short rationale for each change included. Enclosure "A" is a copy of the relevant pages from the document with recommended revisions indicated. Enclosure "B" summarizes the rationale for each change. Rationales are keyed to the markups in the document.

2.0 Conformance to System Safety Document Guidelines Section 4.0, "System Safety Aspects for Reference Concept - Results of our review of the reference system for conformance are summarized in Enclosure C. Each statement of conformance is keyed to the appropriate paragraph in the draft document. In general, complete compliance is indicated for nominal conditions. Compliance for some accident conditions remains TBD pending further analysis.

R. P. Reinert

RPR/rdr

Enclosures

cc: Pete Priest

BOEING

Enclosure "A". Recommended Changes To System Safety Guidelines Document.

TABLE OF CONTENTS (Con't)

ORIGINAL PAGE IS OF POOR QUALITY

[4.1.5 Primary Container and Flight Radiation Shielding

[Waste Form Support Structure.

[4.1.6 Waste Payload Protection System.

1.

ORIGINAL PAGE IS
OF POOR QUALITY

TABLE 3. SPECIFIC COMPONENTS OF CONTAINMENT

Parameters	Components	Mission Phases
• Thermal	• Waste Form (2)	• Fabrication/Assembly
	* • Waste Form Support Structure	
• Mechanical	* • Primary Container	• Terrestrial Transport
• Chemical	* • Radiation Shield	• Launch Site Handling
	* • Impact Absorber	• Launch to Earth Orbit
	* • Ablation Shield	• Orbit Transfer to Destination
	• Shipping Cask	

* may be partially or Totally integrated. (3)

TABLE 4. THERMAL GUIDELINES FOR CONTAINMENT OF HIGH-LEVEL WASTE FOR SPACE DISPOSAL

	Mission Phase				
Component	Fabrication/ Assembly	Terrestrial Transport	Launch Site Handling	Launch to Earth Orbit	Orbit Transfer to Destination
Waste Form	40% Melt/ 90% Melt*	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt
Primary Container	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt
Flight Radiation Shield	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt
Impact Absorber	—	—	40% Melt/ 90% Melt	40% Melt/ 90% Melt	40% Melt/ 90% Melt
Ablation Shield	—	—	40% Melt/—	40% Melt/—	40% Melt/—
Shipping Cask	—	DOT, NRC Reg.	—	—	—

*Note: The normal absolute temperature limit is given first: the accident absolute temperature limit is given second. If the melt absolute temperatures are not appropriate for the material in question, then the 90% of the fabrication absolute temperature should apply.

INSERT 2.2.3.2 Waste form Support structure (Attached.)

4

ORIGINAL PAGE IS
OF POOR QUALITY

2.2.3.2 Primary Container

The primary container, designed to enclose the waste form throughout all mission phases beyond waste form fabrication, is also the primary containment boundary. The thermal limit for normal conditions is 40% of the melt absolute temperature. For accident conditions, 90% of the absolute melt temperature is the guideline. Mechanical limits are yield (normal) and ultimate strengths or low dispersion (accident). Chemical limits are covered by existing federal regulations (U.S. NRC, 1978). 5

[using a safety factor 1.5 ~~under~~ for normal lc 6.

2.2.3.~~3~~-4 Radiation Shield

The radiation shield for flight should be designed to function during all mission phases through transfer to the final destination. The radiation shield should be supplemented with auxiliary shielding materials, as needed during various mission phases, such that radiation exposure limits are not reached. For mechanical strength, 90% of the yield (normal) and 90% of the ultimate (accident) stress limits apply (ultimate does not apply for launch and orbit transfer operations). Thermal limits are 40% of the melt absolute temperature (normal conditions) and 90% of the absolute melt temperature (accident conditions). Chemical requirements will be similar to those in existing federal regulations (U.S. NRC, 1978).

Radiation shielding limits for the payload package (1000 mrem/hr at 1 m) have been assumed for conditions not covered by existing regulations. Conservative limits (such as those for transportation) have not been selected due to the sensitivity of the overall system design (payload/shield mass ratio) to the dose limits. Rather, the guideline limits chosen reflect the fact that the waste payload package will be isolated from the general public throughout all but a small fraction of its lifetime.

2.2.3.~~4~~5 Impact Absorber and Ablation Shield

The impact absorber and ablation shield, have similar containment limits. For thermal guidelines, 40% of the melt (normal) and 90% of the melt (accident) absolute temperature apply to the impact absorber; 40% of ~~[illegible]~~ minimum ablat temperature applies as the upper limit for the ablation shield under normal conditions. For mechanical strength, yield (normal) limits exist. The absorber and ablation shield will be chemically non-reactive with other containment layers (similar to other DOT/NRC regulations). They will be non-pyrophoric. The impact absorber is designed to absorb mechanical energy during accidents. The yield strength of the absorber material is expected to be exceeded. Therefore, the ablation shield which is designed to reduce heating effects during possible reentry phases, is not expected to survive ground or water impact.

2.2.3.~~5~~6 Shipping Cask

During ground-based Earth transport, the high-level waste package will be enclosed within a shipping cask. Current U.S. federal regulations [10 CFR 71

Add paragraph 2.3.3.2 as follows:

2.3.3.2 Waste Form Support Structure

The waste form support structure, or core, is designed to (1), provide structure support to the waste payload during all mission phases by providing the mechanical interface between waste form and primary container. (2), provide a thermal conductance path from th waste form to the primary container, and (3), enhance the primary container structural integrity by serving as a solid, incompressible core with elastic modulus comparable to the primary container material.

- An impact in the worst orientation into 25 C water at a velocity 10 percent higher than the predicted terminal velocity, followed by a descent into the ocean to a depth corresponding to a hydrostatic pressure of 12,000 N/cm^2.

INSERT "C" ATTACHED

2.3.4 Reentry Accidents

(8) The payload package shipped to its space destination must be able to withstand inadvertent 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 that must be considered for each type of disposal mission are defined as follows:

- A decaying reentry trajectory (shallow angle Skylab type) to provide maximum heating energy possible
- A reentry trajectory (steep angle) which provides the maximum heating flux possible
- An impact in the worst orientation onto an unyielding surface (western granite) at a velocity 10 percent higher than the predicted terminal velocity or an impact onto land such that the reentering waste payload is buried in low conductivity soil (k = TBD), but the waste form does not reach 90% of the melt absolute temperature.
- An impact in the worst orientation into 25 C water at a velocity 10 percent higher than the predicted terminal velocity, followed by a descent into the ocean to a depth corresponding to a hydrostatic pressure of 12,000 N/cm^2.

The response of the payload package to the reentry environments mentioned above should be calculated after the possible reentry conditions have been determined by analysis for a specific disposal mission type.

2.4 Criticality

The radioactive waste payload package must be subcritical (calculated K-effective $+3\sigma \leq 0.95$) for normal operations or any possible credible accident during processing, fabrication, handling, storage, or transport to the space destination. Calculations should show that any credible change in waste form geometry and any credible grouping of packages will not cause K-effective $+3\sigma$ to exceed 0.95.

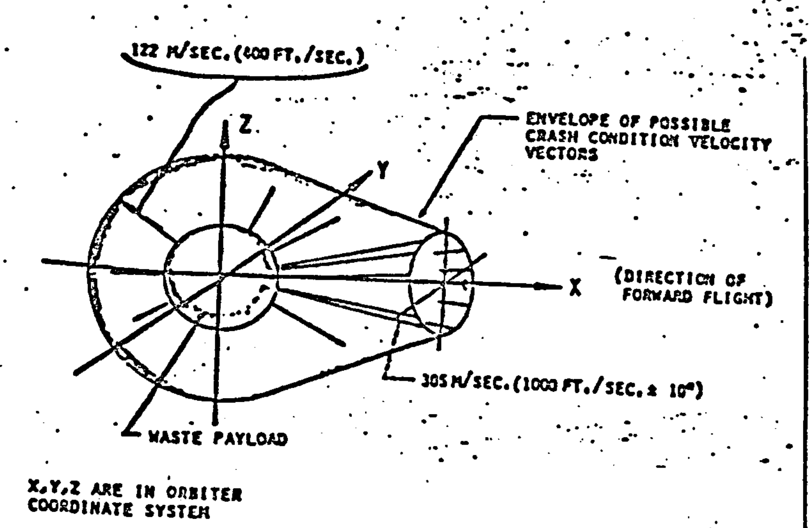
2.5 Postaccident Recovery

Postaccident recovery teams should be made part of the operational disposal system. They should be responsible for all accident recovery operations, including accidents involving processing, payload fabrication and railroad

ORIGINAL PAGE IS
OF POOR QUALITY

Insert "C"

An impact while restrained in the flight support system mounted in the Orbiter cargo bay at any of the combinations of velocity and direction as shown in the following figure, followed by a TBD crushing load imposed by the Orbiter structure.



Recommended Design Criteria for Impact Angle

ORIGINAL PAGE IS
OF POOR QUALITY

3.2 Waste Form

The reference waste form for space disposal is the Oak Ridge National Laboratory (ORNL) iron/nickel based cermet (ceramic/metal matrix), a dispersion of ceramic particles in a continuous metallic phase. This waste form has been chosen over other waste forms because of the expected responses to possible accident environments. The cermet is expected to have a waste loading of the order of 67.4% where 100% is defined as high-level waste in oxide form. The thermal conductivity is expected to be about 9.5 Watts/m-C and the density is about 6.5 g/cc.

3.3 Waste Processing and Payload Fabrication

REPLACE WITH INSERT "B", attached.

The cermet waste form would be made into cylindrical billets approximately 5 cm long and 5 cm in diameter. They would be placed into a solid spherical steel container system where holes have been milled into the solid sphere accommodate the waste form billets. Figure 3 shows the waste container/integral shield concept. At the payload fabrication facility, after the billets have been installed into the container system, the "lid" on the container/integral shield would be placed into position and then welded with an electric beam. The thick steel container wall serves as a radiation shield, as well as an impact structure to protect against breaches of containment. Before further handling, the container outer wall would be decontaminated. Individual graphite reentry tiles would be placed on the container by means of bolts. Then two half shells of steel would be placed around the graphite covered sphere (see Figure 4) and seal welded. Then the system is placed into a shipping cask for shipment to the launch site.

3.4 Shipping Casks and Ground Transport Vehicles

For transport from the waste processing and fabrication facilities, the waste package would be housed in a shipping cask. The cask would be licensed by the U.S. NRC and would be transported by rail to the launch site. The cask system ~~would require~~ will provide an active cooling system if required. 10

3.5 Launch Site Facilities and Operations

Upon arrival at the launch site the waste package would be removed from the cask and placed into its flight support structure system and stored for launch.

3.6 Uprated Space Shuttle Vehicle

The Uprated Space Shuttle vehicle is defined as having oxygen/RP-1 liquid rocket boosters (LRB's) replacing the Solid Rocket Boosters (SRB's). This not only provides for a 45,000 kg ayload, but allows increased safety for the launch ascent phase and a lower launch cost.

Insert "B"

3.3 Waste Processing and Payload Fabrication

The cermet waste form would be made into cylindrical billets approximately 5 cm long and 5 cm in diameter. They would be placed into a solid spherical waste form support structure or core. The core has 241 parallel holes bored in it to accomodate the stacked cylindrical billets (Figure 3). At the payload fabrication facility the billets would be installed in the core using an automatic loading machine. Covers at both ends of each bore would be installed to retain the billets.

The loaded core would then be lowered into the lower half of the container/integral shield. The upper half of the integral shield would then be lowered into place, and upper and lower shield halves are then electron-beam welded together. Almost all of the graphite/steel "tiles" would be preinstalled on the shield halves using boys prior to shield assembly. A "belt" around the equator would be left free of tiles to allow the electron-beam weld. Following the weld, the remaining tiles would be installed using remote handling equipment, (see Figure 4). The waste payload is then ready for placement in a shipping cask for transportation to the launch site.

ORIGINAL PAGE IS
OF POOR QUALITY

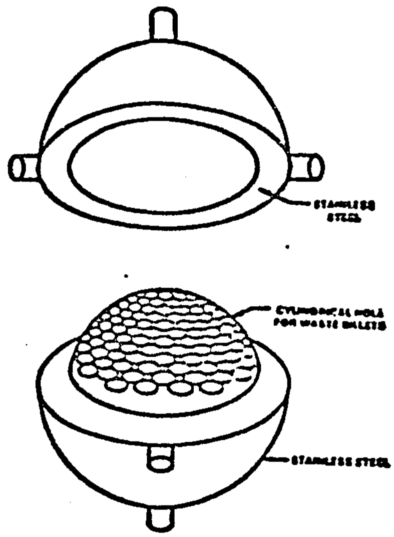
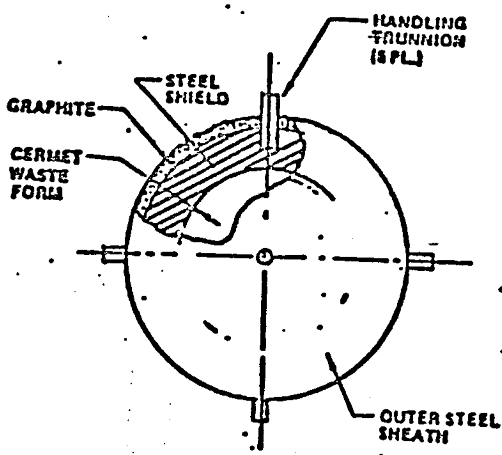


FIGURE 3. WASTE CONTAINER AND INTEGRAL RADIATION SHIELD

we will supply updated figures by 10-13.



110

FIGURE 4. WASTE ~~PACKAGE~~ PAYLOAD SYSTEM WITH ~~CONTAINER~~, INTEGRAL STEEL SHIELD, GRAPHITE NEUTRON MODERATOR AND ~~THERMAL~~ RE-ENTRY PROTECTION AND STEEL PROTECTIVE SHEATH

ORIGINAL PAGE IS OF POOR QUALITY

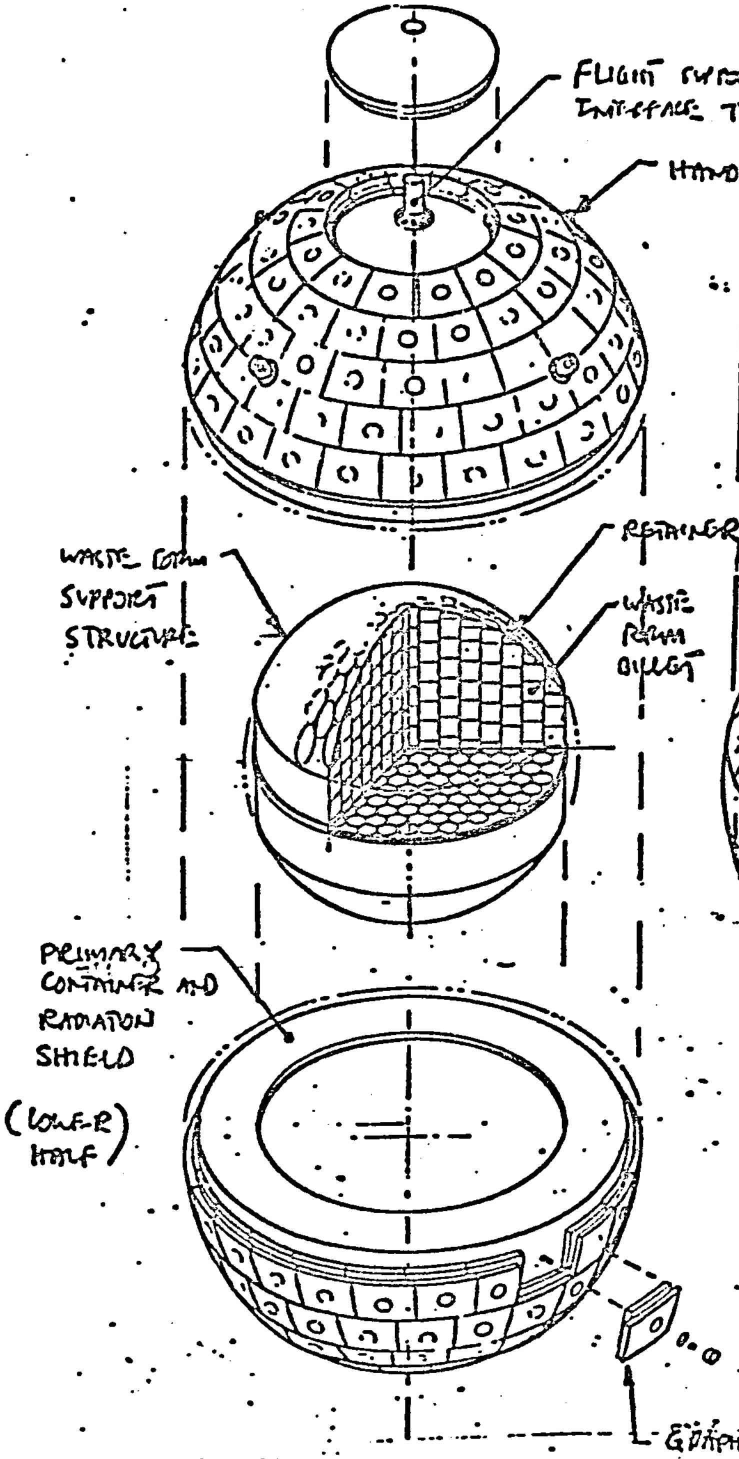


FIGURE 3. EXPLODED VIEW OF WASTE PAYLOAD SYSTEM.

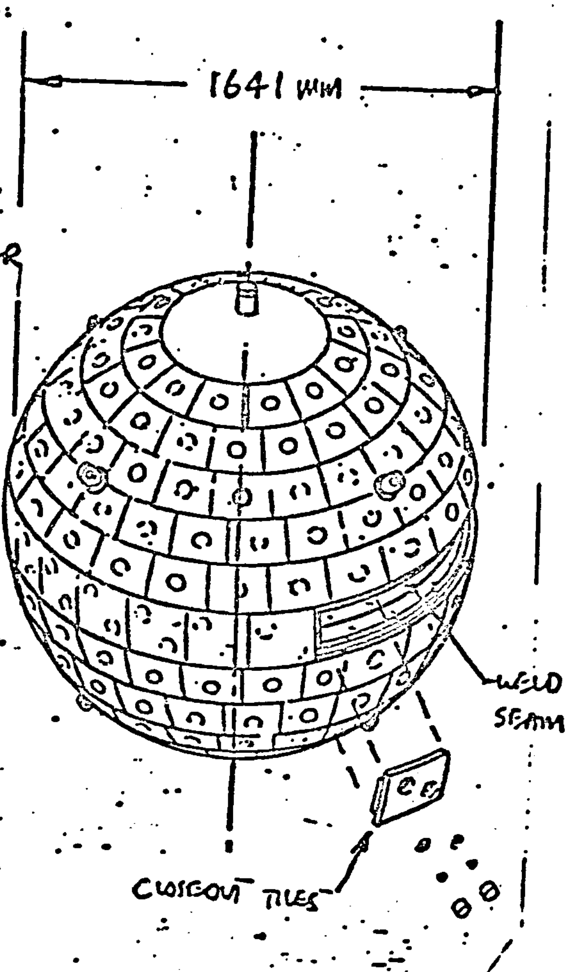


FIGURE 4. ASSEMBLED WASTE PAYLOAD SYSTEM.

4.1.1 Waste Form

For normal conditions, a cermet temperature of 1050 C (90% of fabrication absolute temperature) will not be exceeded. For accident conditions, a cermet temperature of 1280 C (90% of melt absolute temperature) shall not be exceeded. Criticality requirements will also be met.

4.1.2 Waste Processing and Payload Fabrication Facilities

The design and operation of these facilities will follow current proposed regulations, as specified for reprocessing plants.

4.1.3 Shipping Casks and Ground Transport Vehicles

Shipping casks and ground transport vehicles will comply with DOT and NRC regulations. The maximum outside diameter of the shipping cask will be 3.05 meters (10 feet). When required for heat rejection, a redundant cooling system for the shipping cask will be required.

(12) Waste ~~payload~~ forms support structure

4.1.4 ~~Payload Primary Container~~

structure used to support the

For normal conditions, the primary stainless steel container shall not exceed a temperature of 416 C (40% of melt absolute temperature). No chemical and physical interaction will occur between the cermet waste form and the container. For accident conditions, the primary container must not exceed a temperature of 1280 C (90% of melt absolute temperature).

(13) Primary container and

4.1.5 Flight Radiation Shielding

Radiation shielding including outer layer shielding contributions for flight systems, will be designed to limit radiation to no more than 1000 mrem per hour at 1 meter from the package surface under normal conditions. The shield itself, when stripped of all outer "non-shielding" layers of the payload package, will not exceed 2000 mrem per hour at 1 meter from the shield. Auxiliary shielding will be designed such that radiation exposure limits (see Tables 1 and 2) for ground personnel and flight crews are not exceeded during handling or ~~flight~~ operations.

(14) For normal conditions, the temperature limit for the stainless steel ~~flight~~ primary container radiation shield is 416 C (40% of melt absolute temperature). For accident conditions, the stainless steel radiation shield must not exceed the temperature of 1280 C (90% of melt absolute temperature).

Insert "D" (15) Waste payload protection system.

4.1.6 ~~Reentry Systems~~

(16) waste payload protection ~~The reentry system~~ for the Reference Concept during contingency or accident situations is provid ~~includes two basic systems: The booster vehicle reentry system and the payload package reentry system.~~

ORIGINAL PAGE IS
OF POOR QUALITY

INSERT "D"

The ~~reentry system for the~~ waste payload primary container ~~package~~ must include provisions to survive expected on-pad and reentry accident environments. The system must include: (1) provisions for absorbing the expected external impact loads; (2) a fire and reentry thermal protection system; (3) a transmitter for recovery. The thermal protection system will not ablate more than 50% of its initial thickness during postulated worst-case reentry environments. The outer side of the package will have proper labeling.

ORIGINAL PAGE IS
OF POOR QUALITY

The booster vehicle reentry system is the Space Shuttle Orbiter. It has the capability to detach from the ET and perform a controlled maneuver to a proper safe landing site (return-to-launch site, abort-to-contingency landing strip, abort-to-orbit, abort-to-sea or abort-to-land) at almost any time in the flight. The Orbiter has sophisticated and redundant guidance and control systems, an elaborate thermal protection system, as well as a manned crew, which will all aid in the safe return to Earth of the payload package as a result of a critical ascent booster system failure. In addition, the Orbiter will carry a structural pallet (supports the waste during launch) that will reduce the Orbiter crash landing loads placed on the payload package. Also, the Orbiter will provide systems which will allow for Orbiter flotation in the event of a ditching at sea.

The reentry system for the waste payload package must include provisions to survive expected on-pad and reentry accident environments. The system must include: (1) provisions for absorbing the expected external impact loads; (2) a fire and reentry thermal protection system; (3) a transmitter for recovery. The thermal protection system will not ablate more than 50% of its initial thickness during postulated worst-case reentry environments. The outer side of the package will have proper labeling.

4.1.7 Launch Site Facilities

The launch pad used for launching nuclear waste into space should be a dedicated pad. The Nuclear Payload Preparation Facility (NPPF) should be designed to be a total containment facility.

4.1.8 Uprated Space Shuttle Launch Vehicle

The Uprated Space Shuttle launch vehicle design will reflect considerations of keeping on-pad accident environments as low as possible. The overall minimum vehicle launch reliability will be greater than 0.999 (meaning one chance in 1000 that the Orbiter vehicle will be catastrophically lost during liftoff and ascent). The External Tank and Liquid Rocket Booster design will include destruct systems, properly located, which provide for a minimum explosive yield of the liquid propellants when a catastrophic event is in process or when life or property on the ground is at risk. Every effort will be made to save the payload and crew from adverse accident environments.

4.1.9 Earth Parking Orbits

Intermediate Earth parking orbits shall be incorporated into the flight profiles of space transportation systems to allow a minimum of 6 months before orbital decay of the nuclear waste payload package could occur.

sufficient period for waste payload rescue (18.)

ORIGINAL PAGE IS
OF POOR QUALITY

Orbit Transfer Systems. (19.)

4.1.10 ~~Upper Stages~~

The Orbit transfer System
~~All upper stages~~ will have a combined mission delivery reliability greater than 0.99 Achievement of delivery is defined as starting in the proper Earth parking orbit and ending within the bounds of the following: 0.85 ± .01 AU and 1.00 ± 0.20 degrees inclination.

TBD

(20.)

4.1.11 Space Destination

The nominal space destination solar orbit at 0.85 AU, 1° from the Earth's orbital plane, will be verified by proper analysis to provide an expected isolation time of at least one million years.

4.2 Safety Aspects of Mission Phases

Safety aspects for the Reference Concept during the various mission phases have been developed over several years of study. The philosophies presented here are important to the future safe development of the space disposal concepts. Safety aspects peculiar to accidents and malfunction contingency plans for the general phases of the space disposal mission are listed and addressed below:

- Ground transportation from the payload fabrication sites to the launch site
- Preflight operations prior to ignition of the Shuttle's engines
- Launch operations from the launch pad to achieving parking orbit
- Orbital operations.

4.2.1 Ground Transportation

Ground transport (via rail) of the shipping cask would be assigned to the U.S. Department of Energy (DOE), which would supply the necessary accident recovery plans and systems, as needed. Two types of incidents that must be considered are: loss of auxiliary cooling to the waste container, and possible breach of the waste container with a loss of radioactive material. In case of cooling loss, adequate provisions should be made to have self-contained, auxiliary cooling units available within reasonable time. Monitoring equipment for both container temperature and radiation will be required during all ground transport operations. A continuous capability to cope with a container breach will be necessary. A specially trained accident recovery crew will always be ready to act, if necessary.

4.2.2 Preflight Operations

Contingency plans should be provided for potential malfunctions and accidents that could occur while waste payload packages are in the Nuclear Payload Preparation Facility (NPPF), being transported to the launch pad, being transferred from the pad Payload Changeout Room (PCR) to the Uprated Space

ORIGINAL PAGE IS
OF POOR QUALITY

Systems and procedures, in addition to some of those mentioned above, which would minimize the hazard caused by subsystem failures during the boost phase are:

- Intact aborts can be implemented after a few seconds into the flight. Three types of intact aborts are possible for the Uprated Space Shuttle. These are: the return-to-launch-site (RTLS), abort-once-around (AOA) and abort-to-orbit (ATO).
- Contingency aborts could lead to either a return-to-land (runway or crash land) or to a ditching at sea.
- Design of the boost trajectory to avoid land overflight, for example the 38° inclination orbit, should help in reducing overall risk for the early portion of the flight.

4.2.4 Orbital Operations

The orbit transfer vehicle (OTV) propulsion phase provides for transportation from low Earth orbit to the intermediate destination. In the initial years* of the disposal mission the OTV would be a high-thrust, chemical propulsion (liquid hydrogen/liquid oxygen) stage. To minimize possible failures the following systems, procedures and design guidelines are envisioned:

- The use of command OTV engine shutdown in the event of a grossly inaccurate propulsive burn
- The capability to separate the Solar Orbit Insertion Stage (SOIS) and attached payload from the OTV and the use of the SOIS to place the payload in a safe orbit for eventual recovery by a rescue vehicle or Shuttle Orbiter
- A jettison system incorporated into the SOIS payload adapter to separate the waste payload from the OTV/SOIS configuration when necessary to preclude a possible reentry (21.)
- The use of a rescue vehicle to retrieve a waste payload stranded in any given orbit (22.) effective
- The use of redundant systems where ~~feasible~~ to ensure high reliability

*Later on, low-thrust technology (e.g., solar electric propulsion using argon propellant) might be used. With low-thrust systems, both the probability of reentry and magnitude of an explosion are decreased. In addition, there is a much longer decision and response time available in case of a malfunction of the low-thrust propulsion systems while in low Earth orbit. However, because of the large solar arrays needed, the probability of solar array damage caused by an impact with on-orbit, man-made debris could become significant in the future.

ORIGINAL PAGE IS
OF POOR QUALITY

- On-orbit OTV launch crew to obtain instantaneous visual and telemetric status of the OTV propulsive burn (from the Orbiter)

- The proper design of trajectories and propulsive burns of the OTV to reduce the probability for reentry, if a failure occurs

- A waste form which helps insure intact reentry and recovery of the payload, should an unplanned reentry occur and the requirement that the waste payload will not melt after self-burial in low conductivity soil

- The use of thermal protection material on the outside of the package to reduce the risk of atmospheric dispersal on the ground and in the air, as well as an outer steel shield to protect the reentry material in the case of explosion

- The use of a relatively high melting point container and shield material (stainless steel) to reduce the risk of atmospheric disposal of waste. 23

The Solar Orbit Insertion Stage (SOIS) provides for transportation from an intermediate to the final destination. For the Reference Concept, the SOIS is used to reduce the aphelion from 1.0 to 0.85 AU. Systems, procedures and design requirements envisioned to minimize hazards due to SOIS failures are:

- The use of a rescue vehicle to retrieve a cooperative or noncooperative payload stranded in any orbit in heliocentric or Earth orbital space

22

- The use of redundant SOIS systems where effective ~~feasible~~ to ensure high reliability

- The proper use of trajectories and orbits inclined to the Earth's orbital plane that exhibit long-term orbital stability

- The use of long lived tracking systems on board the SOIS to aid in deep space rescue operations. 24

4.2.5 Rescue Operations

Provisions must be made to rescue the SOIS and the nuclear waste payload in Earth orbit in the event of a failure of the OTV during the Earth escape burn. The approach is to rendezvous and dock the rescue OTV with the SOIS and continue the mission from the failed orbit. The rescue mission is based on the premise that, with proper control of the OTV launch, any failure of the OTV will result in an elliptic orbit about Earth. The mission profile for payload rescue is to deliver a rescue OTV to low Earth orbit, transfer by a burn of the OTV to a phase-adjust orbit, and transfer from the phase-adjust orbit at the proper time for rendezvous and docking with the failed system. The lifetime of the rescue OTV, considering the coast time in the phase-adjust orbit, must be as much as 300 hours, compared to the 50 hours for OTV lifetime on the nominal reference mission.

After injection into deep space, the nuclear waste payload could fail to achieve its stable destination orbit, because of a premature shutdown of the OTV engine beyond Earth-escape conditions or a failure of the SOIS to ignite at solar orbit conditions. Studies that address the probability of Earth reentry under these failure conditions have recommended the use of a deep space rescue mission capability as a way of further reducing the overall risk during this phase of the mission (Rice, E. E., and coworkers, 1980b). A deep space rescue mission capability is defined as the ability to send another propulsion system (e.g., OTV and SOIS) to rendezvous with the failed payload in solar orbit and to place it into the desired stable orbit (circular 0.85 AU solar orbit). A capability for uncooperative payload rescue will also need to be provided.

may. 23

5.0 DEFINITION OF TERMS

The following terms are defined in the context of the safety guidelines as used in this document:

Ablation Shield - a layer of protective package material attached to the outside surface of the payload. It is designed to reduce the heating effects during inadvertent atmospheric reentry.

Accident Conditions - as contrasted to normal conditions, are low in probability and high in severity. The corresponding philosophy for the containment barrier is to survive accidents with low consequences rather than remain in an operable state.

ALARA - less than maximum allowable and as low as reasonably achievable. Federal regulations require this principle to be used in most nuclear technology license applications.

Barrier - any medium or mechanism by which either release of encapsulated radioactive waste material is retarded significantly or human access is restricted. Examples of barriers are: waste form, primary container, and isolation.

Containment - a condition in which a hazardous material is isolated from the environment to an acceptable degree.

Criticality - a measure of the capability of sustaining a nuclear chain reaction in a package containing fissile materials.

Decomposition - any significant change in physical or chemical properties resulting in a reduction in mechanical strength, etc.

DOT - Department of Transportation; regarding handling of nuclear materials, Title 49 of the Code of Federal Regulations, Parts 173.389-173.399.

Enclosure "B"

Summary of rationale for recommended changes to System Safety Guidelines Document.

1. Change table of contents to conform to recommended changes in section titles.

2. Add waste form support structure as a generic element of the waste payload or containment system.

3. Add statements to indicate that the components of the containment system may be partially or totally integrated (as in the current reference system, where primary container, radiation shield, impact absorbtion and ablation shield functions are performed by the composite steel/graphite shield).

4. Add functional description of the waste form support structure. Perhaps specific thermal, mechanical and chemical guidelines for the waste payload support structure should be added to Tables 4, 5 and 6. In general they should be identical to the guidelines for the primary container.

5. Percent of yield should be specified. 100% is normal for this kind of spec.

6. Safety factor on yield strength must be applied if 100% of yield in normal operation is specified. An alternate approach would be to specify a mechanical limit of 75% of yield under normal conditions.

7. Ablators usually do not melt. Temperature limits should be specified in terms of minimum temperature for ablation.

8. Add STS crash condition to launch and ascent accident environment description (see Appendix A).

9. The suggested revisions in enclosure "A" incorporate the assembly sequence that we have developed in association with the Boeing manufacturing organization. This assembly sequence will be documented in our reference concept description for our reference waste payload configuration.

10. With the 5 kw limit on thermal dissipation, the reference waste payload does not require active cooling if vents for free convection of air are provided in the cask system. This approach is superior to active cooling, lowers cost and risk and simplifies handling by eliminating the requirements for connect and disconnect of coolant lines.

11. New art and captions are enclosed as part of enclosure "A". They are identical to the configurations in our reference concept description, and will bring the safety guidelines document into conformance.

12. Incorporate description of the waste form support structure, and provide conformance to the reference waste payload concept.

13. Reflect the integrated nature of the primary container and flight radiation shield in the reference concept, and bring terminology into conformance with the reference waste payload description.

14. The primary container in the reference concept is <u>not</u> stainless steel. Even steel may be superseded by a super-alloy such as inconel or hastelloy. Section 4.1.5 should reflect the correct material.

15. Add description of reentry system provisions to Section 4.1.5 to reflect integrated nature of radiation shield/containment/reentry and thermal protection in reference waste payload concept.

16. Title change and additions to paragraph. Brings title into conformance with reference concept description in preparation at Boeing and reflects the comprehensive (not just reentry) protection function provided by the space shuttle Orbiter. Note that reentry system description is moved to 4.1.5 to reflect integrated nature of steel/graphite shield (provides containment, impact absorbtion, radiation shield and reentry protection factors).

17. Delete in accordance with number 15.

18. Change to reflect true requirement - more specifically six months is too general.

19. Change terminology to better reflect actual system (which includes elements like F.S.S. which are not upper stages) and to conform to the reference concept description.

20. Actual value is lower than 0.99. See enclosure C comments on paragraph 4.1.10.

21. Delete jettison system. Waste payload is much better off with SOIS which has beacons and docking provisions, Inadvertant reentry by SOIS is prevented by system fail-safe design. SOIS in effect becomes "lifeboat" for waste payload.

22. use of redundant systems is "feasible" far beyond the point where it is effective in reducing overall risk. Here, use of effective better reflects the intention of the specification. Perhaps "effective in reducing system risk" should be substituted for "effective to ensure high reliability".

23. A specific material callout is not necessary; stainless steel is incorrect in any case.

24. Tracking systems must be long lived to be useful for deep space rescue.

25. Uncooperative payload rescue may not be required to achieve acceptably low levels of long term risk. A study of this issue is definitely needed.

Enclosure "C": Conformance to System Safety Guidelines

Conformance to system guidelines for major elements of the reference concept for space disposal are described below.

4.1 Safety Aspects of Elements:

4.1.1 Waste Form:

Complete Conformance - Waste form maximum temperature under nominal conditions is 700°C (< 1050°C) conformance for accident condition limits is TBD.

4.1.2 Waste Processing and Payload Fabrication Facilities:

N/A to space system: compliance is DOE function.

4.1.3 Shipping Casks and Ground Transport Vehicles:

N/A to space system: compliance is DOE function.

4.1.4 Waste Form Support Structure:

Maximum temperature of the waste payload support structure is about 697°C - this is about 49% of the absolute melt temperature (1422°C). It would be a good idea to raise the spec limit to 50% of the melt absolute temperature, if the 40% was an arbitrary choice, or alternatively to pick an alloy with a melting temperature of 1742°C.

Actually, the 40% of melt temperature limit is only exceeded within a radius of about 30 cm from the core center. The core surface temperature is 316°C. Conformance for accident conditions is TBD.

4.1.5 Primary Container and Flight Radiation Shielding:

Complete Conformance - Shield temperature (at the inner wall under nominal conditions is calculated at 134°C (well under the spec temperature of 416°C.

Temperatures under accident conditions remain to be evaluated. Six hardened recovery beacons have been added to the waste payload at a mass increase estimated at 30 kg per waste payload.

4.1.6 Waste Payload Protection System:

Complete Conformance - An Orbiter flotation system has been added to the flight support system. The system is an inflatable ruggedized nylon bladder inflated by a gas generator which when inflated occupies empty space in the cargo bay, forward of the F.S.S. It is secured to the Orbiter payload bay longerons and the flight support system using steel cables, and will support a combined Orbiter and payload mass of 113,400 kg in a nose-up attitude. The Orbiter already carries location beacons and will have a large radar cross section in the floated condition. The mass of the flotation system is being estimated.

4.1.7 Launch Site Facilities:

Complete Conformance - Complete conformance will be provided by design.

4.1.8 Uprated Space Shuttle Launch Vehicle:

Conformance TBD based on actual STS reliability numbers; suggest that 0.999 be adopted as a goal for the present.

4.1.9 Earth Parking Orbits:

Complete Conformance - With recommended change in wording as specified in enclosure "B".

4.1.10 Orbit Transfer System

Suggest charging 0.99 to TBD pending definition of optimum value. Actual value for existing reference system is still being determined, but will be less than the cumulative injection stage reliability of 0.98610. Note that for risk estimation purposes the system is fail safe with a probability of 0.999999. This is probably the number that should be specified to ensure low risk.

4.1.11 Complete Conformance

4.2 SAFETY ASPECTS OF MISSION PHASES

4.2.1 Ground Transportation:
N/A to space system: DOE function

4.2.2 Preflight Operations:
Complete conformance will be provided by design.

4.2.3 Launch Operations:
Complete conformance will be provided by design.

4.2.4 Orbital Operations:
Complete Conformance - With recommended change to wording as specified in enclosure "B".

4.2.5 Rescue Operations:
Complete Conformance - With recommended change to wording as specified in enclosure "B".

END

DATE

FILMED

JUN 17 1982

End of Document