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Light-Weight Radioisotope Heater Unit Safety Analysis Report (LWRHU-SAR)

Volume I

A. Introduction and Executive Summary
B. Reference Design Document (RDD)

Ernest W. Johnson

October 1985

MOUND
Miamisburg, Ohio 45342

operated by
MONSANTO RESEARCH CORPORATION
a subsidiary of Monsanto Company

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for the
U. S. DEPARTMENT OF ENERGY
Contract No. DE-AC04-76-DP00053

DISCLAIMER

MLM--3293-Vol.1

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Title Per Sue Brummitt E. Blue

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ABSTRACT

The orbiter and probe portions of the NASA Galileo spacecraft contain components which require auxiliary heat during the mission. To meet these needs, the Department of Energy's (DOE's) Office of Special Nuclear Projects (OSNP) has sponsored the design, fabrication, and testing of a one-watt encapsulated plutonium dioxide-fueled thermal heater named the Light-Weight Radioisotope Heater Unit (LWRHU). This report, prepared by Monsanto Research Corporation (MRC), addresses the radiological risks which might be encountered by people both at the launch area and worldwide should postulate mission failures or malfunctions occur, which would result in the release of the LWRHUs to the environment. Included are data from the design, mission descriptions, postulated accidents with their consequences, test data, and the derived source terms and personnel exposures for the various events.

A. INTRODUCTION AND EXECUTIVE SUMMARY

1.0 INTRODUCTION

The Light-Weight Radioisotope Heater Unit (LWRHU)* is planned to be used on the NASA Galileo Mission (GLL) to provide thermal energy to the various systems on the orbiter and probe, which are adversely affected by the low temperature a spacecraft encounters during an interplanetary Mission such as the GLL to Jupiter. Using these plutonia-fueled sources in 1-W increments permits employment of a single design and provides the spacecraft user the option of how many to use and where to position them to satisfy the proper thermal environment for components requiring such consideration. The use of the radioisotope ^{238}Pu in these devices necessitates the assessment of postulated radiological risks which might be experienced in case of accidents or malfunctions of the shuttle (space transportation system or STS) or GLL spacecraft during phases of the mission in the vicinity of the earth.

1.1 Purpose of SAR

This Safety Analysis Report is to present the analyses and results of the evaluation of the nuclear safety properties of the 101 LWRHUs to be employed on the Galileo Mission. The preliminary status of this activity was presented earlier to the Interagency Nuclear Safety Review Panel (INSRP) in a meeting in July, 1984 (Reference 1). The philosophy for issuing a single Safety Analysis Report (SAR) for small heaters is that the heaters are considered an inherently less complex and more compact system; therefore, as in the case of the SAR for the Voyager 1-W heaters (Reference 2), only a single final SAR is issued.** In addition, the analyses performed by the radioisotope thermoelectric generator (RTG) systems contractor on accident scenarios, probabilities, etc. were used to avoid duplication of effort. Also used were the comments provided by the Interagency Nuclear Safety Review Panel (INSRP) and the subpanel experts on documents such as the RTG Preliminary SAR (PSAR) and Updated SAR (USAR). Therefore, considerable use was made of the General Electric USAR (Reference 3) and amendments for the FSAR (Final SAR) for the General Purpose Heat Source (GPHS) RTGs.

1.2 Document Organization

The format of this document conforms to the requirements defined in the Overall Safety Manual (Reference 4). Details of this format are provided in Table A-1 (References 5 and 6).

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*All abbreviations and acronyms are defined in Appendix B-2.

**It should be pointed out also that the two NASA GLL RTGs contain about 8800 W of plutonium, whereas the LWRHUs contain slightly over 100 W of this radioactive material.

TABLE A-1: The LWRHU-SAR Format Defined

VOLUME I

A. INTRODUCTION AND EXECUTIVE SUMMARY

1.0 Introduction

- Purpose of SAR
- Document Organization
- Data Acknowledgements*

2.0 Missions and Flight Systems Summary

- Nuclear System
- Mission Description
- Space Shuttle/Centaur
- Mission Phase Definition

3.0 Accident Summary

- Flowchart of Accidents and Probabilities by Phase
- Tabulation of Accidents/Consequences by Phase
  - Accidents
  - Fuel Release Mechanisms
  - Locations Affected
- Source Terms
  - Expected - Most Probable
  - Worst Case
    - Causative conditions
  - Probabilities

- Summary Discussions of Significant Accidents

B. REFERENCE DESIGN DOCUMENT (RDD)

1.0 Nuclear Power System
2.0 Spacecraft
3.0 Launch Vehicle
4.0 Centaur
5.0 Trajectory and Flight Characteristics
6.0 Launch Site

*Data sources are provided in the References, Appendix B-3, Page 118.

TABLE A-1 (Continued)

VOLUME II: ACCIDENT MODEL DOCUMENT (AMD)

1.0 Introduction

Purpose of AMD
Document Organization
Data Acknowledgements

1.1 Missions and Systems Description

- Mission Phases with Timelines/Events
- LWRHU Design

2.0 Summary of Accident Evaluation

Flowchart
Centaur Domination of Accident Environments Described
Tabulation of Accidents/Consequences by Phase

3.0 Accident Evaluation and Failure Mode Analysis

3.1 Objectives and Approach
Sequence Tree Construction
Mission Accident Evaluation

- Each Phase Addressed Separately
- Each Accident in Each Phase Treated Separately in Entirety

3.3.1 Phase 0-Prelaunch

- o Top-Level Event Tree Showing Accidents with Probabilities
- o Accidents Defined and Characterized from Initiation to Identification of Source Terms Where Applicable

  - Accident Description
  - Accident Environment Definition
  - Initial LWRHU Response to Explosions
  - Synergistic Effects of Environments
  - Intermediate Events Identified
  - Final Disposition of LWRHU
  - Source Term Characterized

Table A-1 (Continued)

LWRHU SAR NRAD (VOLUME III) OUTLINE

1.0 EXECUTIVE SUMMARY

2.0 MAJOR ACCIDENT SCENARIOS AND CONSEQUENCES

Launch
Ascent
Reentry
Mission Aborts to Landing

3.0 INTEGRATED MISSION RISK ANALYSIS

4.0 RELATED ASPECTS OF RADIOLOGICAL IMPACT

Effect of Emergency Countermeasures
Effect of Meterological Launch Restraints
Land Contamination
Water Contamination

APPENDICES

A. KSC Meteorology

B. Population Distribution

Night Launch Spectators*
Occupational - KSC
Resident Population

C. Radiological Assessment Methodology

Source Term Characterization
Plume Rise/Cloud Size
Dispersion Analyses
Atmospheric
Ocean
Radiation Dose Models

D. Uncertainty Analysis

E. Biomedical Considerations of $Pu^{238}O_2$

F. Parametric Radiological Analyses

*Actually day launch

TABLE A-1 (Continued)

All Inputs Positively Identified and Referenced to Appendices

Minimal Analysis and/or Testing to Be Shown or Discussed

- Results to Be Presented
- Reference to Details in Appendices

Repetitious Material Will Be Included Where Applicable for Each Accident for Purposes of Completeness and Ease of Readibility and Understanding

o Complete Phase Event Tree

- Presented After All Accidents Are Defined and Characterized
- Accidents and Events Identified by Designators Referenced in Each Accident Evaluation

3.2.2 Phase I - Above Sequence and Content Repeated for Each Subsequent Phase

APPENDICES: Include all test and analytical data used (and its justification) for all accidents, environments, and LWRHU response modes addressed in the accident evaluations.

A. Accident Definition
B. Accident Environments
C. LWRHU Response to Explosions
D. LWRHU Response to Fragments/Projectiles
E. LWRHU Response to Propellant Fires
F. Spacecraft Reentry Breakup Analysis
G. LWRHU Reentry Response
H. Impact Test Program Results
I. Burial Thermal Analysis
J. Source Term Evaluation

2.0 MISSION AND FLIGHT SYSTEM SUMMARY

2.1 NUCLEAR SYSTEM (LWRHU)

The LWRHU is a radioisotope-fueled system consisting of a 1-W pellet of plutonium-238 dioxide, a clad of platinum-30% rhodium (Pt-30Rh), an insulation system of Pyrolytic Graphite (PG), and an aeroshell/impact body of Fine-Weave Pierced Fabric (FWPF).* The exterior dimensions are 26 mm diameter by 32 mm long; each LWRHU weighs about 40 g. The detailed description of the LWRHU is in Section B., RDD. A cutaway drawing of the LWRHU is shown in Figure A-I.

2.2 Galileo Mission Description

The NASA Galileo spacecraft is scheduled for launching during May/June, 1986, by the Space Shuttle to attain a temporary Earth parking orbit; the Centaur G-prime vehicle will be used to propel the spacecraft from the parking orbit into the escape trajectory toward Jupiter. The spacecraft should arrive in the vicinity of Jupiter after an interplanetary transit of a little more than two years. Plans are for a probe to be deployed from the spacecraft 150 days before arrival at Jupiter and to descend into the Jovian atmosphere. The scientific objectives of the mission are to conduct comprehensive investigations of the Jovian system, including the planet, its satellites, and the vast expanse of the Jovian magnetosphere.

2.3 Space Transportation System (Shuttle)

The Galileo spacecraft is scheduled for launching from Pad B of Launch Complex 39 at the Kennedy Space Center (KSC), Cape Canaveral, Florida. The current launch configuration is shown in Figure A-2, with an enlarged view of the spacecraft on the Centaur G-prime vehicle shown in Figure A-3. Figure A-4 provides a more detailed drawing of the stowed GLL spacecraft with LWRHU locations.

2.4 Mission Phase Definition

To provide a logical approach to the safety analysis, the reference mission for Galileo is divided into six distinct phases for purposes of the safety analysis. These phases cover all mission-related operations beginning with loading the liquid propellants into the Shuttle External Tank after the spacecraft has been installed in the cargo bay and ending with the attainment of the hyperbolic Earth escape trajectory. At this point, with a successful and correct burn of the Centaur having been accomplished, effective escape of the spacecraft from the Earth's gravitational pull will be effected, and the LWRHUs will no longer present a potential risk to the Earth's populace. Definitions of the phases** are as follows:

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*Manufactured by AVCO Systems Division

**Prelaunch and launch are grouped together in the FASTs as Phase 0.

LIGHTWEIGHT RADIOISOTOPE HEATER UNIT

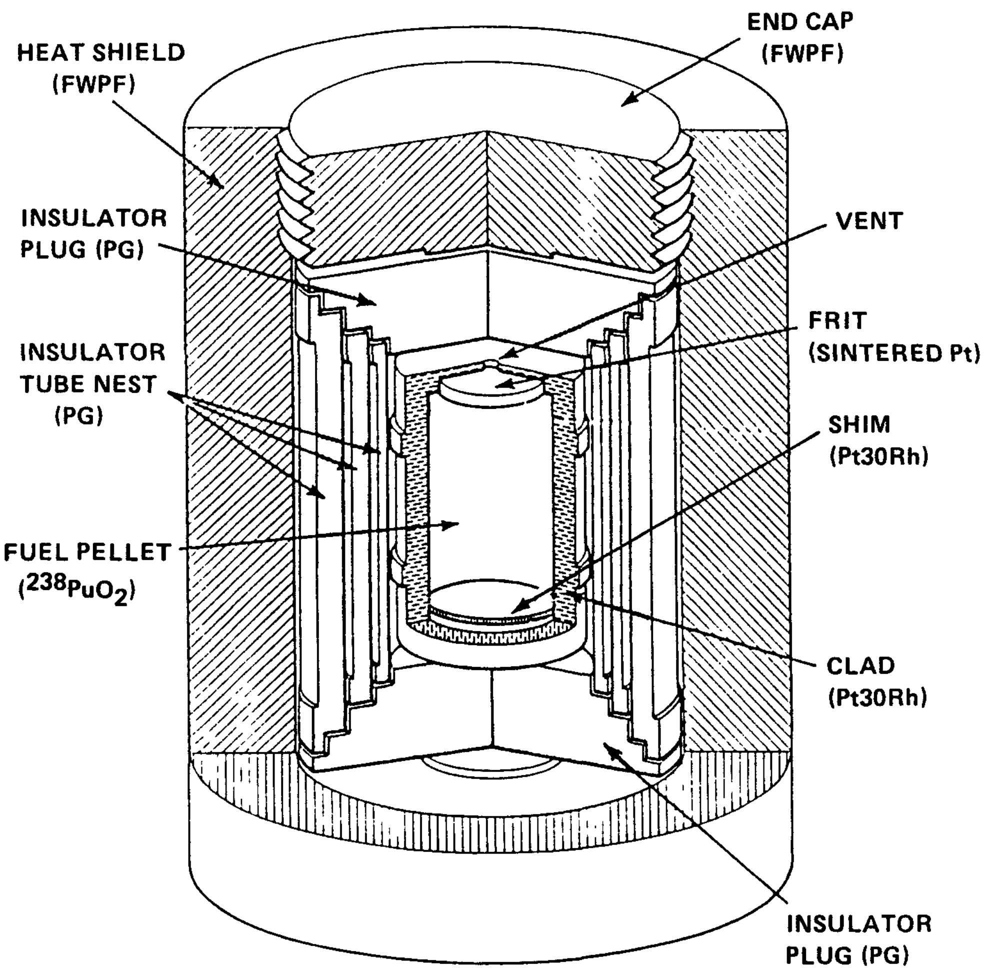

FIGURE A-1: The principal features of the Light-Weight Radioisotope Heater Unit are portrayed above.

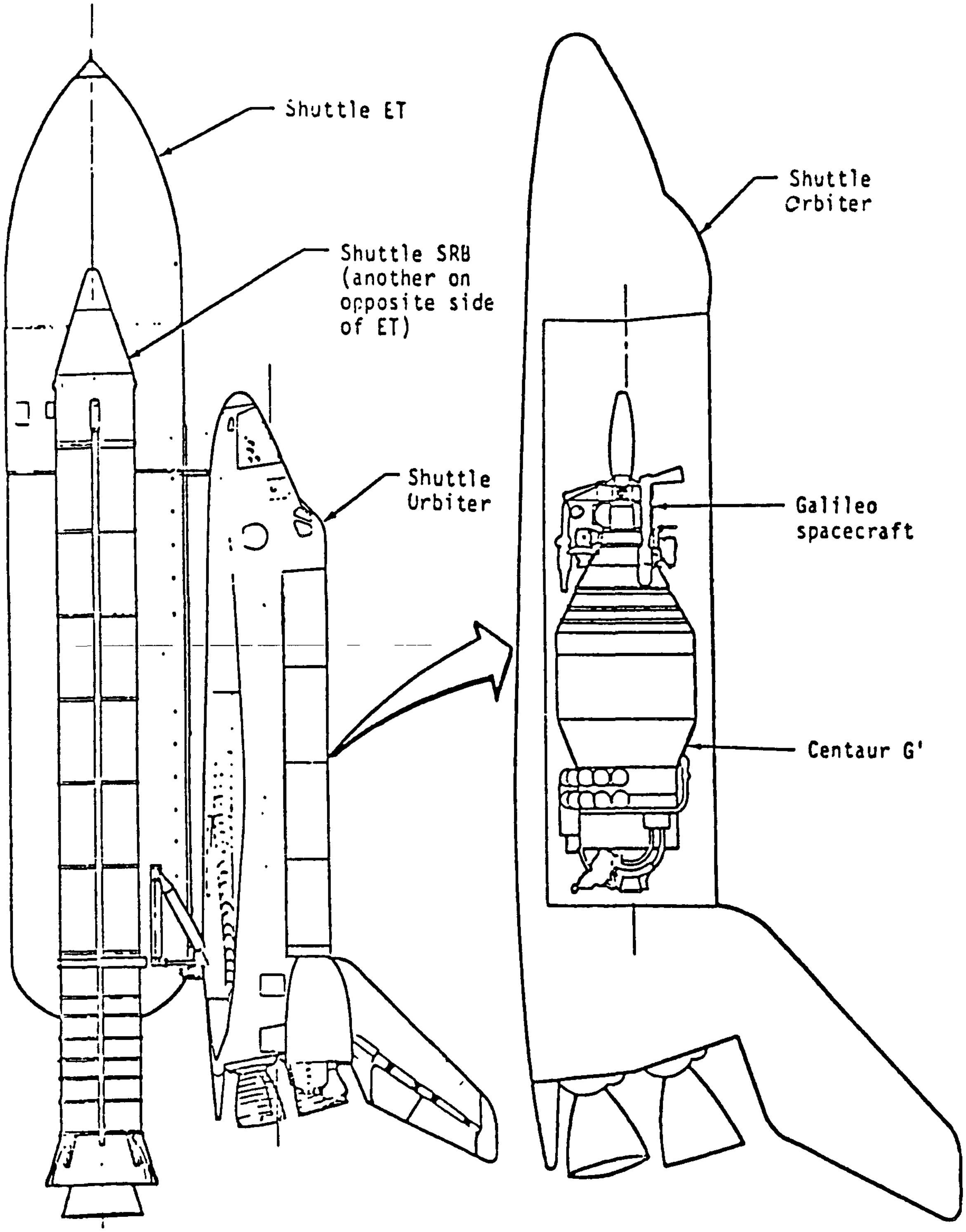

FIGURE A-2: The general spatial relationships between the Galileo spacecraft and the shuttle are shown in this drawing.

PWS = Plasma Wave Subsystem
RTGB = RTG Boom
SCH = Science Cable Hinge
DSE = Despun Electronics

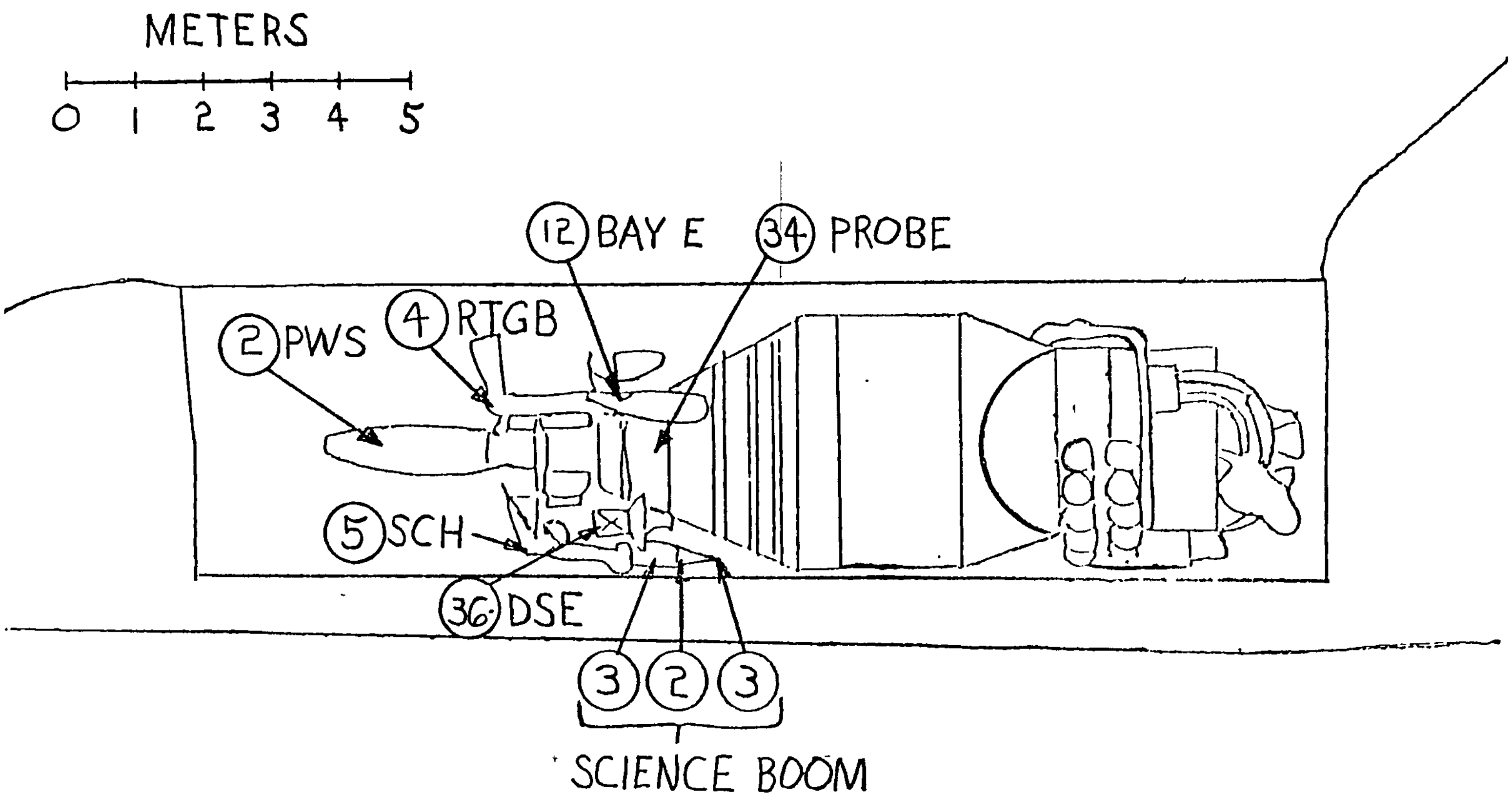

FIGURE A-3: The locations of the 101 LWRHUs on the stowed GLL spacecraft are provided above.

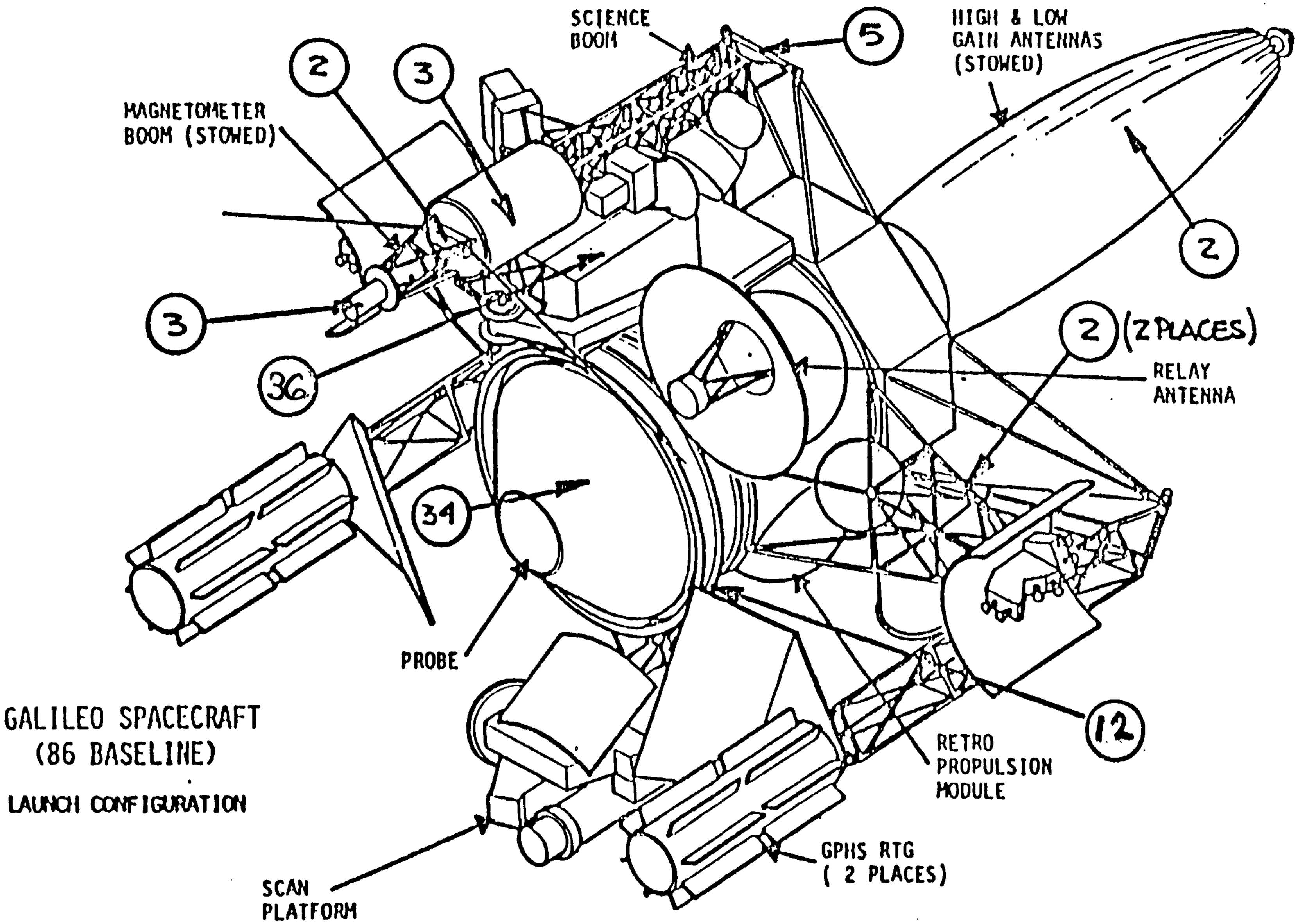

FIGURE A-4: Galileo RHU locations on the stowed spacecraft are shown above.

Phase 0 - Prelaunch - The applicable portions of this phase begin with the initiation of loading the liquid propellants into the external tank (ET) and Centaur and ends with the initiation of the Ground Launch Sequencer. The duration of this portion of Phase 0 is from T-8 hr to T-31 s.

Phase 0 - Launch - This phase begins the automatic launch sequence and continues with the ignition of the SSMEs at T-6.6 s and continues until the solid rocket booster (SRB) ignition command is given at T = 0. It might be noted that there are no fuel releases expected for accidents at T<0.

Phase 1 - First-Stage Ascent - Phase 2 begins at T = 0 s and lasts until the SRBs are jettisoned at T + 128 s.

Phase 2 - Second-Stage Ascent - This phase includes the period from SRB separation (T + 128 s) until main engine cutoff (MECO) at T + 534 s.

Phase 3 - On-Orbit - Phase 4 starts at T + 534 s and includes Centaur deployment from the Orbiter at T + 24,084 s (6 hr, 41 min, 24 s). The significant events that occur during this phase include the first and second burns of the OMS (OMS-1 and OMS-2) for orbit attainment and circularization, the release of the Centaur and spacecraft, and the starting of the Orbiter RCS (Reaction Control System) to move away from the Centaur.

Phase 4 - Insertion - Phase 5 starts at T + 24,084 s and ends when the Centaur has attained Earth escape velocity at T + 27,480 s, even though the spacecraft has not yet been deployed. The Centaur engine ignition occurs at T + 27,084 s (7 hr, 31 min, 24 s).

3.0 ACCIDENT SUMMARY

3.1 Accident

With the sizable quantity of cryogenic propellant contained in the ET and Centaur, there exists the potential for device failure(s) which in turn can result in explosions of varying severity. Table A-2 summarizes accidents which could expose the LWRHUs to environments which could result in the release of contained plutonia. Prelaunch at T-8 hr is the start time for consideration, as this is when the cryogenic propellant loading begins. Centaur CBM explosions can occur through Phase 4, Orbital Insertion (Reference 7). (There may be accidents before and after the time segment encompassed in Table A-2, but these do not result in release of plutonia from the LWRHUs as the environments are not of a sufficient severity to result in clad failure.)

The probabilistic environments generated by the explosions are summarized as follows (from JSC 08116, Reference 8, except for shock wave velocity and LWRHU velocity which are calculated values):

TABLE A-2: Summary of Probabilities for various Cryogenic Explosions is provided below (from JSC 08116, Table 3.5, Reference 8; the column headed Insertion is from TWX, G. J. Schaefer/J. J. Lombardo, July 12, 1985).

	10^{-6} Failure Probability/Mission Phase* (Timeline in Seconds)					
Environment/Accident	Prelaunch -28800 to -31	Launch -31 to 0	1st Stage 0 to 128	2nd Stage 128 to 534	On-Orbit 534 to 24084	Insertion >24084
Maximum Environment	--	--	0.4	--	--	--
ET Bulkhead Curve - ET Failure	--	--	1.7	0.5	--	--
2D Centaur**						
-ET Low Yield	1.4	--	7.8	2.5	--	--
-STS Origin	2.5	6.5	15.7	13.3		
-Centaur Ext.	5	1	13	38		
10 Centaur*** Range Destruct	--	--	9.5	--	--	--
Centaur CBM	12	--	5	14	169	101
FAST Phase (X1)	0	0	1	2	3	4

*The probabilities of STS failure are to be treated as events that could occur at any one time during a defined mission phase. If the event occurs during a subphase, it has happened and should not be repeated (or added in a subsequent subphase). Centaur probabilities are distributed over the mission phase equally. If subphases are defined, the probability for each subphase is equal to: $\frac{\text{Probability X Subphase Time}}{\text{Phase Time}}$.

**ET low-yield event does not add to accident severity but results in a 2D Centaur explosion.

***Includes spill conditions without positive acceleration.

1. Maximum Environment - This is an event caused by the left SRB malfunctioning in the 0-2 s time frame. This results in the maximum environment of 13.8 MPa (2000 psi) shock overpressure with its corresponding large (~52 kPa s or 7.5 psis) static impulse. It also would result in a high-velocity fragment environment. The mission failure probability is 0.4×10^{-6}.

2. ET Bulkhead Failure - Figure A-5 displays the variation of environment with respect to its probability of severity; the total probability of occurrence from Table A-2 is 2.2×10^{-6} over the Centaur mission.

3. Two-Diameter Centaur Explosion (Confined-by Ground Surface, CBGS-2D) Failures - The probabilistic variations of environments for these events are given in Figure A-6. This sort of environment is expected to occur at a mission frequency of 102.2×10^{-6}.

4. One-Diameter Centaur Explosion (CBGS-1D) Failure - Figure A-7 provides the relationship of various environmental parameters as a probability of their occurrence for this rather severe event. Accidents leading to this sort of environmental conditions occur only during the first stage (0 to 128 s) and at a frequency of 9.5×10^{-6} per mission.

5. Centaur Confined-by-Missile (CBM) Failure - The most frequently anticipated accident environment is the Centaur CBM event, which may occur throughout the various phases shown on Table A-2 and totals 301×10^{-6} per mission. The probability of the various environmental parameters being realized is given in Figure A-8.

3.2 Fuel Release Mechanisms

Testing performed upon the LWRHU heat source has demonstrated its ability to withstand the highest shock overpressure regime without clad rupture. It will also undergo the solid rocket fuel fire and cryogenic fuel fireball without significant (<<1 Ci, 37 kBq) release of the plutonia fuel. Reentry and post-reentry impact (including soil burial or water disposed) will not result in significant fuel release from the LWRHU system.

Potential releases of a fraction of the contained inventory could occur if a very high (>775 m/s) velocity fragment hit the stationary LWRHU assembly or a lower velocity (>330 m/s) collision occurred with the bare clad (this assumes the carbon and holder have been lost in an earlier event) and some structure. Note that bare clad fire scenarios are thought to be improbable, as any explosion event of a magnitude great enough to remove the protective carbons would eject the clads (and any undamaged LWRHUs) from the fire environment and the remaining propellants.

The second scenario which can result in the release of plutonia from the LWRHUs is a potential occurrence of an ET explosion in late Phase 2 or the CBM event in late Phase 2 through Phases 3 and 4. Should the explosion overpressure be of sufficient magnitude to remove the protective aeroshell, the now bare clad could be exposed to sufficient aerodynamic heating during its subsequent reentry to melt and release the fuel at high altitudes. A small amount of this now particulate fuel will evaporate but as melting does not occur, most will settle in the earth's surface as discrete particles.

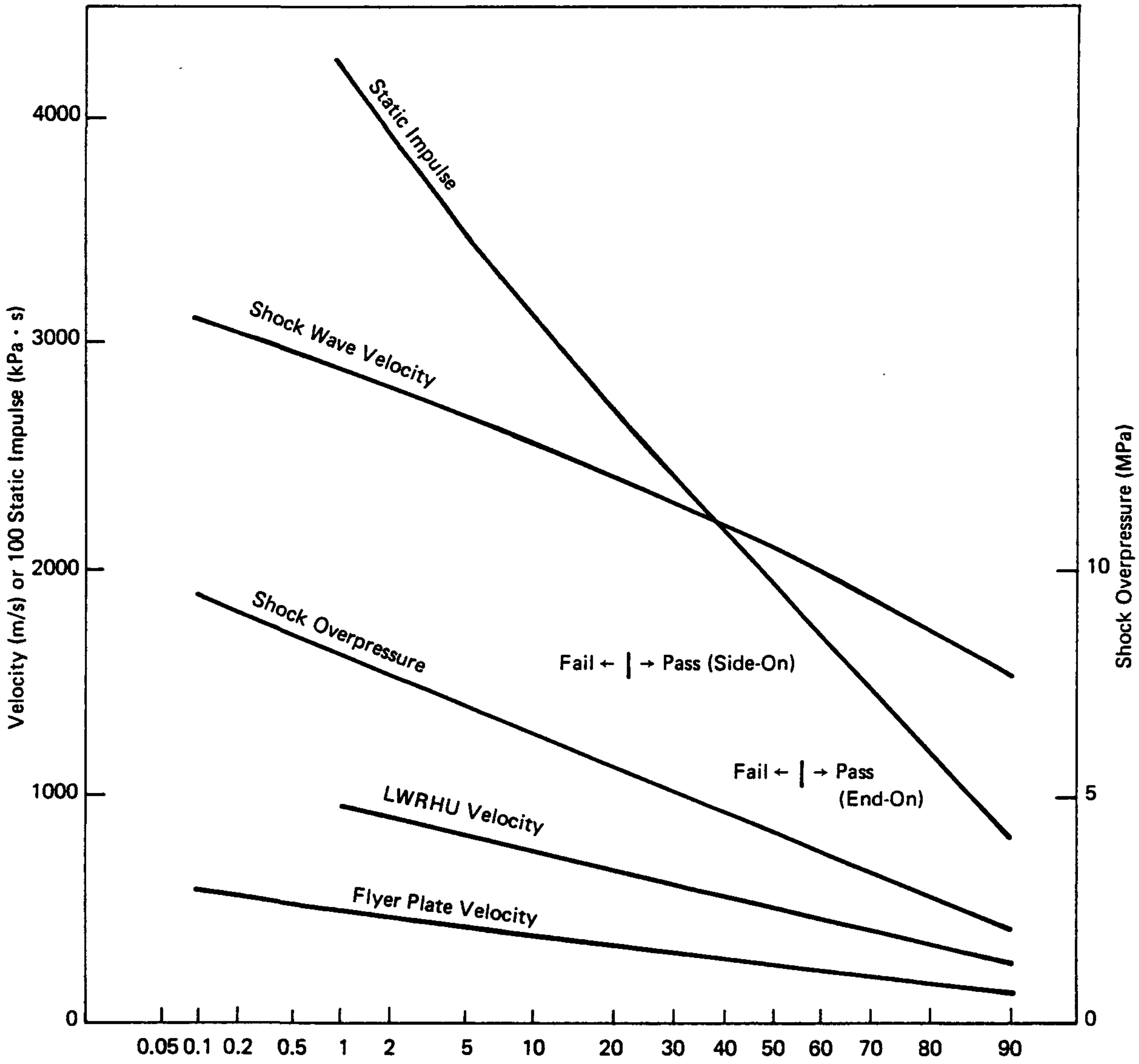

FIGURE A-5: The ET Explosion Mode results in these curves; the abscissa is the percentage of time that variable will exceed value selected.

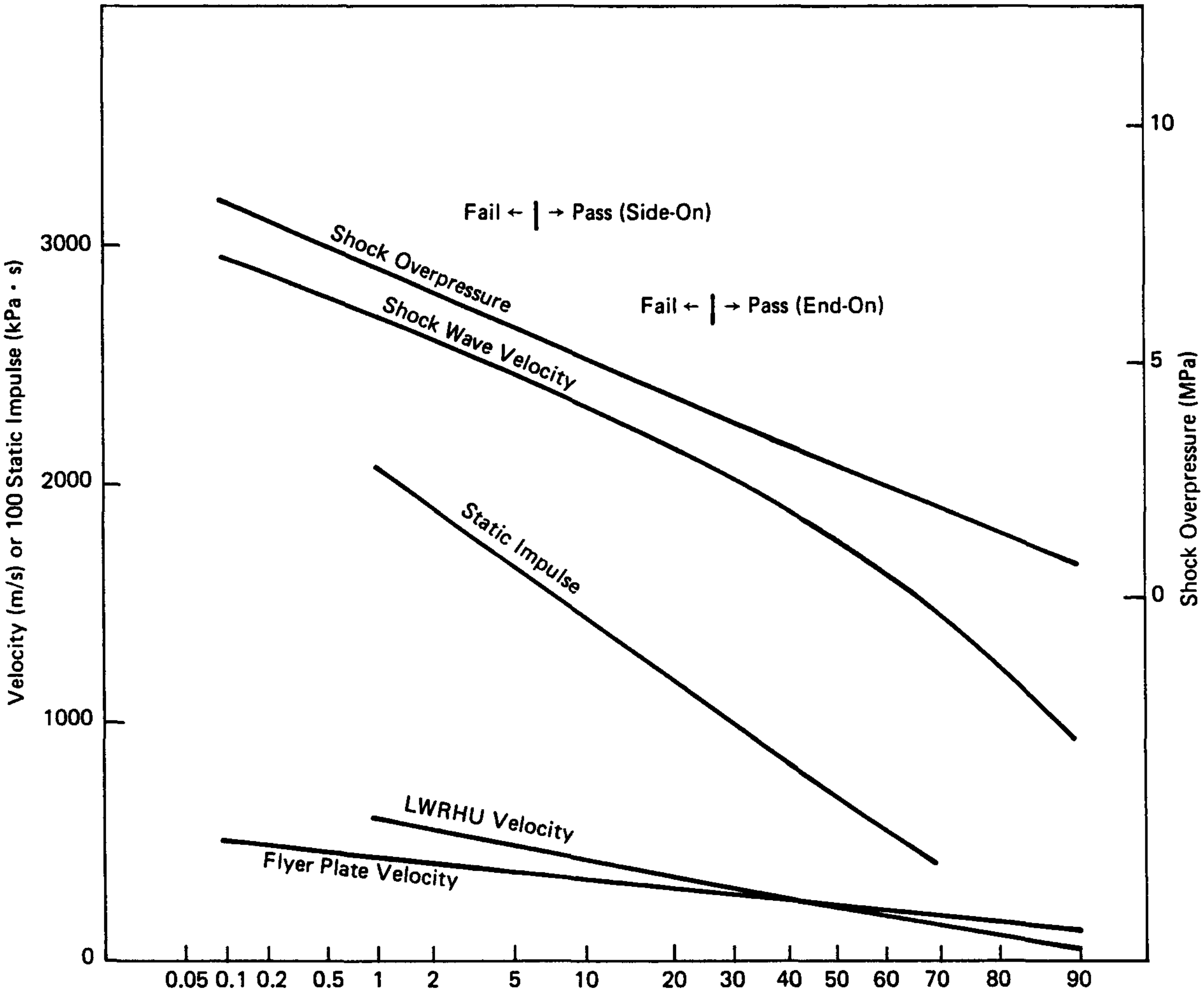

FIGURE A-6: Parameters associated with the Centaur 2D Spill Mode. The absicca is percent probability that the variable will exceed the corresponding value.

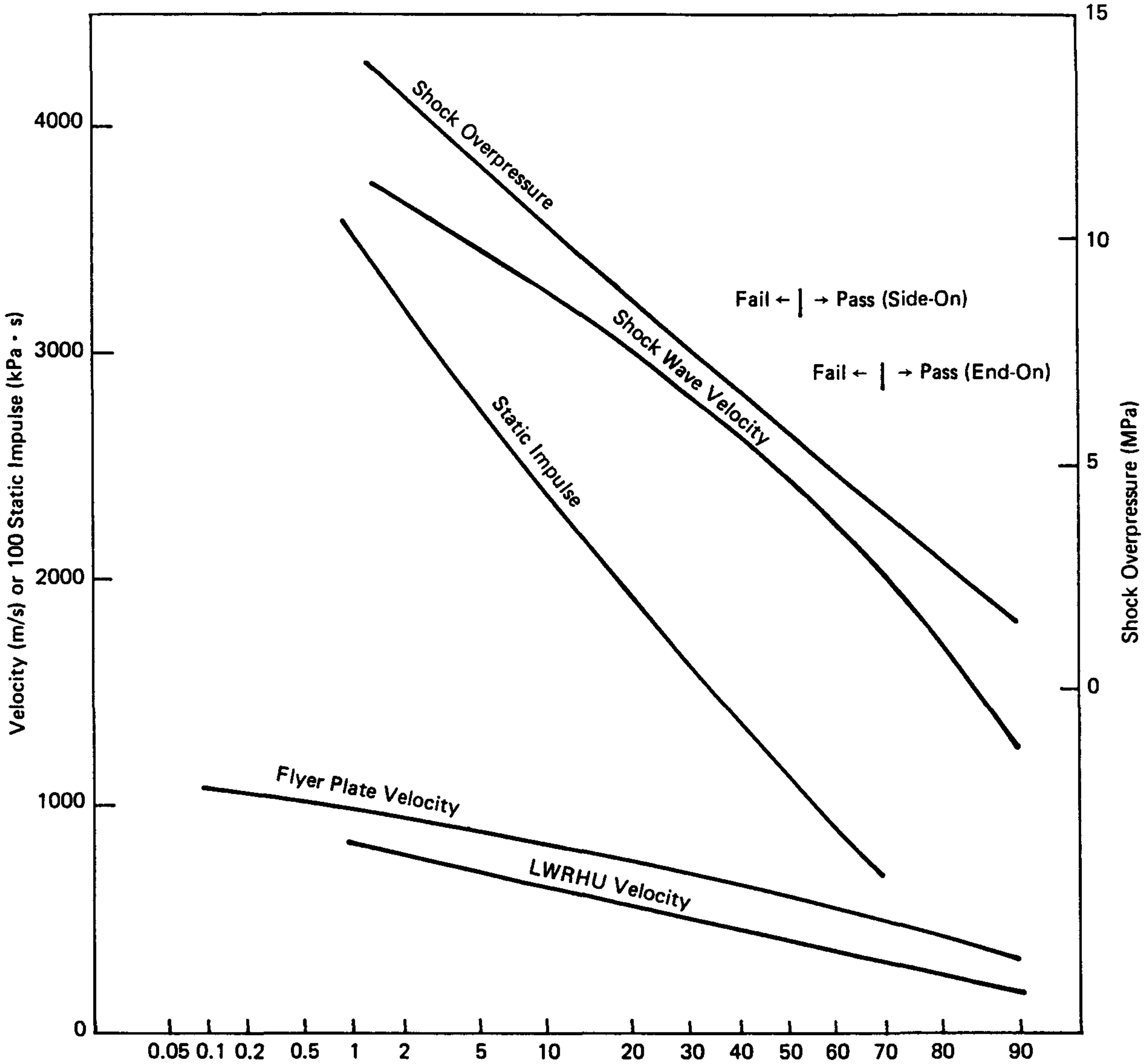

FIGURE A-7: Parameters associated with the Centaur 1D Spill Mode. Probability (in percent) is time that the parameter will exceed that value.

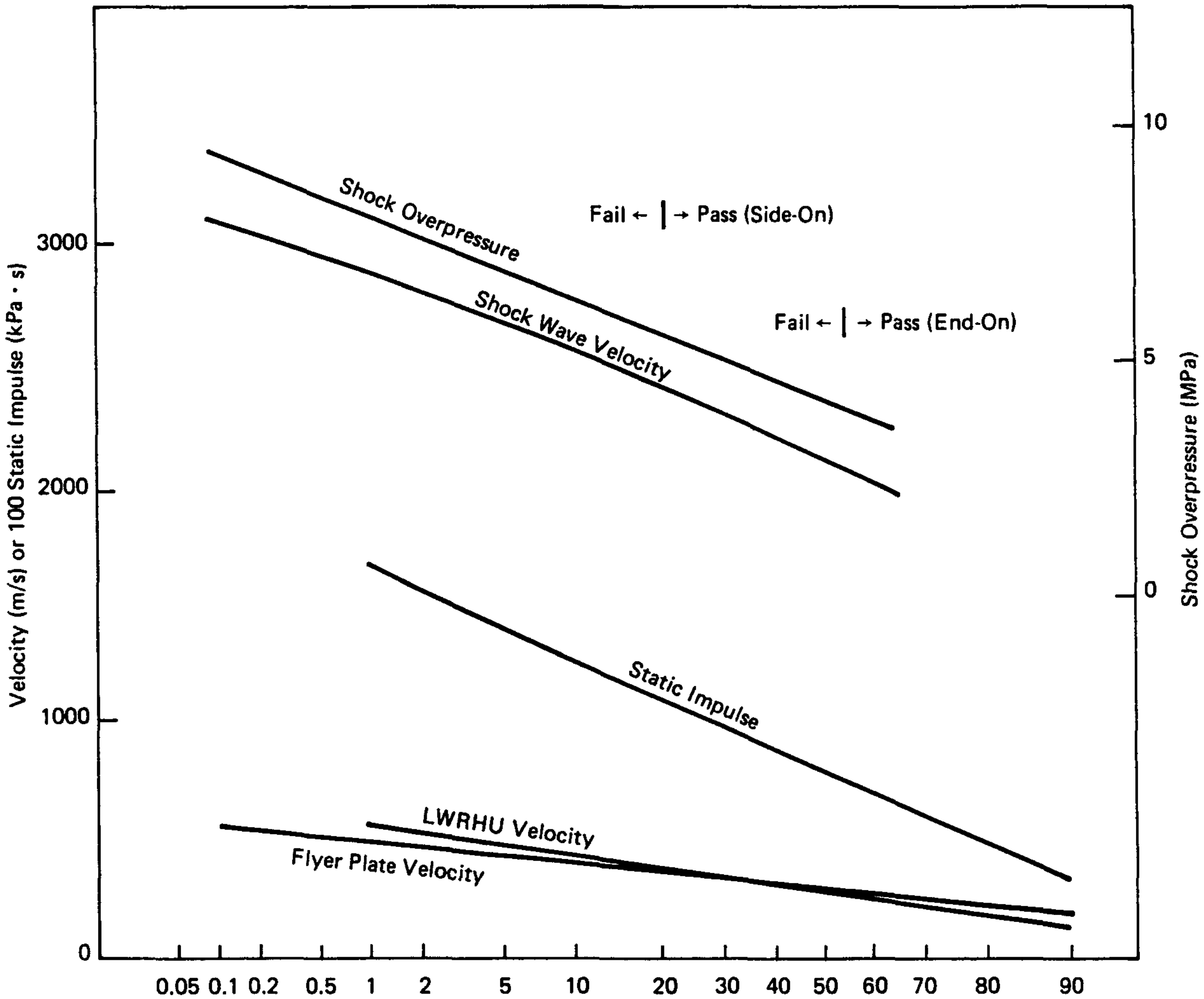

FIGURE A-8: Parameters associated with the Centaur CBM Mode are given as a function of "greater than" probability percentage.

3.3 Locations Affected

Potential release areas as a function of phase are summarized as follows:

1. Prelaunch: Confined to general Pad 39B area.

2. Launch: Confined to general Pad 39B area.

3. 1st Stage: Varies from the vicinity of the launch pad at essentially sea level to 73 km (240,000 ft) altitude and well out (93 km or 50 mi) from the Florida shoreline.

4. 2nd Stage: Events here occur between 73 km and 162 km (490,000 ft) altitude well out into the Atlantic Ocean or over Africa.

5. On-Orbit: Events could occur anywhere in the orbit which varies from 93 X 240 km (58 X 140 mile) to one that is circular (240 km or 130 mile).

6. Aborts: Return to Launch Site (RTLS), Trans-Atlantic Landing (TAL) Abort, Abort Once-Around (AOA), and Abort-to-Orbit (ATO) standard ascent intact aborts are schematically depicted in Figure A-9.

3.4 Source Terms

Volume II, the Accident Model Document or AMD, provides the rationale for arriving at quantities and probabilities of fuel release(s) at various mission phases. Table A-3 summarizes the results of this analysis.

In order to apply this information to the overall risk assessment, Volume III (Nuclear Risk Analysis Document or NRAD) has the potential released quantities identified as:

A. Most probable release case - that FAST sub-branch in a mission phase with a fuel release having the highest probability. This includes any associated concurrent or sequential fuel releases. A most probable case is identified for each mission phase. See Table A-4.

B. Maximum case - that FAST sub-branch in a mission phase with a fuel release having the highest magnitude that maximizes the radiological consequences. A maximum case is identified for each of the mission phases. See Table A-5.

C. Release expectation case - the source term expectation for an event assuming the event has occurred. It is the sum of the products of the individual source terms and their total or mission occurrence probabilities, for all possible releases of a given type which can follow the event, divided by the occurrence probability of the initiating event. Release expectation cases are determined for each mission phase and accident type. See Table A-6.

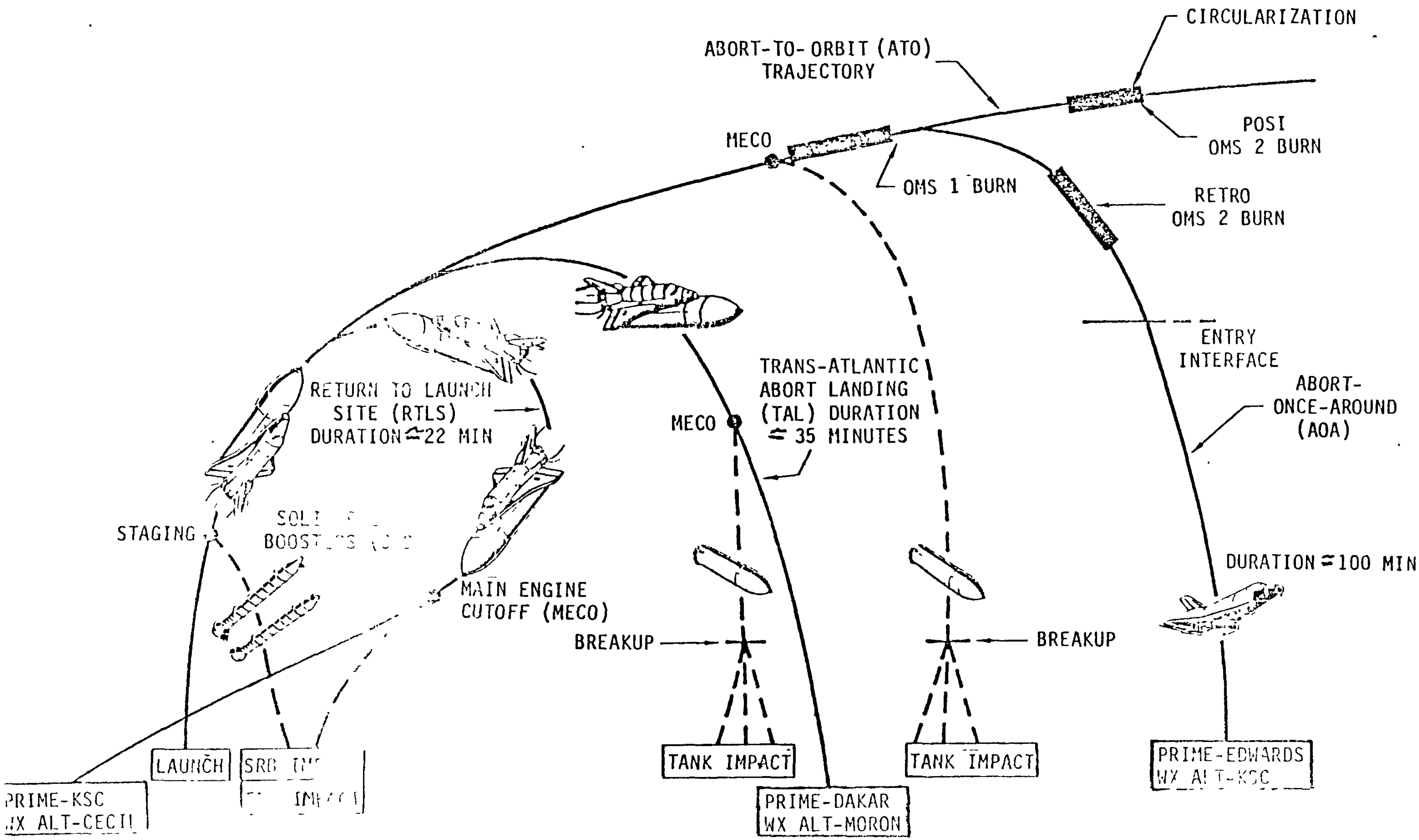

FIGURE A-9: Standard ascent intact aborts are depicted above.

TABLE A-3: SUMMARY OF FUEL RELEASES FOR GALILEO LWRHU

PHASE IDENTIFIER	ACCIDENT ENVIRONMENT	ALTITUDE (m)	QUANTITY VAPOR, Ci (GBq)	QUANTITY PARTICULATE, Ci (TBq)	MAXIMUM PROBABILITY	
1A OBM	Maximum Condition	60	3.6 (135)	92.4 (3.5)	6E-9	Can Occur Together
1A SBPWS		60	2.4 (90)	61.6 (2.3)	5E-9	
1A IBM		60	3.6 (135)	92.4 (3.5)	3E-9	
1A SCH		60	6.1 (225)	153.9 (5.8)	1.1E-9	
1A DSE		60	21.9 (810)	554.1 (20.5)	5E-9	Cannot Occur If Above Happens
1B	ET	30	<1 (<40)	96 (3.6)	8E-9	
1D OBM	CBGS-ID	60-10k	3.6 (135)	92.4 (3.5)	4.3E-8	Can Occur Together
		>10k	∿0	96 (3.6)	4E-8	
1D SBPWS		60-10k	2.4 (90)	61.6 (2.3)	3.2E-8	
		>10k	∿0	64 (2.4)	3E-8	
1D SBPWS		60-10k	3.6 (135)	92.4 (3.5)	3.1E-8	
		>10k	∿0	96 (3.6)	3E-8	
1D SCH		60-10k	6.1 (225)	153.9 (5.8)	6.6E-9	
		>10k	∿0	160 (5.9)	6E-9	
1D DSE		60-10k	21.9 (810)	554.1 (20.5)	1.06E-8	Cannot Occur If Above Happens
		>10k	∿0	576 (21.3)	1E-8	
2A SO+EO	ET	∿100k	0.377 (14)†	2144 (79.3)§	4E-8	
ΔEO		∿100k	0.135 (5)†	768 (28.4)§	6E-8	
2B SO+EO	CBGS-2D	∿100k	0.377 (14)†	2144 (79.3)§	1.2E-6	
ΔEO		∿100k	0.242 (9) †	1376 (50.9)§	3.8E-6	
2C SO+EO	CBM	∿100k	0.377 (14)†	2144 (79.3)§	1.0E-6	
ΔEO		∿100k	0.242 (9) †	1376 (50.9)§	1.6E-6	
3A SO+EO	CBM	120k	0.377 (14)	2144 (79.3)	3.7E-5	
ΔEO		120k	0.242 (9)	1376 (50.9)	5.8E-5	
4C SO+EO	CBM	120k	0.377 (14)	2144 (79.3)	2.2E-5	
ΔEO		120k	0.242 (9)	1376 (50.9)	3.4E-5	

†At End of Phase 2, T=534s
§After T=400s, or After Clad Melt Without Fuel Melt

OBM = Outboard Magnetometer
SBPWS= Science Boom Plasma Wave Subsystem
IBM = Inboard Magnetometer
SCH = Science Cable Hinge
DSE = Despun Electronics
SO = Side-On LWRHU
EO = End-on LWRHU
Ci = Curie (1Ci = 37 GBq)
TBq = Terabecquerel
GBq = Gigabecquerel
ΔEO = EO - SO

Revised: 9/5/85

Table A-4: Most Probable Release Cases for LWRHUs on the Galileo Mission

Mission Phase	Accident Branch	Release Branch	Accident Probability	Conditional Fuel Release Probability	Total Fuel Release Probability	Source Term, Ci Vapor	Source Term, Ci Particulates	Release Location
0		(No Releases)						
1	1D	SRB Impact @ 10 km of OBM RHUs	9.5E-06	4.2E-03	4.0E-08	---	96[b]	10 km above ocean
2	2B	End-on clads	5.36E-05	9.3E-02	5.0E-06	0.242[a]	1376[b]	10 km above ocean
3	3A	End-on clads	1.69E-04	5.6E-01	9.5E-05	0.242[a]	1376[b]	High altitude, 0° lat
4	4C	End-on clads	1.01E-04	5.6E-01	5.7E-05	0.242[a]	1376[b]	High altitude, 0o lat

a = 0.02 μm particle size
b = 50 μm particle size

Table A-5: Maximum Release Cases for LWRHUs on the Galileo Mission

Mission Phase	Accident Branch	Release Branch	Accident Probability	Conditional Fuel Release Probability	Total Fuel Release Probability	Source Term, Ci		Release Location
						Vapor	Particulates	
0		(No Releases)						
1	1D	SRB Impact of DSE RHUs	9.5E-06	1.1E-03	1.0E-08	21.9[a]	554[b]	Pad
2	2B	Side-on clads End-on clads	5.36E-05	2.2E-02	1.2E-06	0.377[a]	2144[b]	10 km above ocean
3	3A	Side-on clads End-on clads	1.69E-04	2.2E-01	3.7E-05	0.377[a]	2144[b]	High altitude, 28° N latitude
4	4C	Side-on clads End-on clads	1.01E-04	2.2E-01	2.2E-05	0.377a	2144[b]	High altitude, 28° N latitude

a = 0.02 μm particle size
b = 50 μm particle size

Table A-6: Release Expectation Cases for LWRHUs on the Galileo Mission

Mission Phase	Accident Branch	Release Branch	Accident Probability	Source Term, Ci Vapor	Source Term, Ci Particulates	Release Location
0		(No Releases)				
1	1A	SRB/FSS Impacts	4.0E-07	4.0E-01[a]	10.2[b]	Pad
	1B	Flame Trench/RRS Impact	1.7E-06	--	4.5E-01[b]	Pad
	1D	SRB/FSS Impacts SRB Impacts	9.5E-06	6.5E-02[a]	1.65[b] 1.62[b]	Pad 10 km above ocean
	Phase	SRB/FSS/Flame Trench/ RSS Impacts SRB Impacts	5.31E-05	1.46E-02[a]	3.86E-01[b] 2.90E-01[b]	Pad 10 km above ocean
2	2A	End-On/Side-On	5.0E-07	4.64E-02[a]	2.64E+02[b]	Mid-trajectory
	2B	End-On/Side-On	5.36E-05	2.56E-02[a]	1.46E+02[b]	Mid-trajectory
	2C	End-On/Side-On	1.4E-05	5.46E-02[a]	3.10E+02[b]	Mid-trajectory
	Phase	End-On/Side-On	6.81E-05	3.18E-02[a]	1.81E+02[b]	Mid-trajectory
3	3A (Phase)	End-On/Side-On	1.69E-04	1.65E-01[a]	9.40E+02[b]	High Altitude, 0^{o} Latitude
4	4C	End-On/Side-On	1.01E-04	1.65E-01[a]	9.40E+02[b]	High Altitude, 0^{o} Latitude
	Phase	End-On/Side-On	2.08E-02	8.03E-04[a]	4.56[b]	High Altitude, 0^{o} Latitude

a = 0.02 μm particle size
b = 50 μm particle size

B. REFERENCE DESIGN DOCUMENT

1.0 NUCLEAR SYSTEM (LIGHT-WEIGHT RADIOISOTOPE HEATER UNIT OR LWRHU)

1.1 General Description

The LWRHU is to be used on the NASA Galileo Mission to provide heat to various systems on the orbiter and probe. Many areas of the spacecraft are adversely affected by temperatures to which the spacecraft would meet during an interplanetary mission such as GLL. A singular design, in which auxiliary heat is provided in increments of 1 W, can fulfill the many requirements by proper distribution to those areas needing thermal input. This significantly cuts the spacecraft electrical energy drain, had these items been resistance heaters and, consequently, reduces the launched quantity of plutonium-238 for the same power consumption requirement.

The LWRHU is a heat source which contains sufficient plutonium-238 dioxide to provide 1 W of heat at the time of fabrication. Monsanto Research Corporation (MRC) is responsible for fabrication of the nonradioactive components and for issuing this document. Los Alamos National Laboratory (LANL) is responsible for design, assembly, and testing activities with analytical support from other Department of Energy (DOE)-authorized participants, such as Applied Physics Laboratory and Fairchild Industries. The overall configuration is shown in Figure B.1. External dimensions are 26 mm in diameter with a length of 32 mm. Part drawings are provided in Appendix B.1.

A general summary of the components with typical weights is given in Table B.1. In addition to those control documents cited in this table, the following process controls were employed:*

1. MD-70384, "Light-Weight Radioisotopic Heater Unit (LWRHU) Program Subassembly Manufacture." This document contains the procedures and in-process acceptance criteria employed for manufacturing the clad components and subassembly.

2. SPA-780244, "Specification for Marking of Parts and/or Assemblies." This document provides requirements for maintaining identification of parts.

3. SPA-790616, "Marking Parts." Laser marking requirements are identified in this general manual for both FWPF and metallic components.

4. SPA-790617, "LWRHU Subassembly." This document defines the acceptance of the clad assembly, including frit vent flow rate.

5. 26Y318194, "Light-Weight Radioisotope Heater Unit (LWRHU) Product Index," (LANL). This document lists the drawings, procedures, documents, and flow charts applicable to the LWRHU activities at LANL.

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*NOTE: MRC's documents may be obtained by contacting MRC's Drawing Control at Telephone (513)-865-3160.

LIGHTWEIGHT RADIOISOTOPE HEATER UNIT

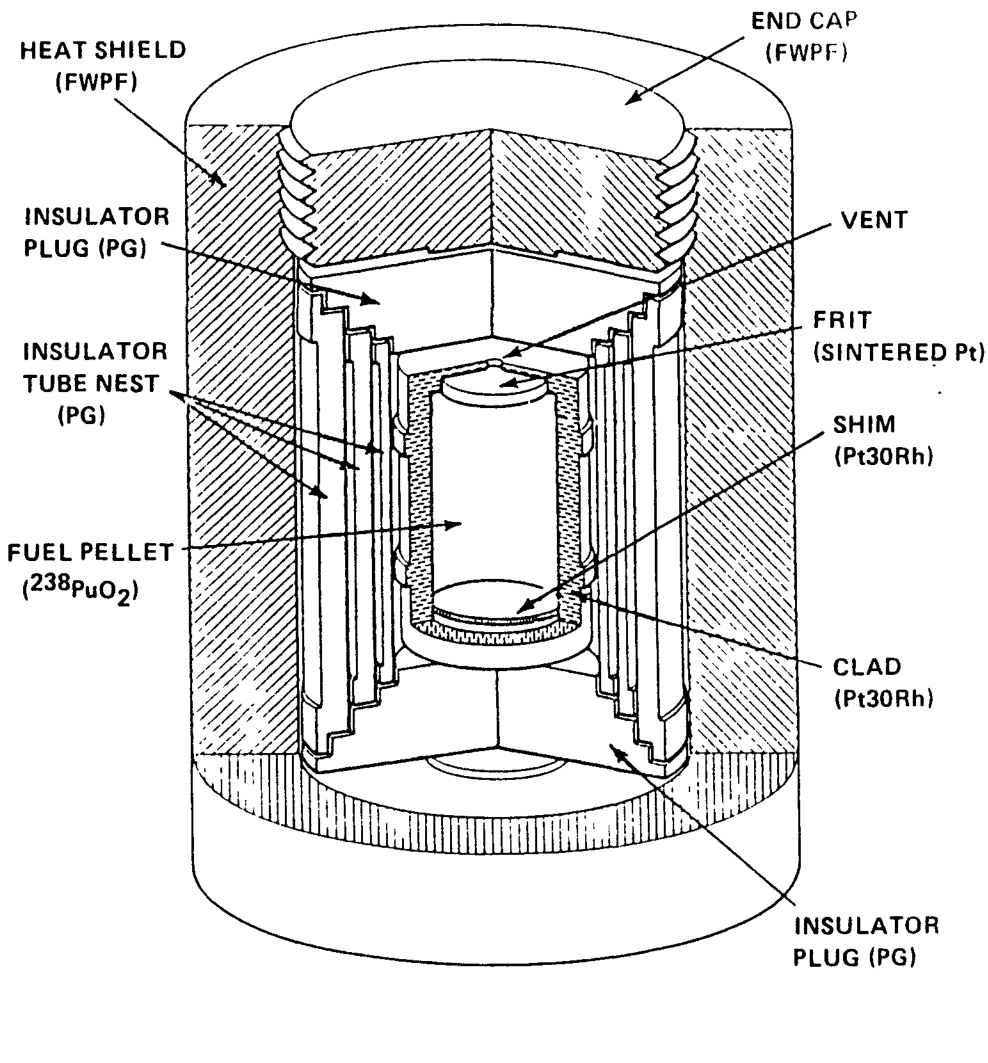


Figure B-1: The principal features of the Light-Weight Radioisotope Heater Unit are portrayed above.

TABLE B-1: General Description of the LWRHU Components

Component	Material	Material Specification	Drawing Number	Typical Weight, g
Fuel Pellet	PuO_2	--	AYC-790106	2.664
Frit Vent	Pt	SPA-790605	AYC-790098	
Clad Cap, Vent End	Pt-30Rh	SPA-790139	AYC-790099	
Clad Cap, Closure End	Pt-30Rh	SPA-790139	AYC-790105	5.702
Clad Body	Pt-30Rh	SPA-790139	AYC-790100	
Shim	Pt-30Rh	SPA-790139	AYC-790101	
Insulator Cap*	PG	SPA-790171	AYC-790381	2.532
Outer Insulator Body	PG	SPA-790171	AYC-790382	3.166
Middle Insulator Body	PG	SPA-790171	AYC-790383	1.486
Inner Insulator Body	PG	SPA-790171	AYC-790384	0.624
Aeroshell End Cap	FWPF	SPA-790053	AYC-790385	2.543
Aeroshell Body	FWPF	SPA-790053	AYC-790380	21.070
			TOTAL WEIGHT =	39.787

*Two Required

6. 26Y318189, "Light-Weight Radioisotope Heater Unit (LWRHU) Product Specification," (LANL). This specification defines the physical and chemical requirements of the plutonium oxide pellet, welding, vent activation, and graphite assembly requisites for the LWRHU.

7. Drawings: (LANL)
   26Y318190, "LWRHU Assembly"
   26Y318191, "LWRHU Fueled Capsule"
   26Y318192, "LWRHU Fuel Pellet"
   26Y200174, "Packaging Container - Shipping Fixture - LWRHU Assembly."

1.2 Design Requirements

The design requirements as excerpted from JPL's Spec No. ES513279, Revision A (c. October, 1984) are as listed below.

1.2.1 Function - The function of the LWRHU is to provide 1 W(th) by means of radioactive decay of its plutonium fuel. The unit is used to obtain a local desirable temperature of the GLL Orbiter and the GLL Probe.

1.2.2 Components - The LWRHU consists of the following:

a. Aeroshell and end cap (Fine Weave-Pierced Fabric Graphite*).

b. Insulator body (pyrolytic graphite).

c. Fuel clad (Pt-Rh-30).

d. Fuel pellet (PuO_2).

1.2.3 Power - The thermal power of the unit shall be 1.1 plus or minus 0.03 W, when back decayed to the date of pellet pressing.

1.2.4 Dimensions - The unit shall be a right circular cylinder with dimensions as shown in Figure B-2.

1.2.5 Weight - The unit shall have a maximum weight of 42 g.

1.2.6 Compressive Load - The unit shall withstand the following compressive load applied by the enclosure during the mission:

a. Radial compressive load of 890 N maximum uniformly applied of 120^{o} minimum arc by a 0.5-in. wide band.

b. End uniformly distributed compressive load of 445 N maximum.

1.2.7 Surface Temperature - The unit shall have operational capability with surface temperatures between -130 and + 180^{o}C.

*A product of AVCO Systems Divison, 201 Lowell Street, Wilmington, MA, 018870

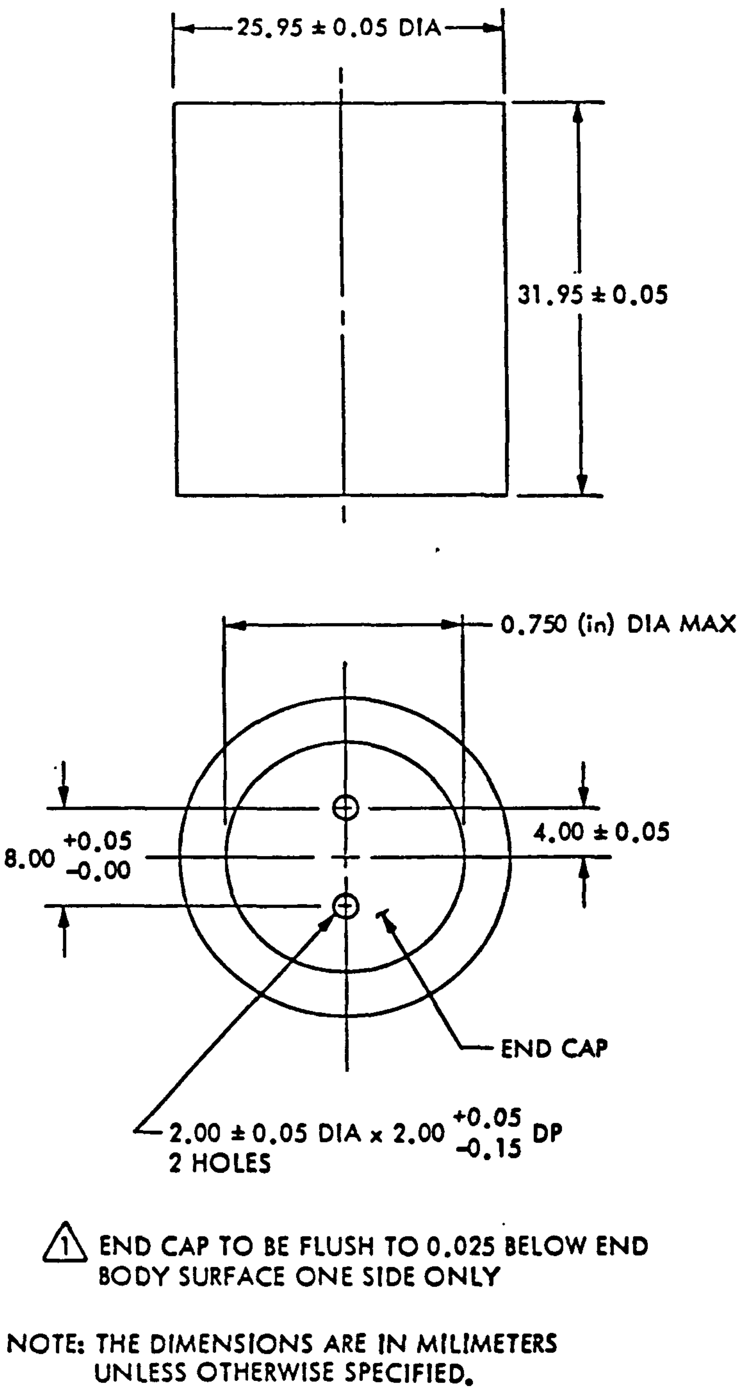


FIGURE B-2: LWRHU dimensions.

1.2.8 Magnetic - The unit shall contain no significant amount of ferromagnetic material. For the purpose of this specification, impurity amounts (a few hundred ppm) of Fe, Cr, and Ni in the fuel and in the clad shall not be considered to be significant ferromagnetic material.

1.2.9 Radiation - The unit shall be packed with plutonium dioxide fuel with maximum neutron emission of 6,000 n/g of plutonium-238/s. The concentration of plutonium-236, when decayed to the date of fuel production, shall not exceed 2 ppm of total plutonium.

1.2.10 Tests

1.2.10.1 Acceptance Tests - Each unit shall be subjected to the following measurements to verify compliance with the requirements of this document.

1.2.10.1.1 Mass - The mass of each unit shall be measured with accuracy of plus or minus 0.01 g.

1.2.10.1.2 Power - Each unit shall be subjected to thermal power measurements with accuracy of 3 percent.

1.2.10.1.3 Radiation - The neutron emission rate of each unit shall be measured with accuracy of 5 percent.

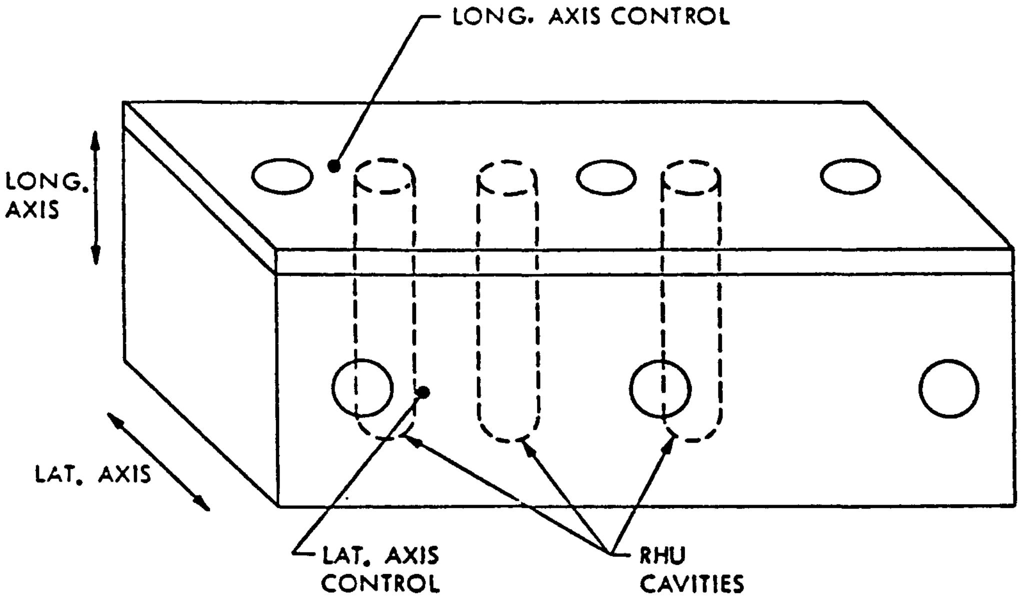
1.2.10.1.4 Geometry - Each unit shall be measured to ensure that it falls within the envelope dimensions defined by Figure B.2

1.2.10.2 Qualification Test - One unit shall be subjected to random vibration (1.2.10.2.1) sinsoidal vibration (1.2.10.2.2), and acceleration tests (1.2.10.2.3). In order to successfully complete the qualification test, the unit under test shall maintain its mechanical integrity during and after the qualification testing.

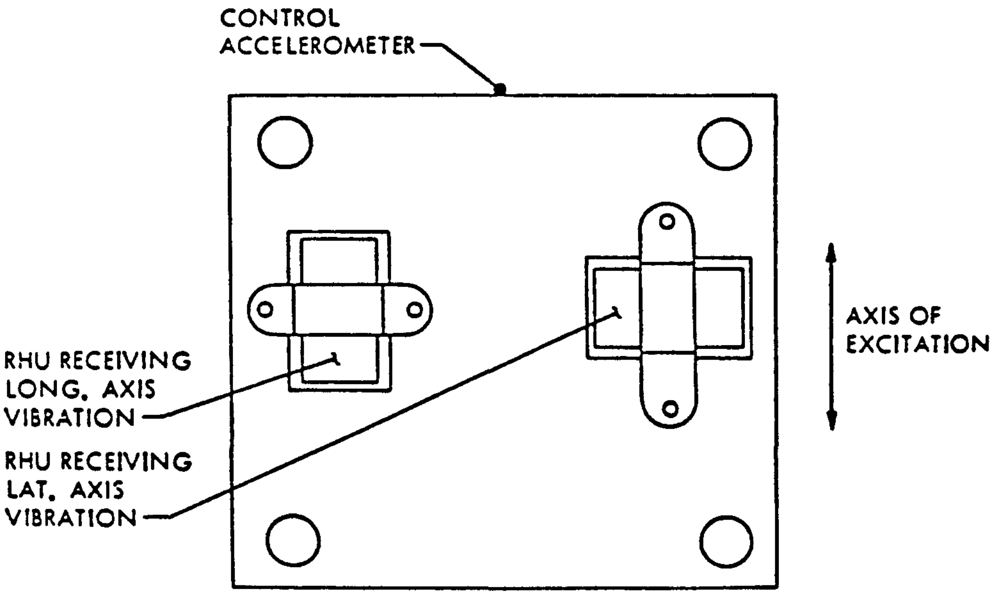
The unit selected for the qualification testing shall be a development unit with a depleted uranium dioxide fuel pellet, which is mechanically identical to the flight units. The qualification of the unit shall be based upon demonstrating that it is mechanically intact and meets the requirements of this and other applicable documents.

1.2.10.2.1 Random Vibration - The unit shall be vibrated along each of the two perpendicular axes while mounted in both Orbiter and Probe configuration as illustrated in Figure B.3. The random vibration levels shall be in accordance with Figure B-4.

1.2.10.2.2 Sinusoidial Vibration - The unit shall be subjected to the sinusoidal vibration along its perpendicular axis while mounted in the Orbiter and Probe configuration as shown in Figure B.3. The sinusoidal test level will shall be in accordance with Figure B.5.



ORBITER CONFIGURATION



PROBE CONFIGURATION

FIGURE B-3: LWRHU test configuration.

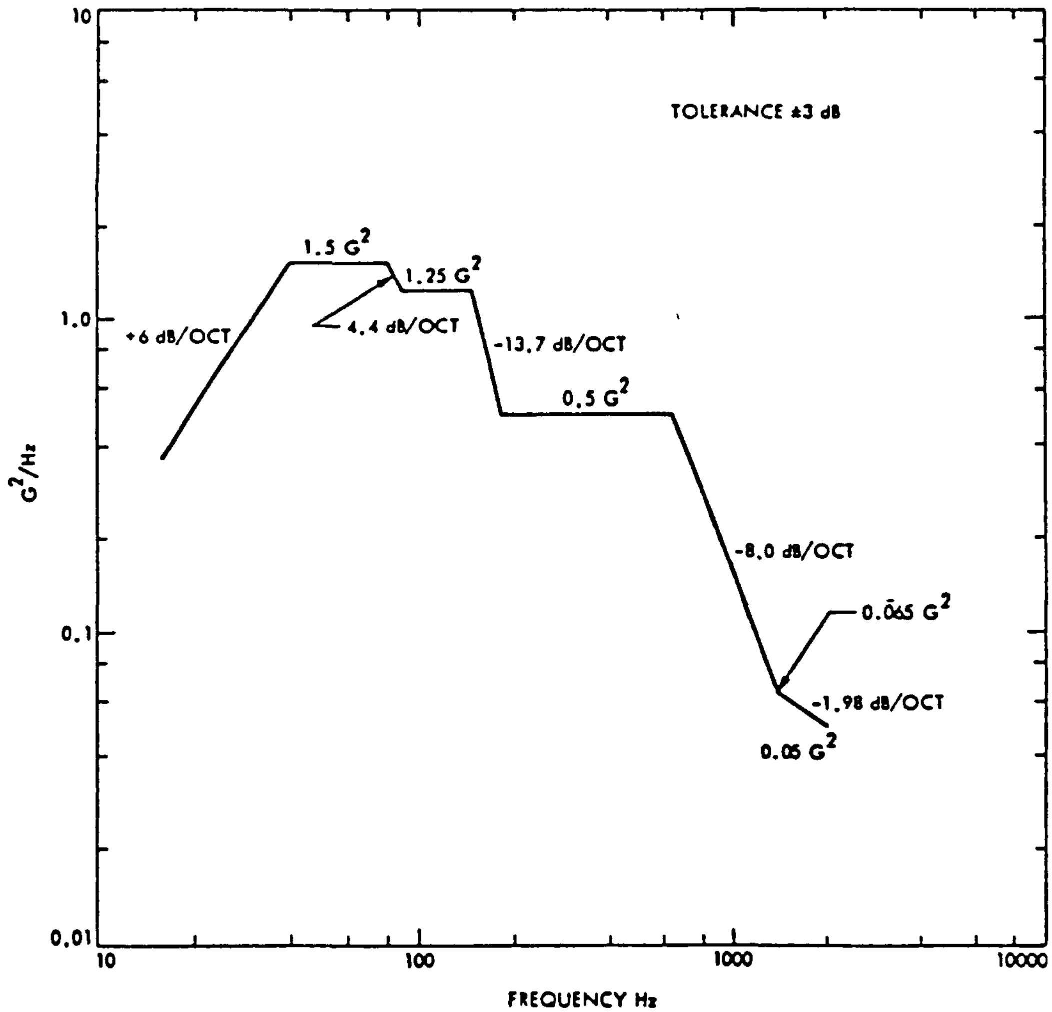


FIGURE B-4: Random vibration test levels.

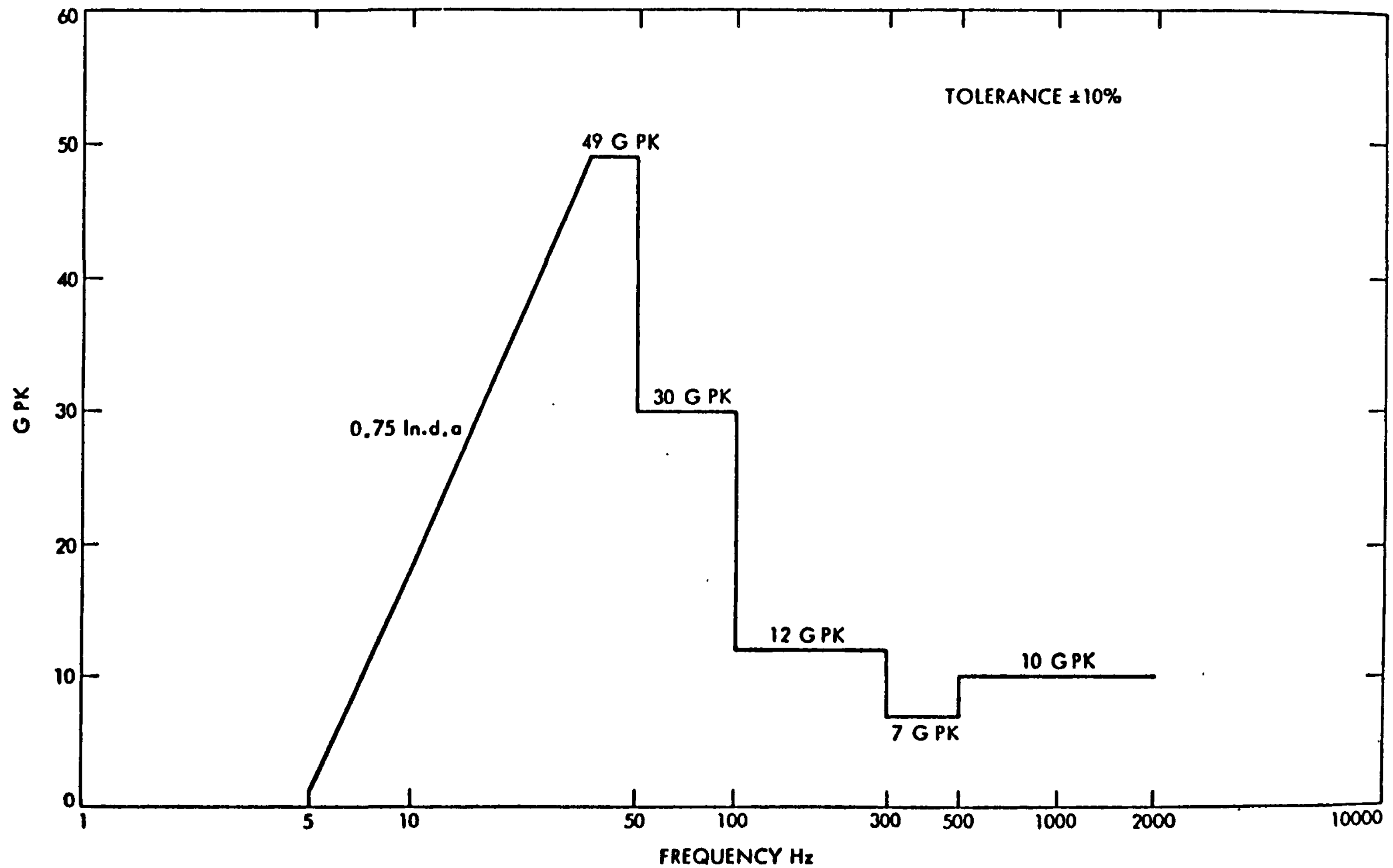


FIGURE B-5: Sinusoidal vibration test levels.

1.2.10.2.3 Acceleration Test - The acceleration test is performed to determine the response of the unit to forces that will occur when the GLL Probe descends into the Jovian atmosphere. The g loading force shall be perpendicular to the mounting plate and in a direction to stress the mounting strap and fasteners. The unit shall be subjected to a load of 4.25 g for 15 s and a load of 300 g for 30 s.

1.2.10.3 Quality Assurance Provisions

1.2.10.3.1 Quality Assurance Program - The DOE shall establish and implement a quality assurance program, which essentially conforms to the requirements of OSNP-2, Quality Assurance Program Requirements for Space and Terrestrial Nuclear Power Systems.

1.2.10.4 Safety Requirements

1.2.10.4.1 Safety Policies - The LWRHU shall meet the provisions of NASA NHB 1700.7A, Safety Policy Requirementds for Payloads Using the Space Transportation System. Safety data and analysis for the LWRHUs will be delivered by JPL with the overall Spacecraft Safety Data.

1.3 Materials and Their Properties

Material description and properties are being addressed in this section from an inside-to-outside basis.

1.3.1 PuO_2 Fuel - The 1.1-W plutonium oxide fuel pellets were hot pressed from Savannah River Plant feed which contains isotopic percentage (based on total Pu) and actinide concentrations (in weight ppm) as follows (Reference 9):

Pu-236:	0.00007
Pu-238:	83.62
Pu-239:	13.98
Pu-240:	1.96
Pu-241:	0.41
Pu-242:	0.14
Am-241:	254
Np-237:	203
Th-232:	232
U-234:	6191

These pellets weighed 2.664 $\pm$ 0.002 g with a density of 87.4 $\pm$ 0.7% of theoretical. The neutron emission was measured at 5190 $\pm$ 103 n/s - g Pu-238 with an average power of 1.106 $\pm$ 0.005 W per pellet (this works out to be 33.6 Ci or 1.24 TBq per pellet) (Reference 9).

The USAR for the GPHS RTG (Reference 3) provides an excellent summary of general physical properties data for PuO_2. The only properties employed in these analyses are:

Vapor Pressure: log P (total) = 7.381 - 28,890/T where P = atm and 1900 K < T < 2200 K,

and Heat Capacity:

T =	300	500	700	900	1100
C_p =	256	314	335	338	342

where T = °K and C_p= J/kg·K.

1.3.2 Pt-30 Rh Clad - The platinum-30 wt % rhodium material was obtained from the Matthey-Bishop Company of Malvern, Pennsylvania.

This material has been demonstrated to exhibit excellent compatability with stoichiometric plutonia (Reference 9).

Relevant tensile properties as measured by ORNL and cited in Reference 9 are shown in Figure B.6, A. and B. The thermal expansion coefficient is about 13 X 10^{-6} K^{-1}.

Physical properties of interest to this document include (Reference 10):

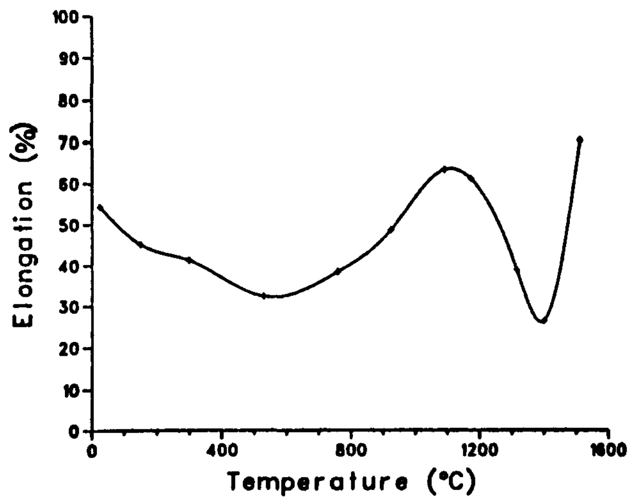
Melting Point = 2183 K
Pt-30 Rh/C eutectic ~2033 K
Emissivity = 0.2 (total normal)
Heat Capacity = 158 178 194 207 (J/kg·K)
@ Temperature = 400 900 1300 1700 K.

The vent is composed of sintered platinum powder. This item is an insignificant contributor to most thermal analyses, but the melting point of platinum (2043 K, Reference 11) is important in some accident scenario assessments.

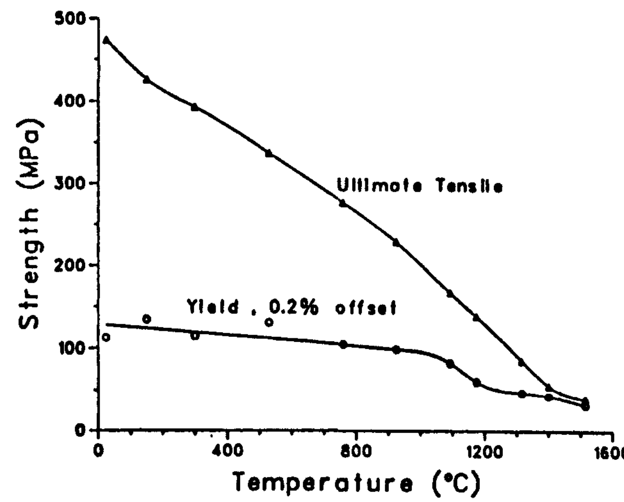
1.3.3 Pyrolytic Graphite Insulators - The insulation package which surrounds the clad and is immediately inside the aeroshell consists of pyrolytic graphite (PG) manufactured by Pfizer Company, Easton, Pennsylvania. The components are formed such that the c-axis is in the direction of heat flow (the AB plane normal to the heat flow direction). The properties of PG as they relate to this SAR are provided in Figure B.7 (Reference 12). The emissivity is given as 0.845 at 1590 K (Reference 12) and increases linearly from 0.41 at 256 K to 0.87 at 2200 K (Reference 11).

1.3.4 FWPF Aeroshell - The LWRHU aeroshell not only affords the required reentry protection (with ~50% ablation in a worst case) but also provides added impact resistance and fragmentation survivability to the fueled clad, should unplanned events be encountered which would result in perturbation of those sorts. The material used to fabricate the aeroshell is FWPF (Fine-Weave Pierced Fabric®), a product of AVCO Systems Division, Wilmington, Massachusetts. This 3DCC material consists of layers of fine-weave graphite fabric (x-y plane) that is pierced by stiffened graphite yarn bundles in the z axis. The bulk density is > 1.95 g cm^{-3}. The minimum room temperature tensile requirements are (Reference 13):

	Tensile Strength MPa	Tensile Modules MPa
x-y direction	125	58.6
z direction	114	6.55



A. Elongation of Pt-30 Rh at temperature (Oak Ridge)



B. Strength of Pt-30 Rh at temperature (Oak Ridge)

FIGURE B-6: Measured tensile properties of Pt-30 Rh are shown above.

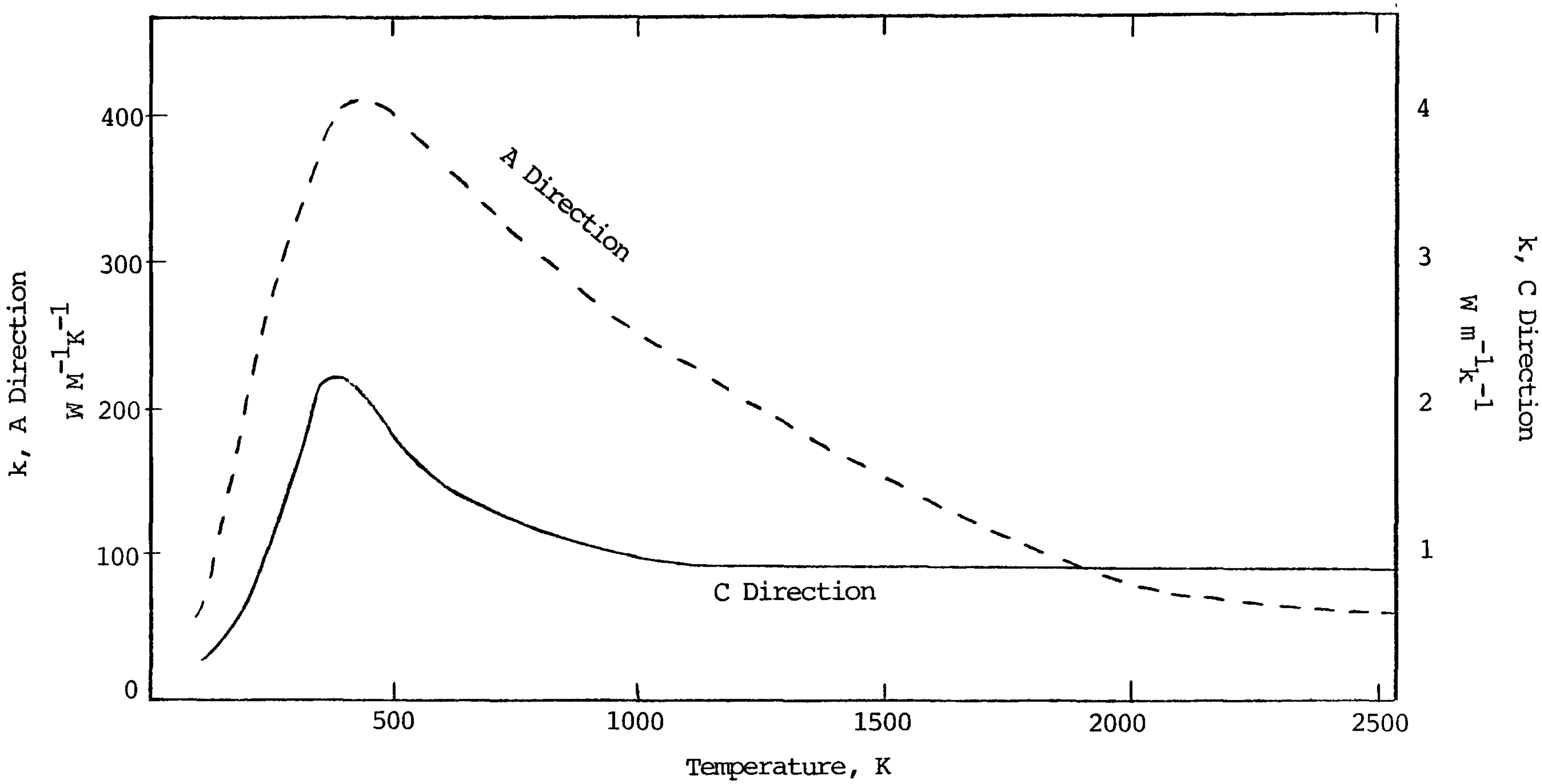


FIGURE B-7: The thermal conductivity, k, of pyrolytic graphite is plotted below for both A (parallel to basal plane) and C (perpendicular to basal plane) directions as a function of absolute temperature (Aerojet General, Vol. 2, NERVA Program, Report No. 2275). (Continued)

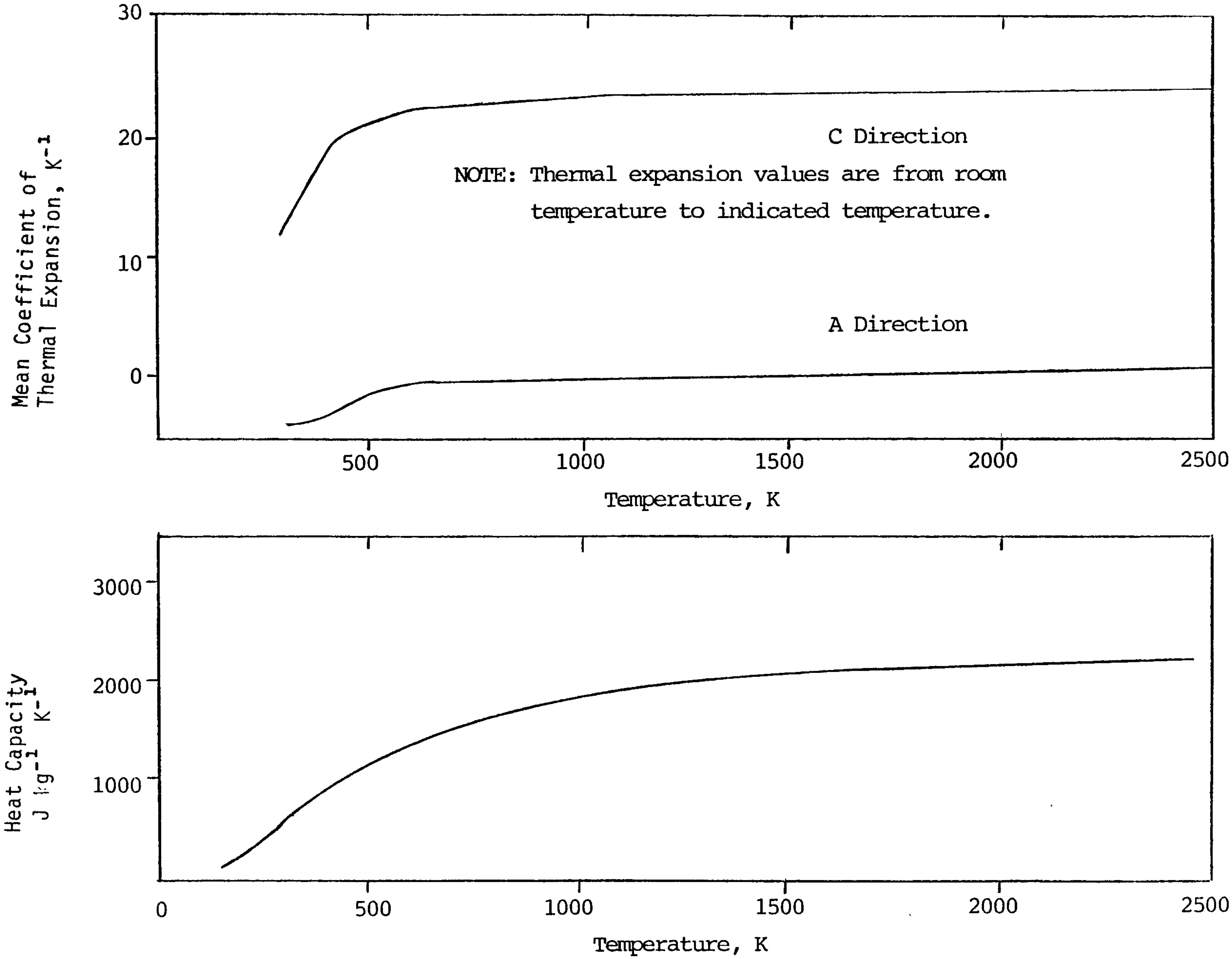


FIGURE B-7 (continued): Thermal expansion and heat capacity values for pyrolytic graphite are given as a function of absolute temperature (Aerojet General, Vol. 2, NERVA Program, Report No. 2275).

Thermal conductivity and emissivity behavior for FWPF as a function of temperature is provided in Figure B.8. The USAR (Reference 3, Figures B-15 through B-38) provides a more comprehensive compilation of mechanical properties for this material.

1.4 Radiation Properties

1.4.1 Gamma - The intensity and energy distribution of gamma radiation emanating from the LWRHU is not a specification. Representative gamma dose rates on LWRHU Fueled Clad No. 033 are shown in Figure B.9 (Reference 14).

The gamma spectrum for the LWRHU would not be significantly different from the GPHS Qual RTG measurement, which is shown in Figure B-10 (Reference 15). These data were taken by an Ortec GMX-25195-S high-purity germanium gamma-ray detector.

1.4.2 Neutron - The measured neutron emission from LWRHU clads has been reported as 5190 $\pm$ 103 n/s.g ^{238}Pu (Reference 9).

The anticipated neutron dose rate as a function of distance is provided in Figure B.9 (Reference 14). The neutron spectrum for the Qual GPHS RTG is given in Figure B.11 (Reference 16). The LWRHU neutron spectrum would be slightly different, due to fewer neutron- induced fission neutrons which are dependent on fuel pellet size.

1.5 Ground Suppport Equipment (GSE)

Other than tooling and fixturing required to produce the LWRHU, the only GSE required is the 5790 Shipping Container described in Reference 17. The overall configuration is shown in Figure B.12; LANL modified slightly the internal configuration to restrain the LWRHUs for shipment.

2.0 SPACECRAFT

2.1 A more detailed description of the GLL Mission than was given in the Introduction is provided in the following (excerpted virtually verbatim from JSC-08116, Reference 8):

The overall scientific objectives of the Galileo Mission are to conduct comprehensive investigations of the Jupiter planetary system by making *in situ* and remote measurements of the planet, its environment, and its satellites. Investigations of the Galilean satellites of Jupiter will constitute a major objective of the mission. A close-up study of the planet and its principal satellites will greatly extend our knowledge of the role of the Jovian system in the complex and analogous relationships existing between the Sun and its planetary system.

The Galileo spacecraft consists of both a Jovian Orbiter and a Jovian atmospheric entry probe. The Orbiter Mission encompasses an equatorial tour of the planet system and multiple encounters with the Galilean satellites Europa, Ganymede, and Callisto. An Io encounter is possible before orbit insertion. The Orbiter will conduct a synoptic study of the Jovian atmosphere, determine the distribution and stability of trapped radiation, and define the topology and dynamics of the outer magnetosphere, magnetosheath, and bowshock.

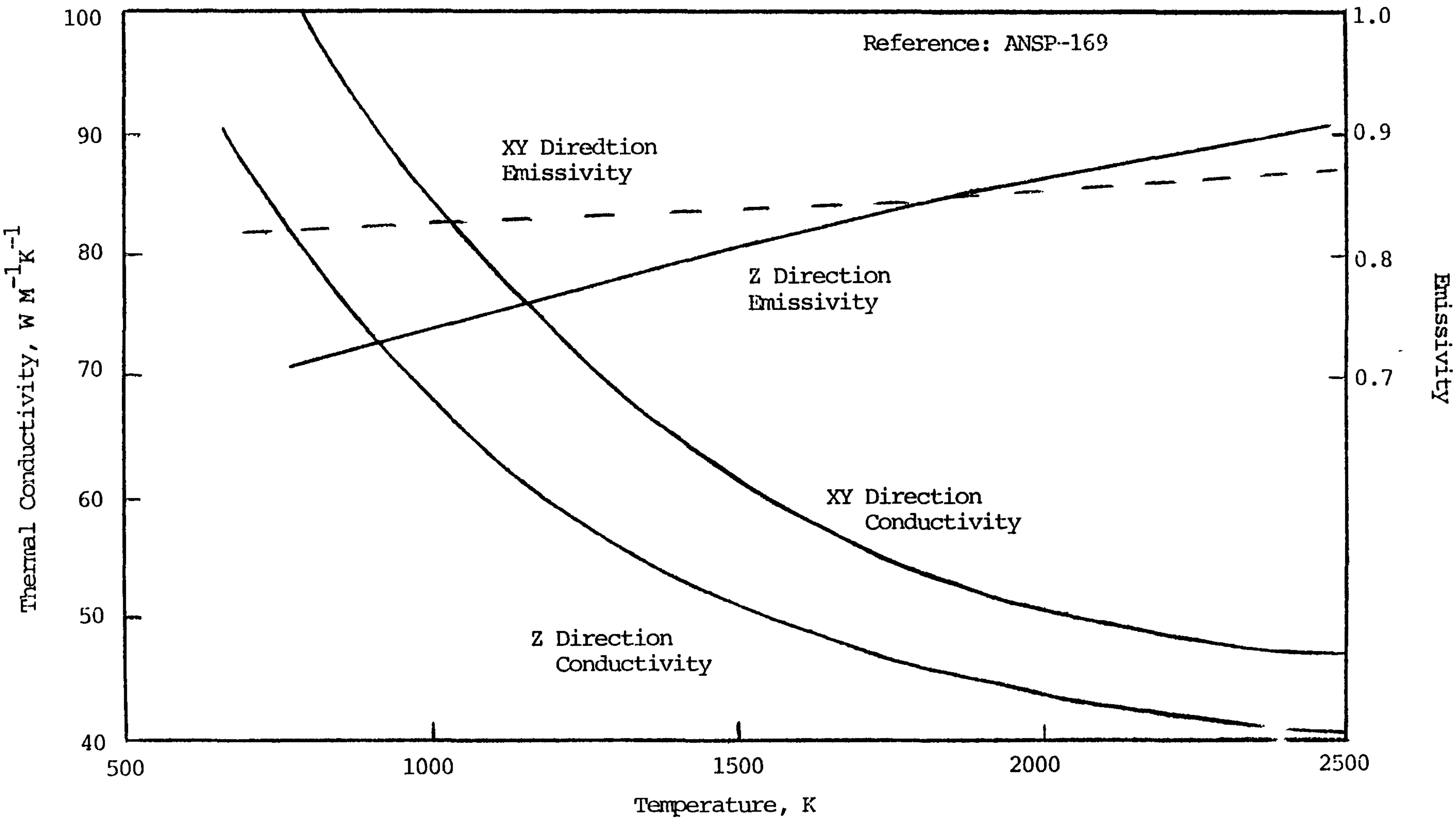


FIGURE B-8: Thermal conductivity and emissivity as a function of absolute temperature for FWPF.

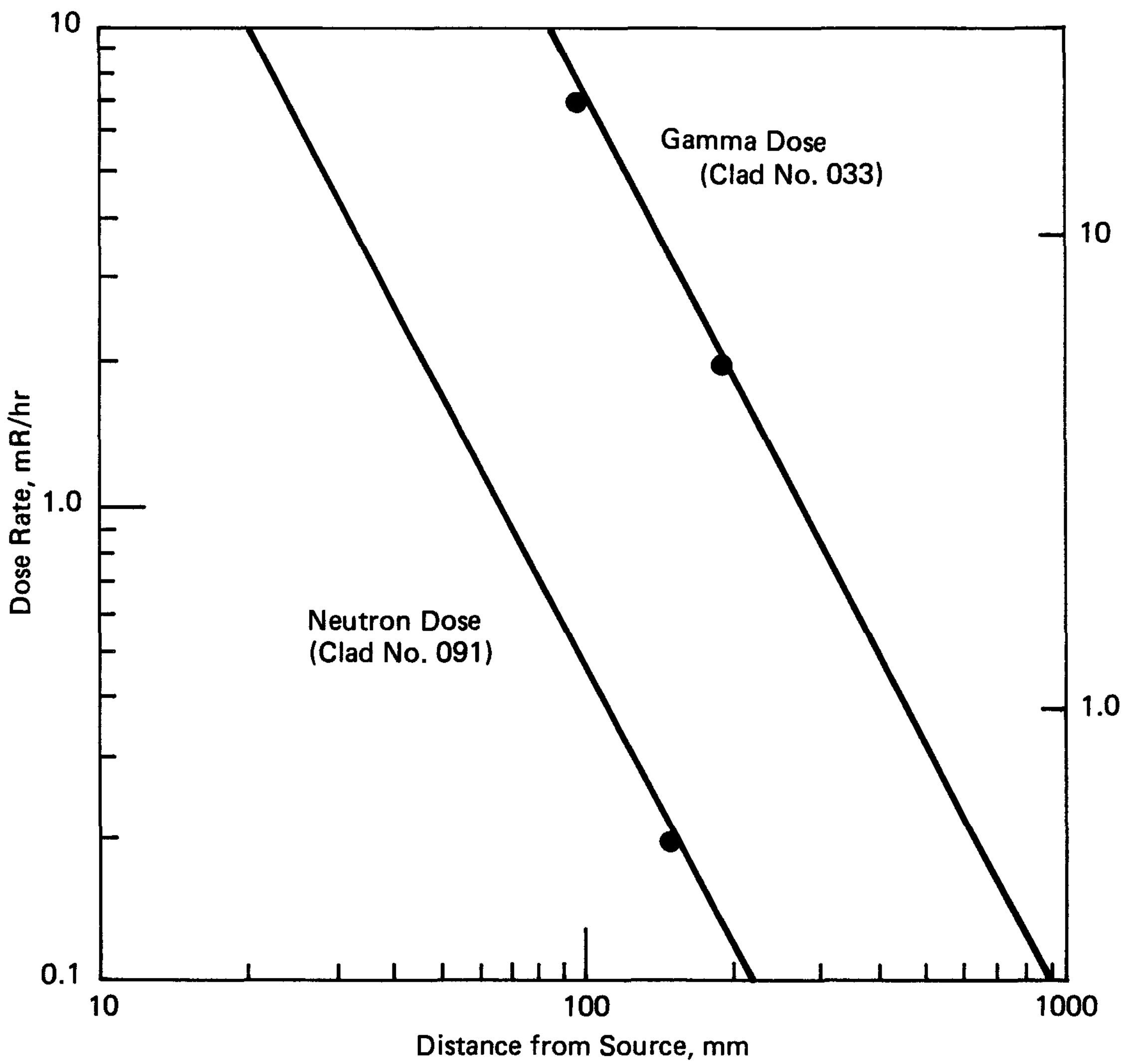


FIGURE B-9: Dose rates as a function of distance for gamma and neutrons from two representative LWRHU fueled clads are provided above.

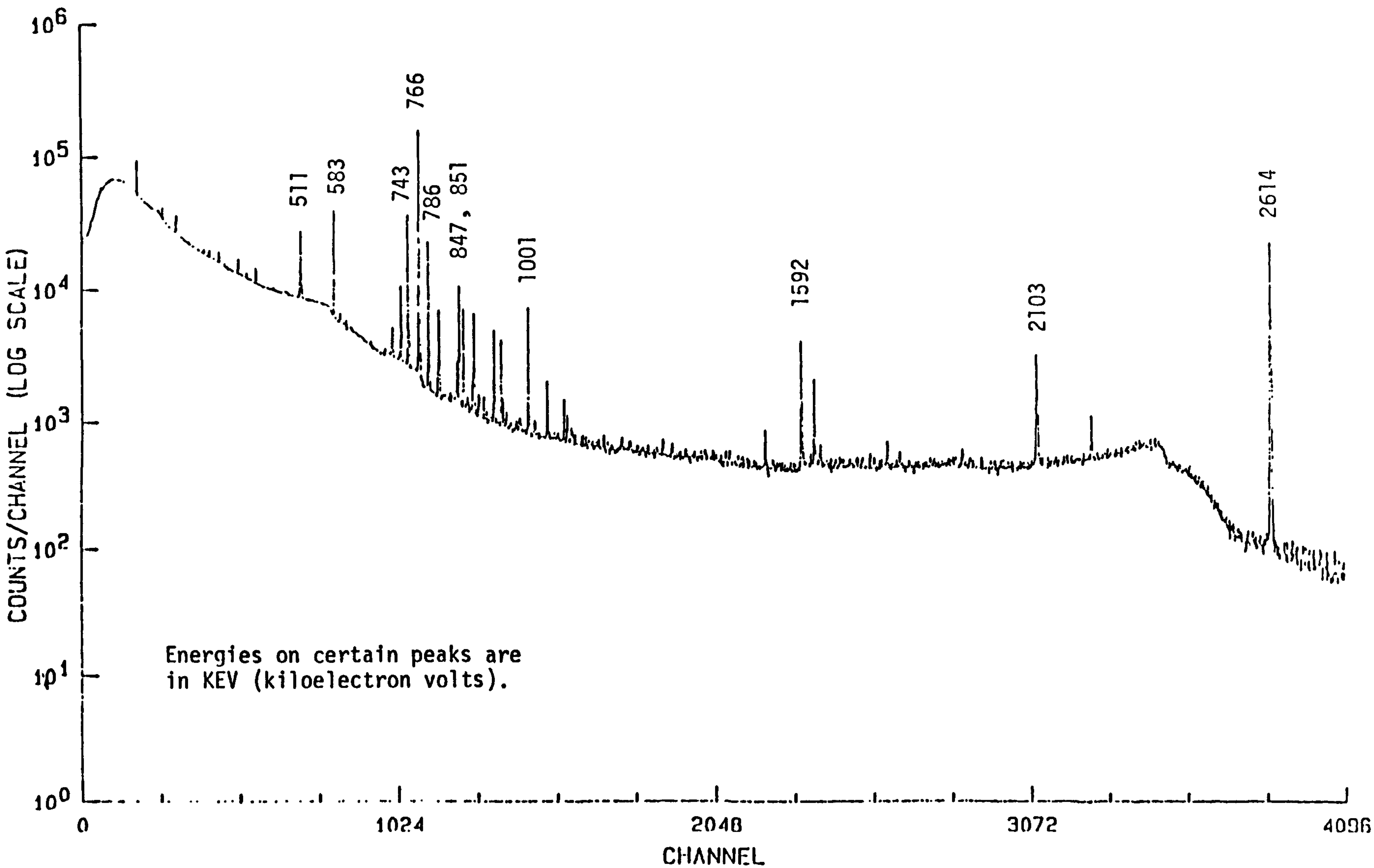


FIGURE B-10: The gamma spectrum of the Qual GPHS RTG is shown above. It is expected that the LWRHU spectrum would resemble this.

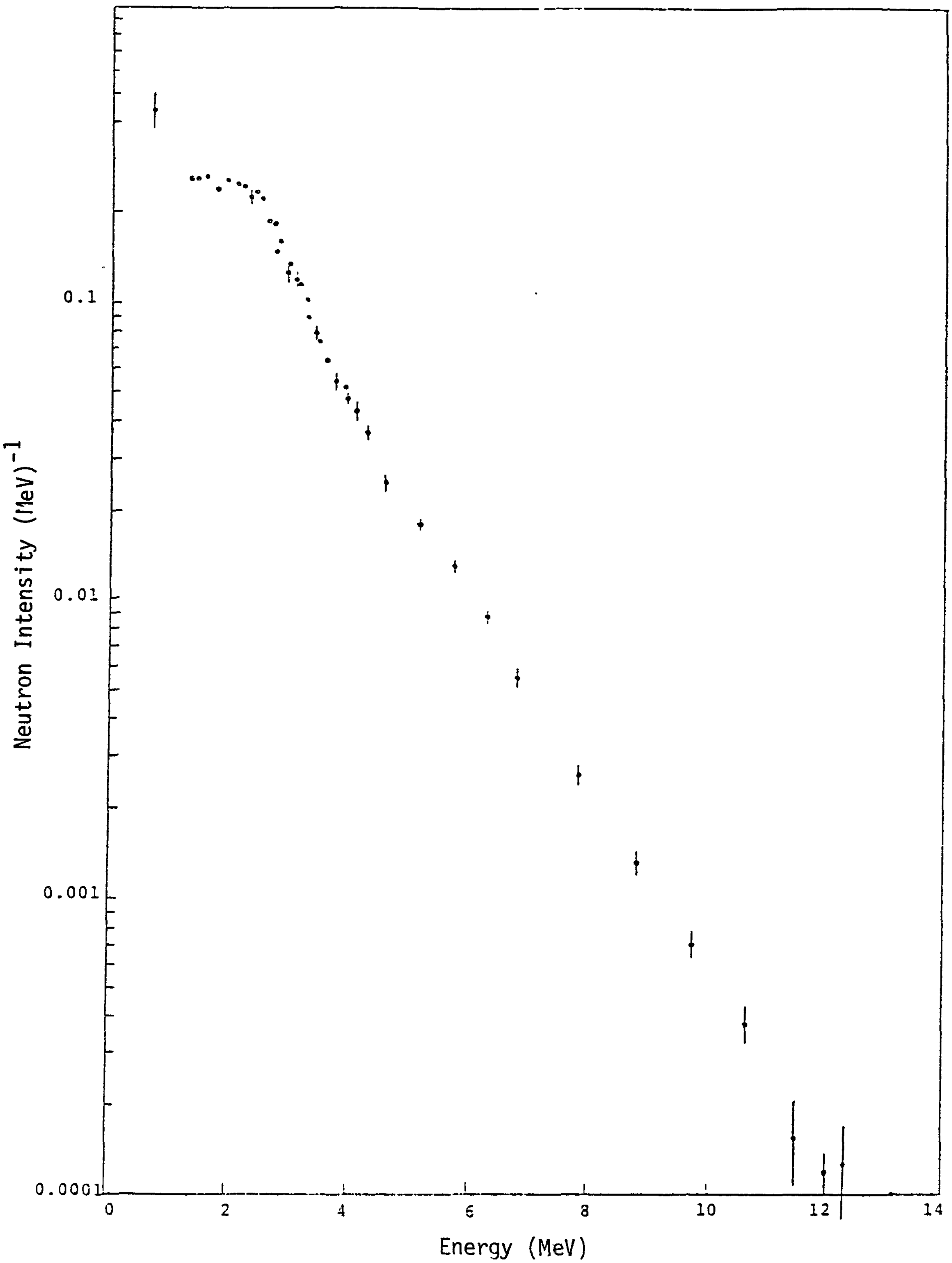


FIGURE B-11: The measured Qual GPHS neutron spectrum is shown here. The LWRHU neutron spectrum would be only slightly different in appearance.

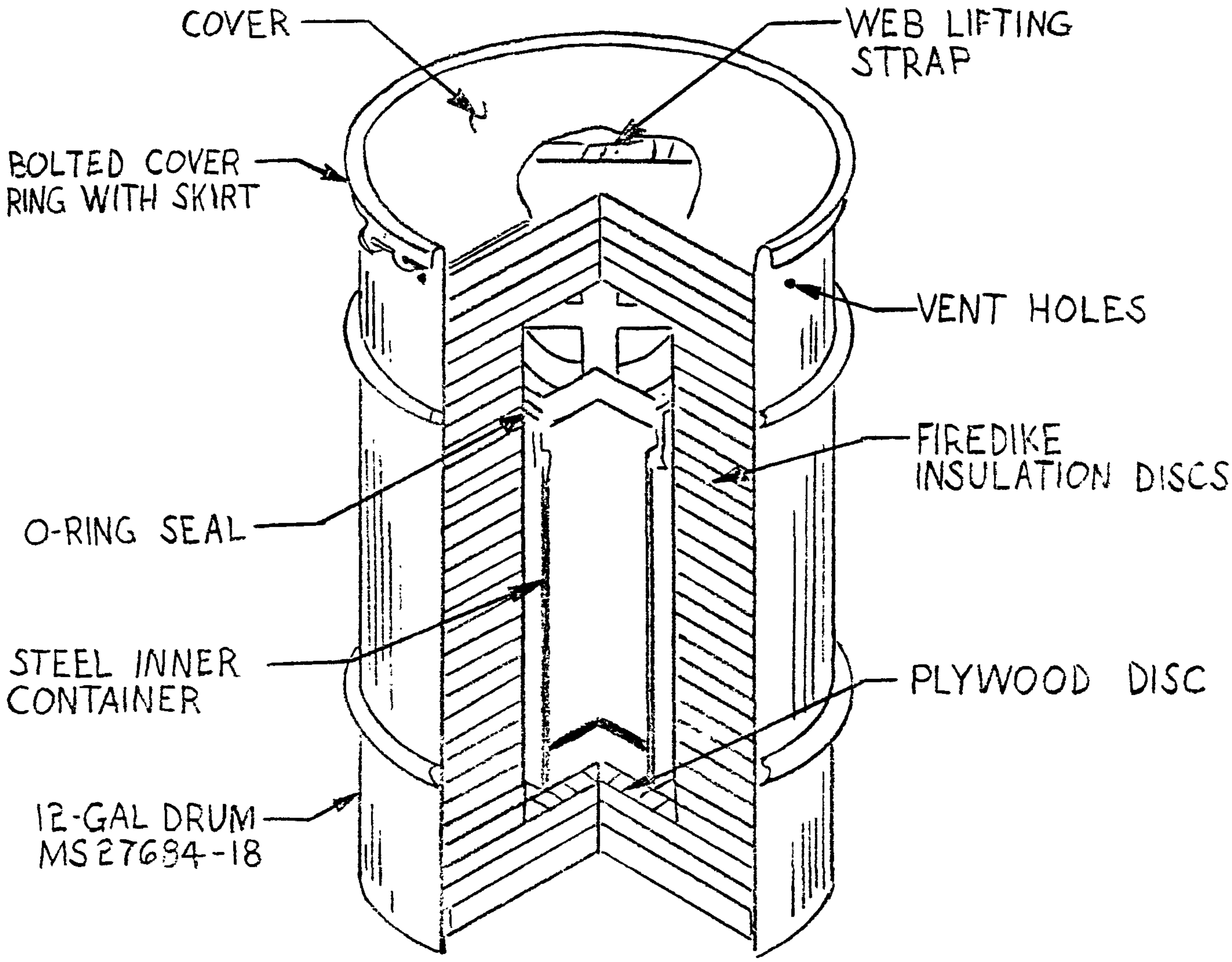


FIGURE B-12: The shipping container for transportation of the LWRHU is shown schematically above.

The probe will seek to determine the physical structure and chemical composition of the Jovian atmosphere to a pressure depth of at least 10 bars, the location and structure of the Jovian clouds in the troposphere, and the thermal balance of the planet. It will also characterize the uppe atmosphere and the nature and extent of the cloud particles.

Tables B.2 and B.3 provide a more detailed list of the experiments and objectives of the orbiter and probe portions of the GLL Mission.

2.2 Nominal Prelaunch Sequence of Events

The overall launch period begins with the transporting of the spacecraft (with many LWRHUs already installed) to KSC, extends through lift-off, and terminates with a successful Deep-Space Network (DSN) acquisition. The nominal prelaunch (prior to lift-off) sequence of events begins when the Galileo spacecraft is transported from JPL to KSC by truck. After being unpacked in the Spacecraft Assembly and Encapsulation Facility (SAEF), the spacecraft is inspected to determine if any damage was sustained during transit. Baseline tests, including the use of S- and X-band RF subsystems, are evaluated.

At the conclusion of the propulsion loading and pyrotechnics installation, the spacecraft will again be activated for further testing and will be prepared for transfer to the Vertical Processing Facility (VPF). At the VPF, the spacecraft will be mated to the Centaur G' upper-stage vehicle. Tests via the Merritt Island Launch Area (MILA) will be conducted to verify Space Shuttle end-to-end communications using cargo-integration test equipment to provide simulated Shuttle data interfaces. A series of operational tests will also be conducted. Upon satisfactory completion of these tests, the spacecraft will be placed into the storage mode to await shipment to the launch pad.

About 30 days before launch, the spacecraft will be removed from storage and transported to the launch pad, preceding the shuttle's arrival by two days. After cargo preparation procedures are completed, the spacecraft will be installed in the Shuttle Orbiter Bay. About four days before launch, the RTGs are reinstalled onto the spacecraft. Final end-to-end communication tests, using MILA and the STS Tracking Data Relay Satellite (TDRS) communication links, will then be conducted. These tests should last about two days. At their conclusion, the launch countdown will commence.

2.3 Normal Launch to Jupiter Arrival

The STS launch initiating the Galileo Mission is scheduled to occur in May or June, 1986. The Centaur G' vehicle will be used to boost the spacecraft from the low-altitude Earth orbit attained by the Shuttle. Although direct trajectories to Jupiter in 1986 require launch energy (C_3) in excess of 84 km^2/s^2, broken lane trajectories can be flown with lower values of C_3 with the spacecraft supplying additional ΔV to change the heliocentric orbit plane. The Centaur will accelerate the Galileo spacecraft from the Shuttle deployment orbit (i.e., parking orbit to a C_3 of about 80 km^2/s^2).

TABLE B-2

THE GALILEO ORBITER WILL STUDY THE AREA SURROUNDING JUPITER, INCLUDING ITS SATELLITES

INSTRUMENT NAME	OBJECTIVES
REMOTE SENSING INSTRUMENTS	
Solid state imaging	Image Jupiter and its satellites for studies of atmospheric dynamics and physical geology.
Near infrared mapping spectrometer	Study mineralogy of satellite surfaces, as well as morphology and structure of Jovian clouds.
Photopolarimeter radiometer	Study photometric and thermal properties of satellite surfaces as well as cloud and haze properties in Jovian atmosphere.
Ultraviolet spectrometer	Study compostion and structure of high neutral atmospheres of Jupiter and Galilean satellites.
FIELDS AND PARTICLES INSTRUMENTS	
Magnetometer	Study magnetic field of Jupiter and search for magnetic fields associated with the satellites.
Plasma	Study Jovian plasma.
Plasma Wave	Study time-varying electric and magnetic waves in the Jovian plasma.
Energetic particles detector	Measure detailed energy and angular distribution of protons, electrons, and ions.
Dust detector	Study physical and dynamical properties of small dust particles in the Jovian environment.
RADIO SCIENCE	
Celestial mechanics	Study gravity fields of Jupiter and its satellites, as well as the space environment.
Radio propagation	Study structure of atmospheres of Jupiter and satellites by use of radiosignals from Orbiter and probe.

TABLE B-3

THE GALILEO PROBE WILL ENTER JUPITER'S ATMOSPHERE AND PERFORM NUMEROUS MEASUREMENTS

INSTRUMENT NAME	OBJECTIVES
Atmospheric Structure Instrument	Determine state properties (temperature, pressure, density and molecular weight) of Jovian atmosphere.
Neutral Mass Spectrometer	Determine chemical and isotopic composition of Jovian atmosphere.
Helium Abundance Detector	Perform precision determination of helium abundance measurement in Jovian atmosphere.
Nephelometer	Determine microphysical characteristics (particle size distribution, number, density, and physical structure) of Jovian clouds.
Net Flux Radiometer	Measure vertical distribution of net flux of solar energy and planetary emissions.
Lightning and Radio Emission/ Energetic Particle Detector	Study lightning in the Jovian atmosphere and energetic particles near Jupiter.
Radio Science	Study composition and structure of Jovian atmosphere.

2.3.1 Separation from Shuttle and Centaur - Galileo launch mode telemetry will be available to near-real-time via the STS-TDRS communications link, from lift-off to Shuttle-Centaur separation. Based on the telemetry data, a decision to continue with the planned flight to Jupiter must be made by Launch plus 3 hr. If a "go" decision is made, the Shuttle-Centaur separation should occur during the third Shuttle orbit, at about 4 hr after launch. During the more favorable periods of the launch window, the separation could be delayed until the fourth or the fifth orbit without jeopardizing the objectives of the Galileo Mission. After separation, a Centaur S-band link can be used to route data between the spacecraft and Shuttle or DSN. The maximum useful range of the Centaur to Shuttle link is 10 km.

About 45 min after Shuttle-Centaur separation, the Centaur main engine will burn for approximately 10 min. After MECO, the Centaur will initiate a slow thermal roll of 0.1 rpm, and the Galileo spacecraft will start deployment of the RTG, science, and magnetometer booms. The spacecraft transmitters will then be activated to provide a down-link through TDRS just prior to Centaur-spacecraft separation. Centaur will turn the spacecraft to point 8^{o} to the Earth side of the Sun and will increase the spin of the spacecraft to 2.9 rpm. The spacecraft will then separate from Centaur, which will maneuver to avoid the same trajectory path as the Galileo spacecraft. The spacecraft transponder will now be the only means of exchanging data between the Earth and the spacecraft.

2.3.2 Post-Separation to Jupiter Arrival - Immediately after separation, the high-gain antenna will be deployed, and the retropropulsion module (RPM) will be pressurized. During these events, DSN acquisition of the down-link will be established (approximately 15 min after Centaur-spacecraft separation). After RPM pressurization, the Galileo spacecraft will perform a Sun acquisition (approximately 1 hr after Centaur-spacecraft separation).

About eight months after launch, the spacecraft propulsion system will impart a ΔV of about 200 m/s using its RPM 10-N thrusters. The purpose of this maneuver is to change slightly the heliocentric inclination, so that the spacecraft will intercept Jupiter at the desired arrival date. The position of Jupiter at the nominal arrival date is about one degree below the ecliptic. Ballistic trajectories without this plane change maneuver require large heliocentric inclinations at launch. This plane change maneuver thus reduces the C_3 significantly, since the spacecraft trajectory departing the Earth lies nearly in the ecliptic.

The interplanetary cruise operations activities (e.g., cruise science, navigation, spacecraft monitoring, etc.) on the Earth-to-Jupiter trajectory will be comparable to those planned for other missions. The trajectory must be targeted for a Jupiter arrival no earlier than August, 1988, in order to satisfy the probe-to-Orbiter relay link geometry. Arrival dates between mid-September and late October are excluded, so that probe separation does not occur near the time of solar conjunction. Therefore, arrival at Jupiter will probably occur sometime between early August and mid-September. Arrival dates after October are possible, but less desirable due to increased flight times.

During the cruise to Jupiter, the probe will be in a nonenergized condition with both power and command capabiltiy inhibited, except for periods of probe checkout. After a cruise period of about 675 days, and about 150 days prior to arrival at Jupiter, the Orbiter and probe will be spun up to 10 rpm, and the probe will be separated from the Orbiter. Except for a timer, the probe is passive during the 150-day coast towards Jupiter. The probe is targeted to a daylight entry within a few degrees from the equator. At an altitude of 450 km above 1-bar pressure, it will have a nominal speed of less than 47.8 km/s and flight path angle of -8.6° relative to the atmosphere. Deceleration from this point to sonic velocity will take place over a period of about 2 min, during which the probe will experience a nominal peak deceleration of about 250 g and peak heating rates approaching 500 MW/m^2. At approximately Mach 1, a pilot parachute will be deployed by a mortar whose pyrotechnics firing signal is derived from measurements of the deceleration profile. The pilot parachute, in turn, will remove the deceleration module aft cover and deploy a main parachute. The main parachute will then separate the descent module from the deceleration module and control the speed of the descent module throughout the remainder of the mission. Those instruments and subsystems which had not been actuated prior to entry will be turned on at the time of descent module separation. Instrument data acquired prior to and during entry will then be transmitted to the Orbiter, together with data acquired during the remainder of the descent through the atmosphere. The probe is designed to operate to a pressure level of at least 10 bars.

2.4 Jovian System Exploration

About 150 days after separating from the probe, the Orbiter will fire its retro engine, thereby entering orbit about Jupiter. Following orbit insertion and the perijove raise maneuver that is necessary to reduce the radiation dose the Orbiter will receive, a series of close encounters with the Galilean satellites will be targeted. These encounters will not only permit close-in scientific investigations, they also will provide gravity assists to the Orbiter, providing the necessary trajectory shaping required to reach subsequent satellites in the tour. The remaining Orbiter propulsive capability is used primarily for navigational purposes. Multiple encounters with the same satellite over the duration of the satellite tour will allow exploration of both equatorial and polar regions. In addition, the satellite tour will allow intensive scientific measurements of Jupiter and its magnetosphere. The nominal mission will end 20 months after orbit insertion.

2.4.1 Galileo Payload Description - The paragraphs below use narrative provided in Reference 3 to describe the GLL payload. The Galileo spacecraft (S/C), consisting of a Jovian Orbiter and an atmospheric probe and weighing over 2500 kg, is mounted atop a Centaur upper stage.

Figure B-13 depicts the S/C in its stowed configuration, and Figure B-14 depicts the S/C in its cruise configuration. Separation of the S/C from the Centaur occurs shortly after the Centaur burn, which impels the S/C from Earth orbit towards Jupiter. The S/C is controlled by the Orbiter during the cruise to Jupiter. During this cruise, both status telemetry and occasional probe checkouts are powered by the Orbiter. The probe and Orbiter remain an integral unit until about five months before arrival at Jupiter, when the probe goes on its own internal power just prior to separating from the Orbiter.

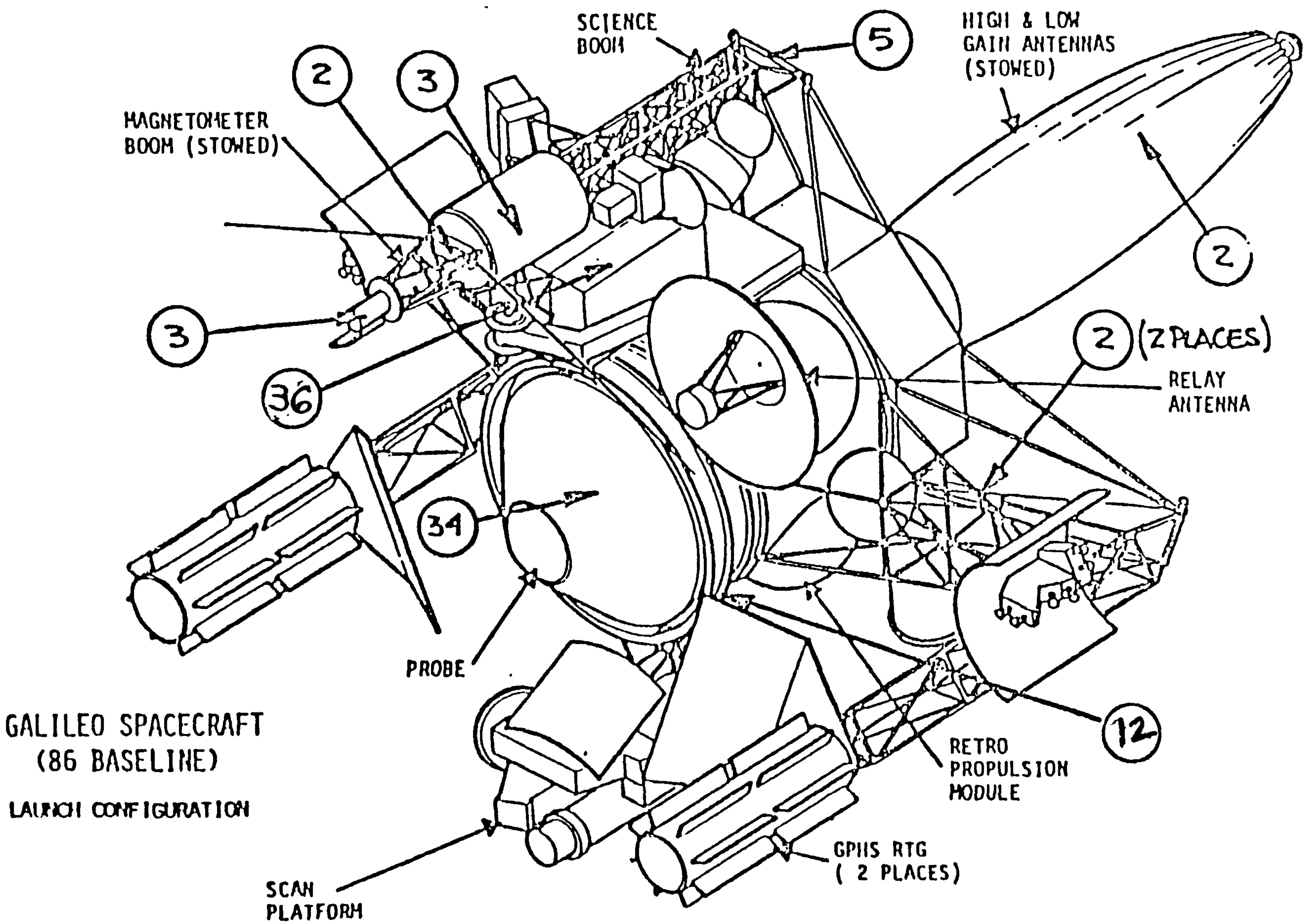


FIGURE B-13: Locations of LWRHUs on the stowed GLL spacecraft are shown above (same illustration as A-4).

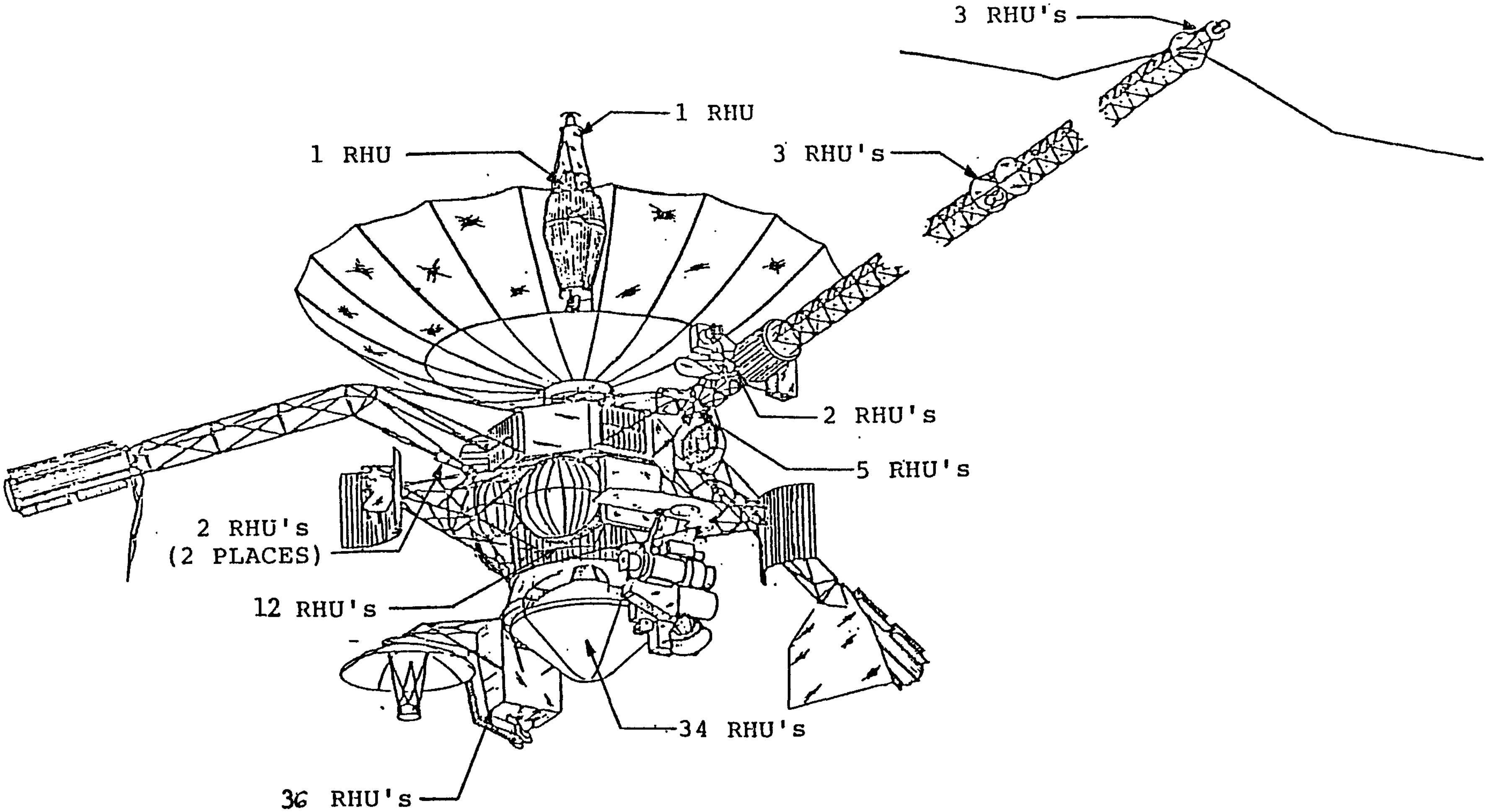


FIGURE B-14: The cruise configuration of the GLL spacecraft showing the location of the 101 LWRHUs. (Continued)

2.4.1.1 Orbiter Description - The Galileo Orbiter (Figure B-14 Continued) is a dual-spin spacecraft. Part of the spacecraft will be three-axis stabilized to provide a steady base for the remote-sensing instruments. These instruments must be precisely pointed. The despun section carries its related electronics. The main portion of the Orbiter will spin at 3 rpm to provide stability and to allow its instruments to continuously "sweep" the sky to make their measurements. The spun section contains both the high- and low-gain antennas, the RPM for all propulsive and attitude maneuvers, the nuclear power sources, most of the electronics, most of the command and data equipment, and the fields and particles science instruments.

The RPM has one 400-N engine and two clusters of 10-N thrusters that are used for attitude control and the smaller propulsive maneuvers. Because of minimum burn size and the total wetted lifetime constraints, the 400-N engine will be used only for the deflection maneuver after probe release, orbit insertion, and the perijove raise maneuver. Interplanetary maneuvers will be done using the 10-N thrusters.

Since Jupiter is too far from the Sun for solar cells to provide enough electrical power, Galileo will use radioisotopic thermoelectric generators (RTGs) similar to those flown on the two NASA Voyager spacecrafts. To conserve electrical power, 67 LWRHUs are dispersed about the orbiter as defined in Figures B-13 and B-14. The one-watt heaters are the other major source of radiation aboard the GLL S/C (although considerably less than the RTGs).

2.4.1.2 Probe Description - The general appearance and dimensions of the probe are shown in Figure B-15. A more detailed depiction of the protective housings which would tend to provide additional protection for the 34 LWRHUs contained within the probe during accident situations is given in Figure B-16. Within the probe itself are the series of battery-powered supplies, which are activated just prior to the probe's entering the Jovian atmosphere. The instruments and experimental objectives of the probe were provided in Table B-3.

3.0 LAUNCH VEHICLE

The launch vehicle description, summarized in Reference 8, is given essentially verbatim below.

3.1 General Description

The launch vehicle for the Galileo Mission consists of two main components, the Space Shuttle and the Centaur, designed to boost the spacecraft into an Earth-escape trajectory. The Space Shuttle, the tracking and communications relay satellites, the Kennedy Space Center and Johnson Space Center control centers, the Goddard Space Flight Center, and the worldwide space tracking and data network form the Space Transportation System (STS).

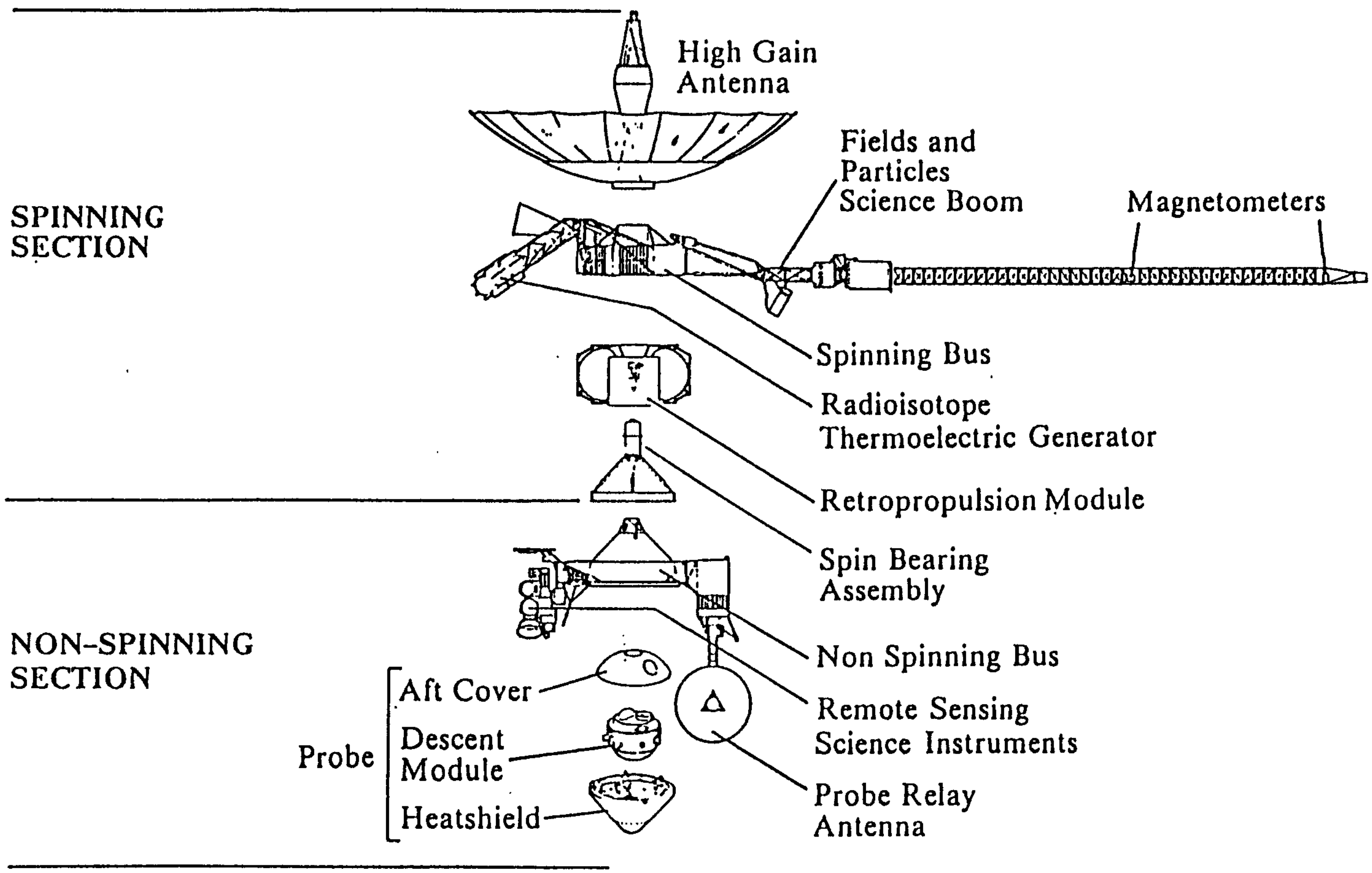


FIGURE B-14 (continued): Exploded view of the Galileo spacecraft showing major components for both spun and despun sections.

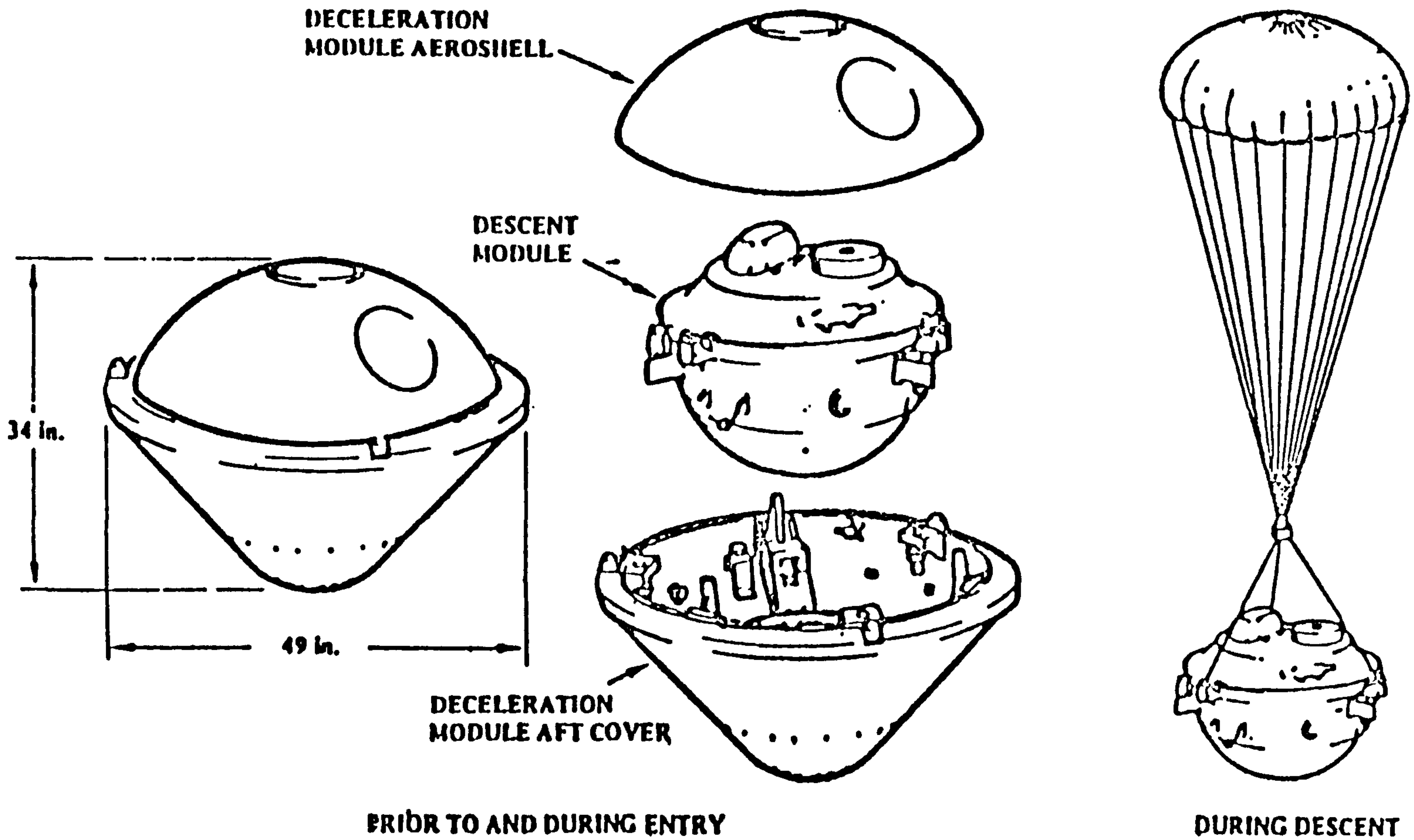


FIGURE B-15: The major components of the GLL probe are illustrated above.

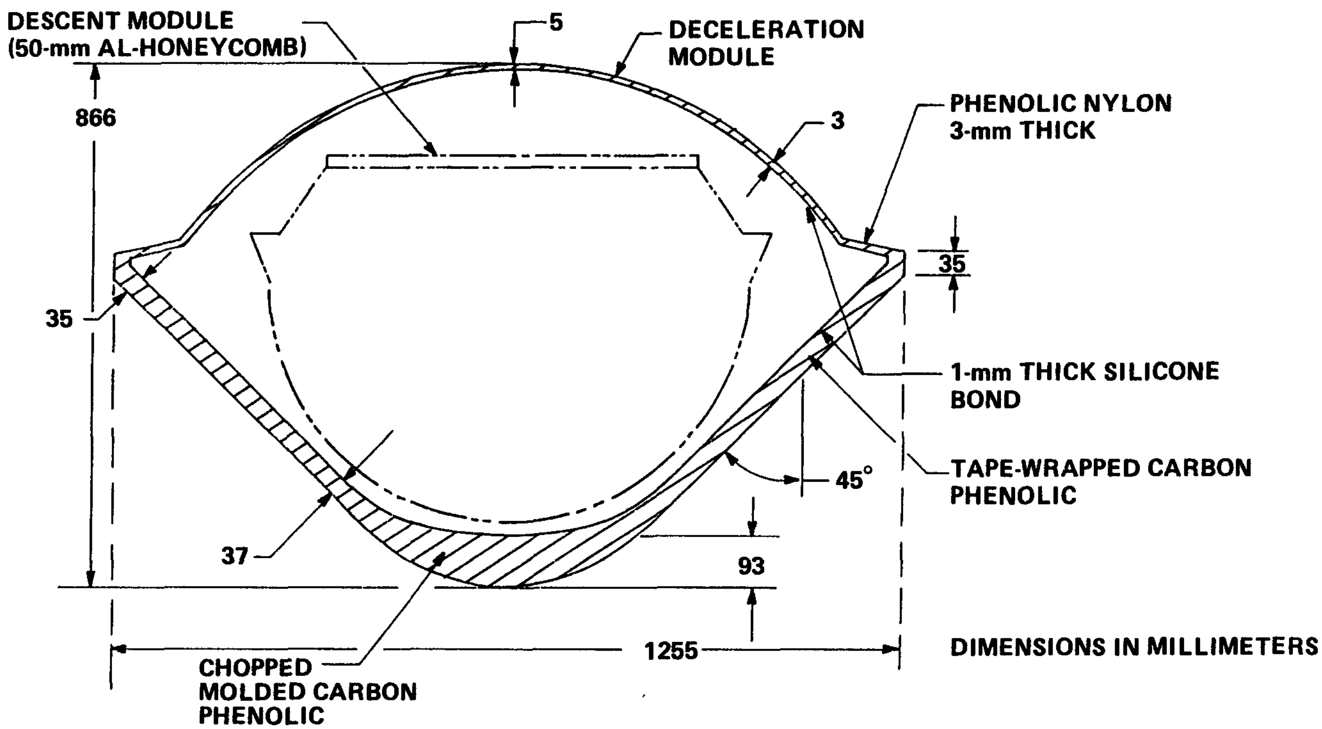


FIGURE B-16: The 34 LWRHUs inside the probe are quite well protected by the deceleration module (shaded area) structure.

3.2 Space Shuttle

The basic mission of the Space Shuttle is to be a space transport carrying payloads into space and returning to Earth for another load. The Shuttle consists of the Orbiter, the External Tank (ET), and the Solid-Rocket Boosters (SRBs). The arrangement of these components is shown in Figures B-17 through B-20.

Figure B-21 shows the basic mission cycle for the Space Shuttle. It starts with the launch of the Shuttle vehicle (with the SRBs and Orbiter main engines firing). After approximately 2 min (about 37 km altitude), the SRBs are jettisoned and fall back to Earth for recovery. After approximately 8 min of flight (before the Orbiter attains orbit), the main engines are shut down, and the ET is separated and falls back to Earth. The orbital maneuvering engines thrust the Orbiter into orbit. For these missions, the orbital operations would include ejecting the Centaur spacecraft from the Orbiter. The Orbiter would subsequently return to Earth to be made ready for its next flight.

3.2.1 Main Engines - The main propulsion subsystem of the Orbiter consists of three Space Shuttle Main Engines (SSME) which burn for approximately 8 min from just before lift-off until slightly before attaining orbit. The SSMEs are reusable, high-performance, liquid-propellant rocket engines with variable thrust. The propellants, LH_2 (fuel) and LOX (oxidizer), are carried in the external tank (ET). Each engine weighs about 2860 kg (6300 lb) and has a rated thrust of 1.67×10^6N (375,000 lb) at sea level and 2.09×10^6 N (470,000 lb) in vacuum. The nozzle is gimbaled with hydraulic actuators for steering.

3.2.2 Solid-Rocket Boosters The signals to light the two solid-rocket boosters (SRBs) will be sent 40 ms prior to lift-off. The initiation of these signals is dependent upon normal operation of the SSMEs with at least 90 percent of rated thrust. The simultaneous operation of the SRBs and the main engines, which are started in sequence about 6.6 s before lift-off, will cause lift-off to occur.

Each SRB provides 12.9×10^6 N (2.9×10^6 lb) of thrust at sea level and contains 505,000 kg (1.11×10^6 lb) of propellant. The propellant is a composite-type solid propellant formulated of polybutadiene acrylic and acrylonitrile terpolymer binder, ammonium perchlorate, and aluminum powder with a small amount of iron oxide added as a burning rate catalyst. The nozzle of the SRB is gimbaled with hydraulic actuators for steering.

Figures B-22 and B-23 show the SRB components and overall construction.

3.2.3 Orbital Maneuvering Subsystem - The Orbital Maneuvering Subsystem (OMS) includes two engines, located in pods at the top of the aft fuselage on either side of the vertical tail. The arrangement of these OMS pods is shown in Figure B-24.

These engines provide thrust for carrying the Orbiter into orbit after the main engines are shut down, for maneuvering while in orbit, and for providing retrothrust to retard the Orbiter out of orbit for reentry. The rated (vacuum) thrust of each engine is 26,700 N (6000 lb). The fuel is

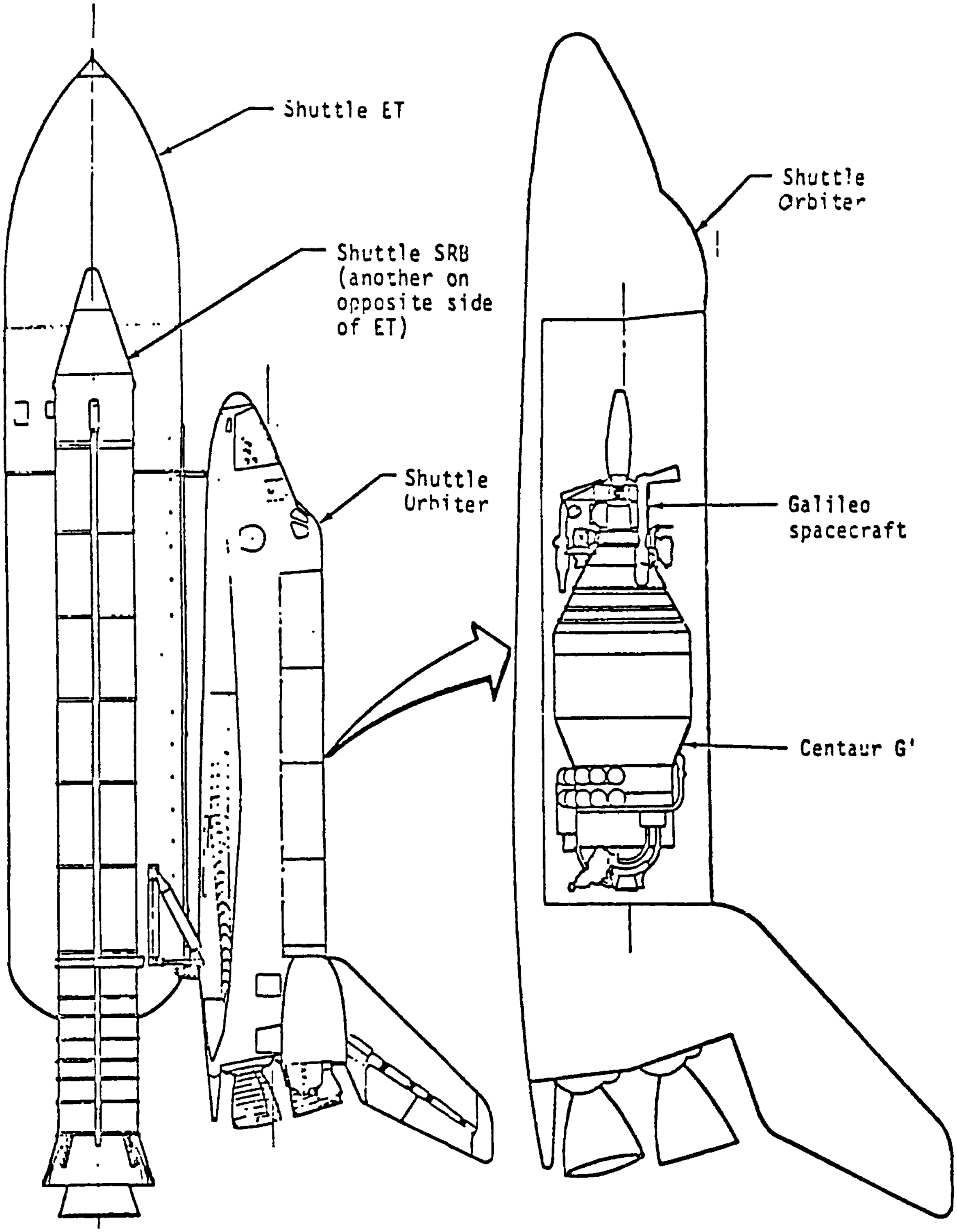


FIGURE B-17: The Galileo launch configuration is shown schematically above.

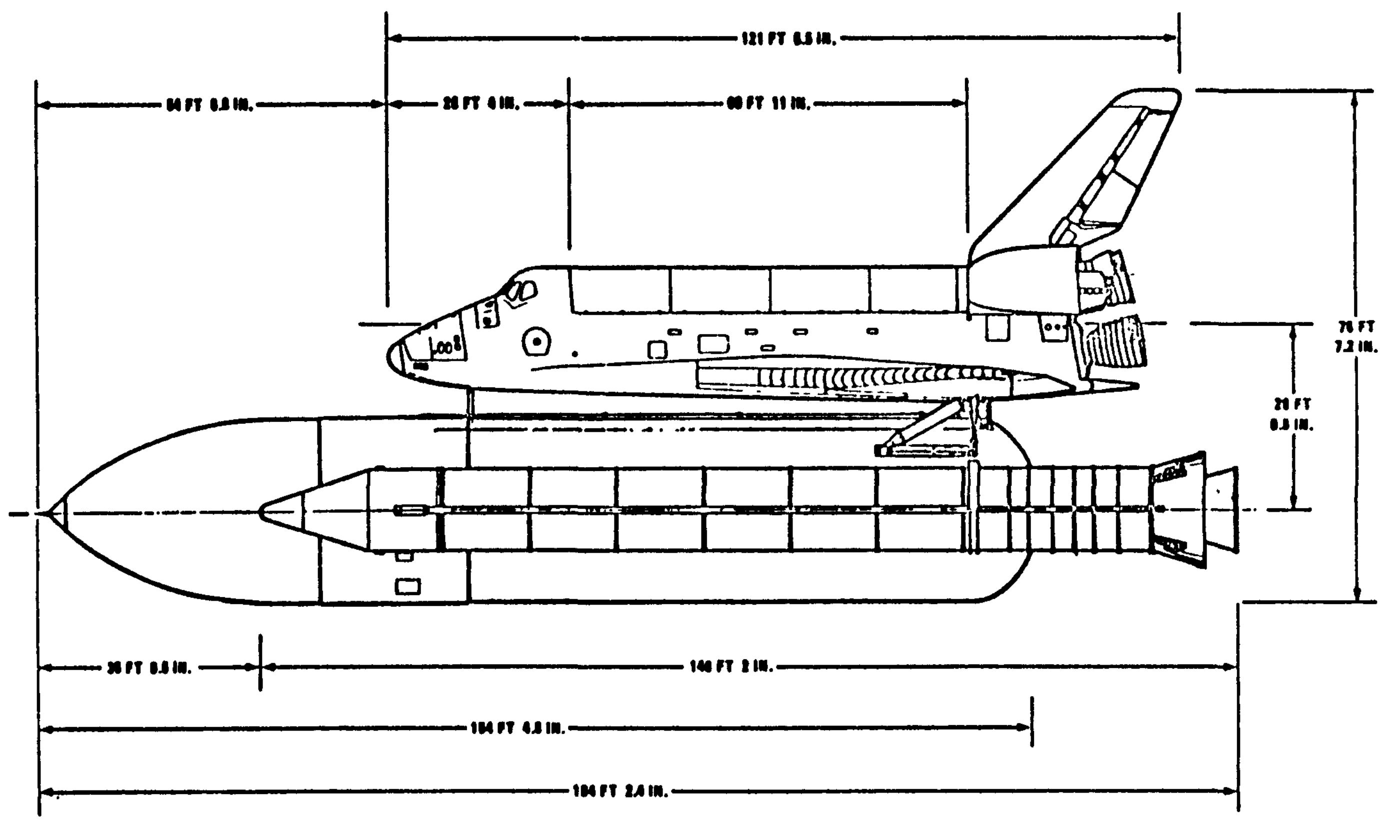


FIGURE B-18: Shuttle vehicle, side view.

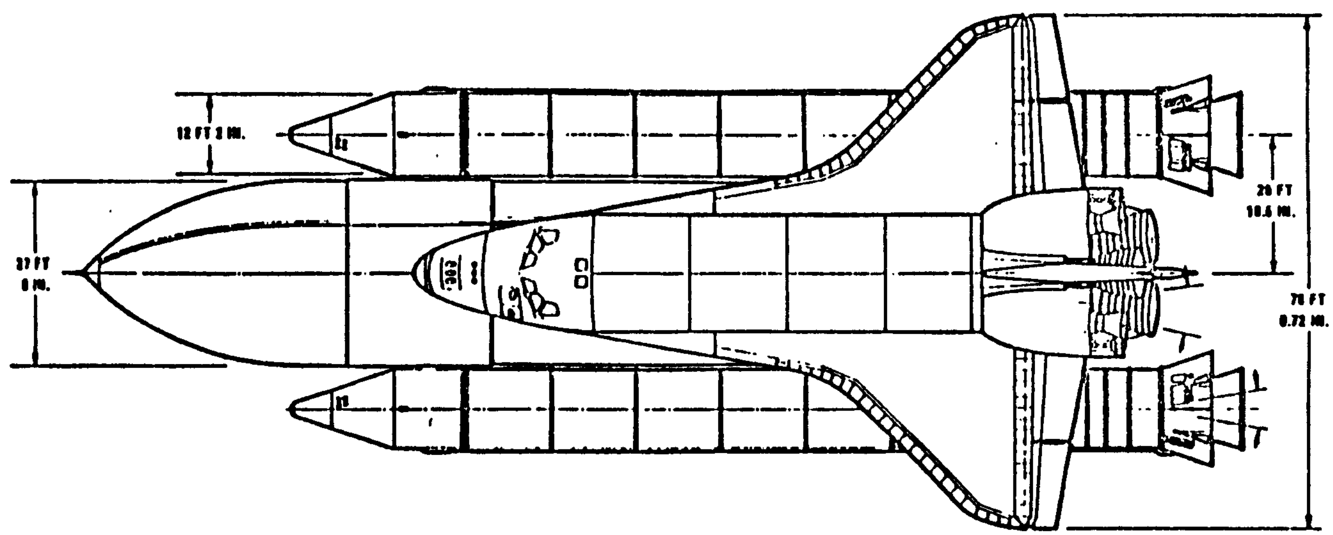


FIGURE B-19: Shuttle vehicle, top view.

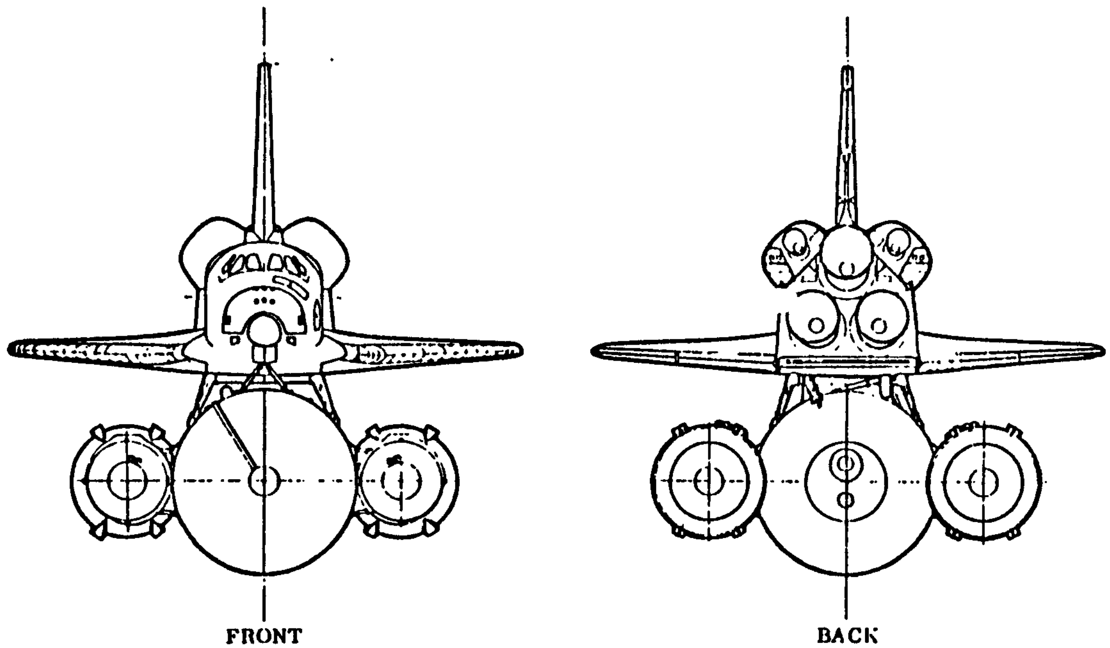


FIGURE B-20: Shuttle vehicle, front and back views.

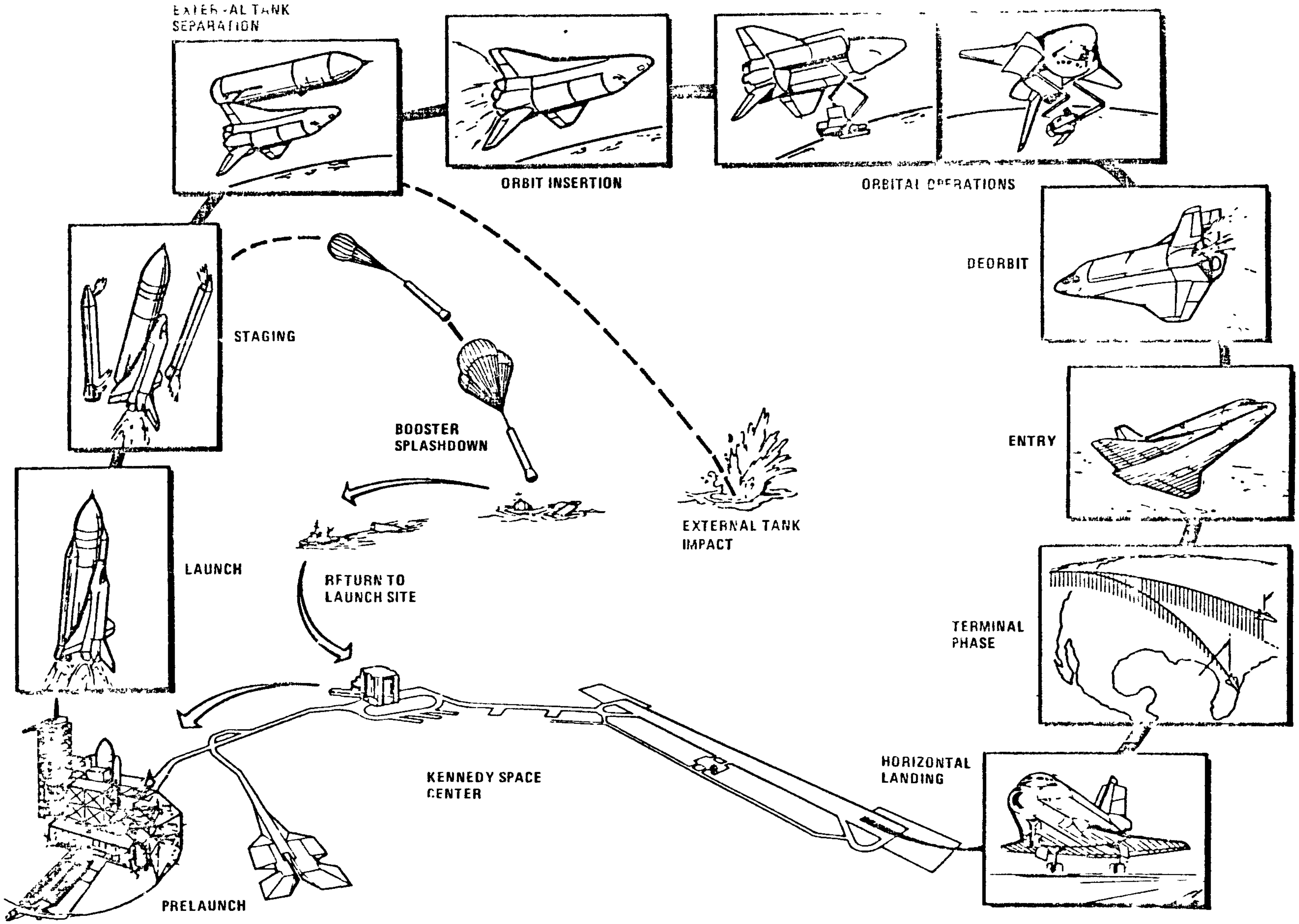


FIGURE B-21: Basic mission cycle for shuttle.

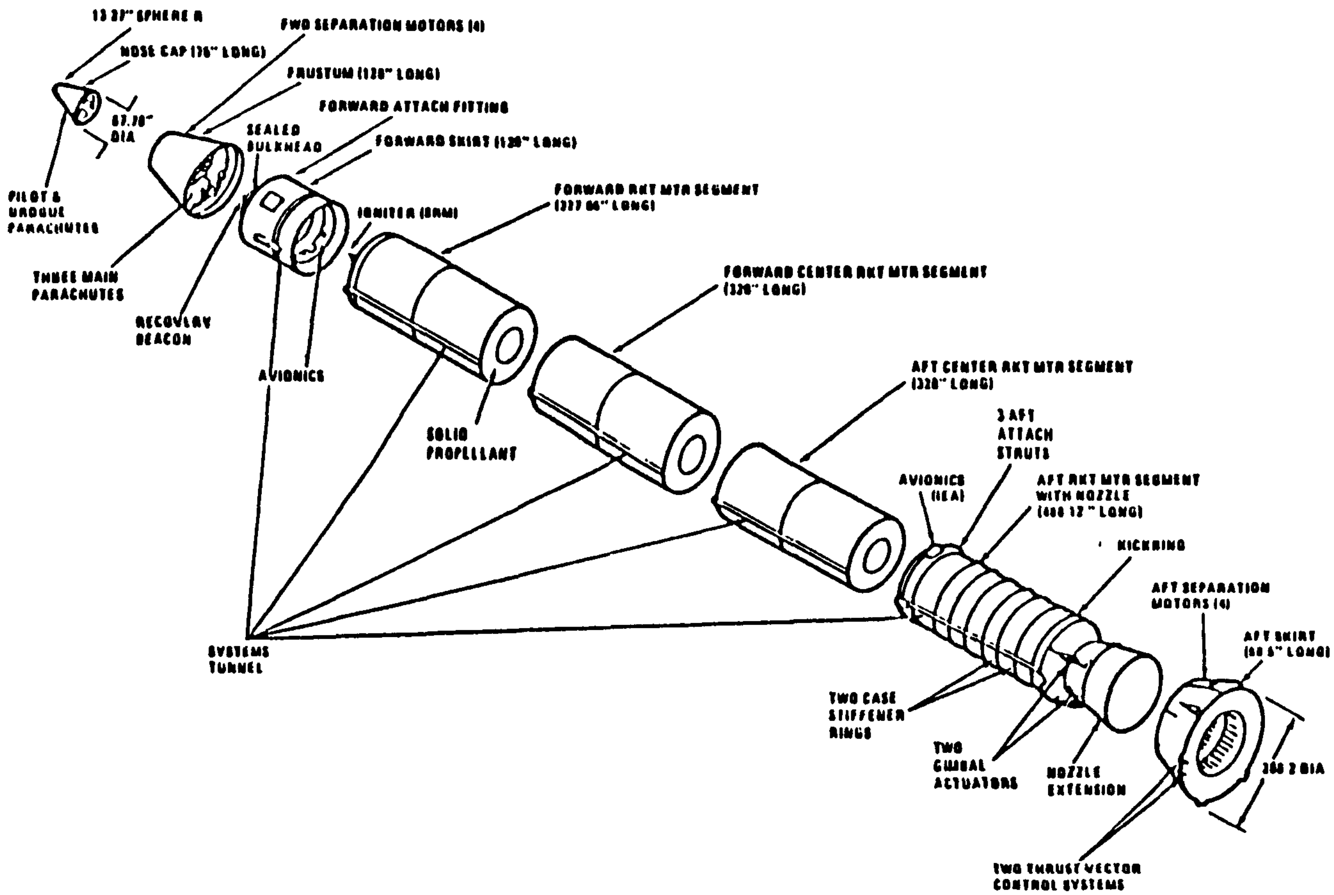


FIGURE B-22: Solid-rocket booster parts.

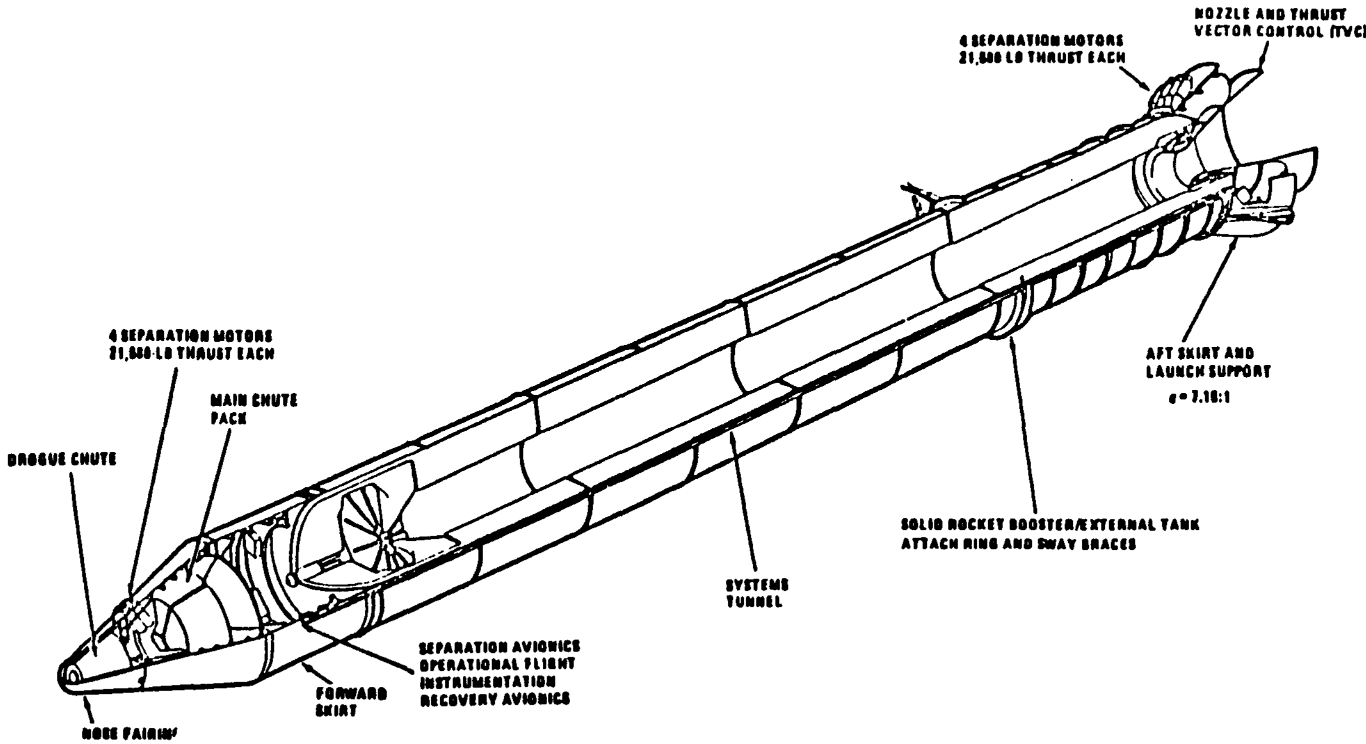


FIGURE B-23: Shuttle solid-rocket booster.

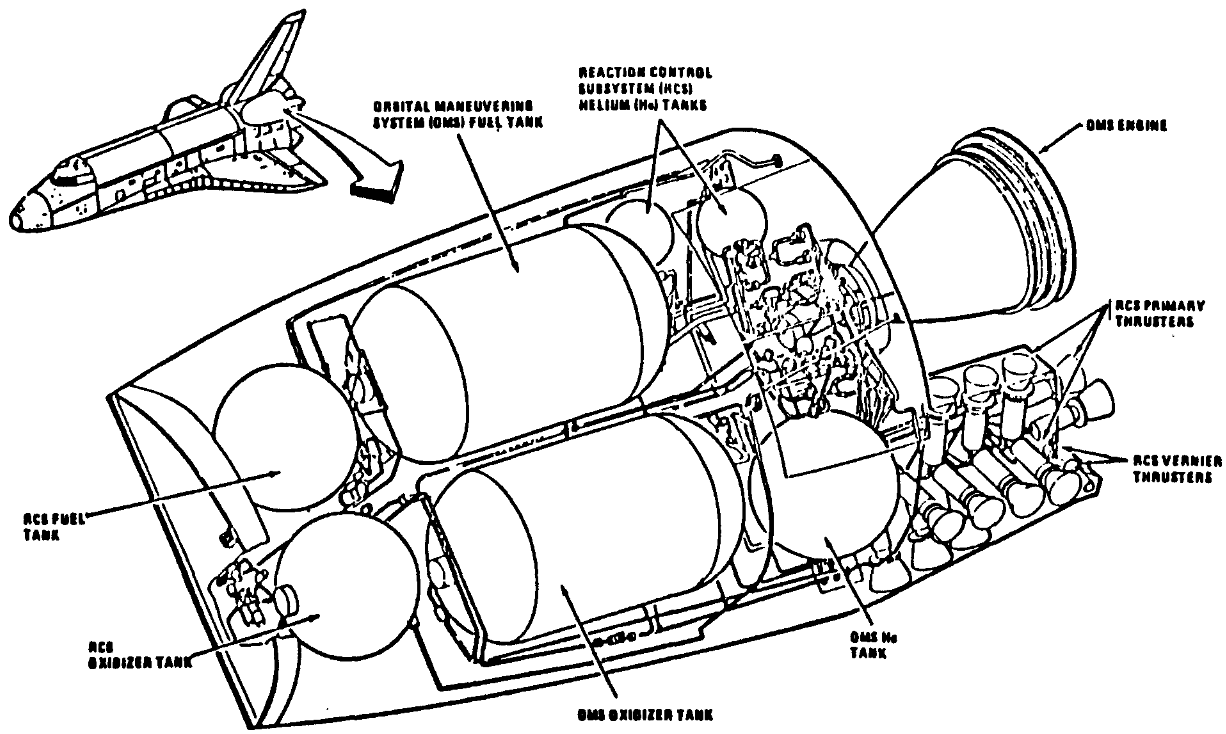


FIGURE B-24: Orbital maneuvering subsystem engine pod.

monomethylhydrazine (MMH), and the oxidizer is nitrogen tetroxide (N_2O_4). The nozzles are gimbaled for steering with electromechanical actuators.

3.2.4 Orbiter Reaction Control System - The Reaction Control System (RCS) includes 44 thrusters (38 primary, 6 vernier) for altitude control and for three-axis translation during orbit insertion, on-orbit, and entry phase of the Orbiter flight. The RCS propellants are monomethyl hydrazine and N_2O_4. The thrust (in vacuum) is 3870 N (870 lb) for each primary thruster and 107 N (24 lb) for each vernier thruster.

3.2.5 External Tank - The External Tank (ET) supplies the Orbiter main propulsion system with liquid hydrogen (LH_2) and liquid oxygen (LOX). The ET contains 703,000 kg (1.55 X 10^6 lb) of usable propellant for main engines at lift-off. At Main-Engine Cutoff (MECO), the ET is separated from the Orbiter before orbital velocity is achieved. The ET then proceeds on a ballistic reentry for a safe impact in the ocean. Figure B-25 shows the ET construction.

4.0 CENTAUR

The Centaur-G upper-stage vehicle is a wide-body version of the standard Centaur booster which was developed by General Dynamics/Convair specifically as a high-performance upper stage for Shuttle-Launched missions such as Galileo. The Centaur system is composed of the booster vehicle and other airborne-support equipment used in deployment and in the operational interfaces with the Orbiter. The complete Centaur system is shown schematically in Figures B-26 and B-27.

The Centaur upper-stage vehicle structure consists of a 3-m (10-ft) diameter liquid oxygen tank that tapers into a 4.3-m (14-ft, 2-in.) diameter liquid hydrogen tank. The propulsion system for the Centaur booster is composed of two gimbaled Pratt & Whitney RL10A-3-3A liquid-fueled engines developing 147 kN (33,000 lb) of total thrust. The overall length of the Centaur-G vehicle, without the Spacecraft interface, is 8.8 m (29.1 ft). The cryogenic fuel tanks are insulated with combinations of helium-purged foam blankets and insulation shields. Forward and aft adapter sections provide mounts for the Spacecraft interface; for on-board Centaur Avionics/electronic packages; and for propulsion, attitude control, and hydraulic systems. The Centaur avionics system performs the functions necessary for autonomous control of the Centaur vehicle after deployment from the Shuttle Orbiter.

The airborne-support equipment, designated the Centaur Integrated-Support System (CISS), provides the structural, fluidics, and avionics support in the payload bay prior to deployment and is returned to Earth in the Orbiter as a reusable Centaur system. The CISS consists of a Centaur Support System (CSS), a deployment adapter, and associated electronics and fuel lines.

4.1 Vehicle Structure

The Centaur vehicle structure is composed of the propellant tanks and adapters, which serve as structural interface support and transition

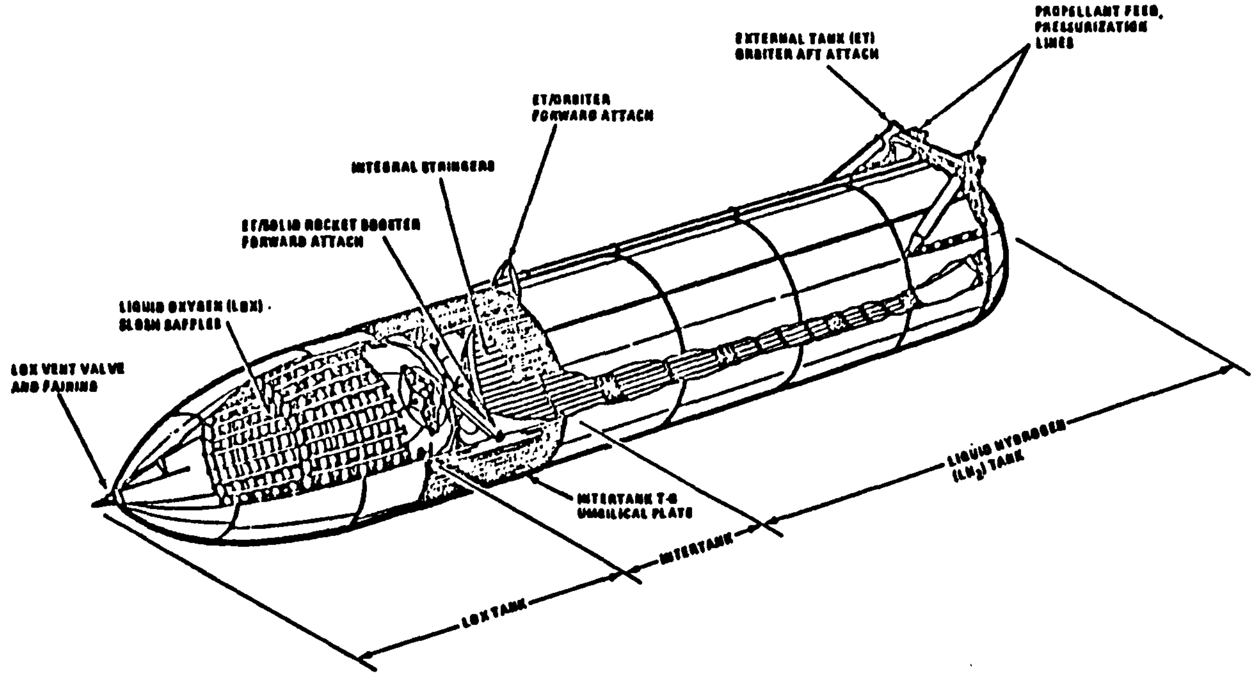


FIGURE B-25: Shuttle external tank.

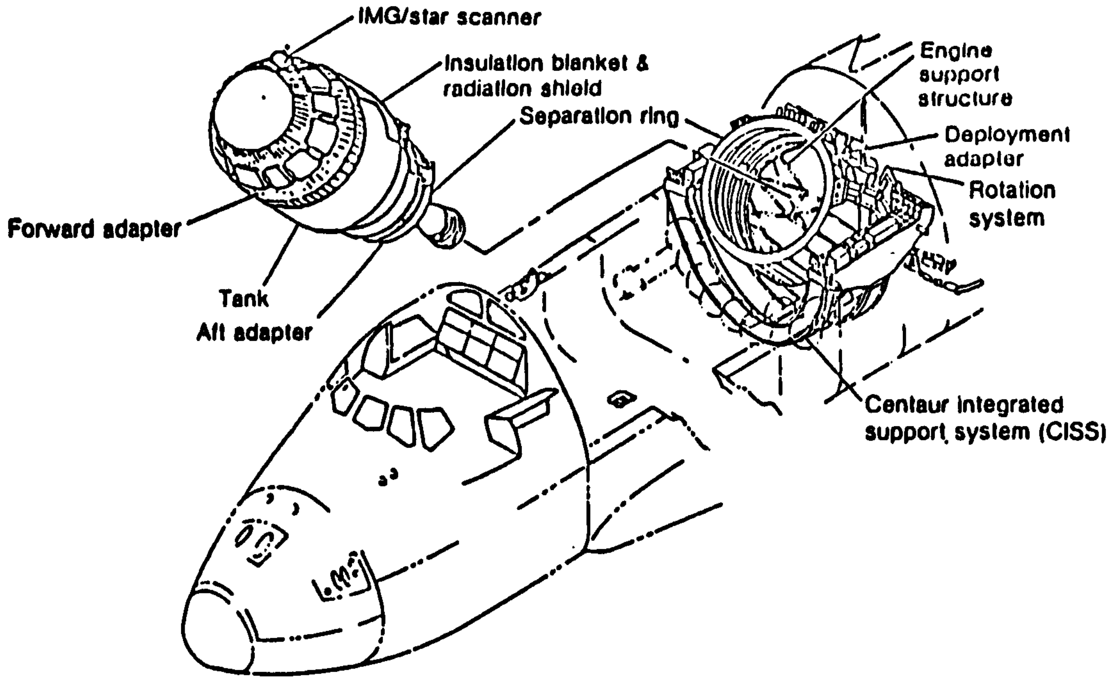


FIGURE B-26: Shuttle/Centaur system summary.

members. Besides containing the main engine propellants (liquid oxygen oxidizer and hydrogen fuel), the tanks also provide primary structural integrity of the vehicle and support vehicle systems and components.

The liquid hydrogen tank consists of a 4.3-m (14.2 ft) diameter cylindrical section closed by an ellipsoidal forward bulkhead and a conical aft bulkhead, as shown in Figure B-27. A conical forward adapter section is attached to the forward bulkhead and serves as a mount primarily for the various avionics boxes, as shown. The forward face of the conical adapter section provides the structural interface for mounting the Spacecraft. The conical aft bulkhead of the hydrogen tank is attached to the oxygen tank at the 3-m (120-in.) diameter joint between cylindrical and forward ellipsoidal bulkhead. Also, the fuel and oxidizer tank are separated by a double-walled intermediate bulkhead to prevent mixing in the event of a tank rupture. Structural integrity of the tanks is provided in pressurization, either supplied by external sources or maintained by propellant boil-off when fueled. The aft bulkhead has welded brackets and fixtures which support propellant lines, helium and hydrazine bottles, the main engines, electrical boxes, and radiation shields.

The two propellant tanks are insulated to prevent excessive propellant boil-off. The forward bulkhead insulation for the liquid hydrogen tank consists of a three-layer radiation shield and a two-layer foam blanket. The tank sidewall insulation comprises a radiation shield and two layers of foam insulation extending along the tank. The liquid oxygen tank has an aft bulkhead and sidewall radiation shield but no foam insulation bulkhead. The equipment compartments and insulation blankets are purged with helium to provide a dry inert atmosphere within the insulation during fuel loading.

The aft adapter is a cylindrical graphite/epoxy structure with attachment rings at each end. This adapter provides the interface with the separation system and distributes the support loads into the Centaur tank structure. The forward ring bolts to the oxygen tank and to the aft ring are attached to the separation ring. The aft adapter is also used to mount the vehicle separation springs, field disconnect panels, radiation shields, and wiring.

4.2 Vehicle Propulsion, Hydraulic, and Pneumatic Systems

Primary Centaur thrust is provided by two Pratt & Whitney RL10A-3-3A liquid-fueled, regeneratively cooled engines. Turbopumps are used to inject the propellant into this, the the throat chamber, where combustion is initiated by an electric spark igniter. Redundant sets of pyro valves installed in the propellant feedlines provide two failure tolerances against inadvertent engine operations. The pyro valves are opened automatically after the Centaur vehicle has performed proper orbital orientation maneuvers to get in position for a main engine burn.

After deployment from the Shuttle Orbiter, the Reaction Control System (RCS) is used for spin control and orientation maneuvers. The Reaction Control System is a system of small rocket engines mounted on the periphery of the Centaur liquid oxygen tank. This engine system provides thrust for attitude control, for settling of propellants, and for making separation and orientation maneuvers. The system consists of twelve-27-N (6 lb) thrust units, a positive expulsion tank with 77 kg (170 lb) hydrazine capacity, two parallel sets of pyro valves, one fill/drain valve, two pneumatic checkout valves, and heated feedlines. The pyro valves are fired open, pressurizing the system and allowing hydrazine flow to the thrusters after Centaur is deployed a safe distance from the Orbiter. The arming mechanism is provided by the dual-failure tolerance arm/safe sequencer and is two-failure tolerant against inadvertent operation.

Two identical and separate hydraulic power supply systems provide the force to gimbal each of the main engines. A power package assembly and two actuators are the main components of the system. Each power package contains an engine-driven pump that supplies pressure to the actuators. This pump is coupled to the engine turbine drive and operates during the burn phase when the engines are firing. During the coast phase, a recirculation pump circulates hydraulic fluid through the system.

The pneumatic system of the Centaur vehicle consists of the propellant tank pressurization system, pneumatically actuated valve control, system purge systems, helium supply, and the intermediate bulkhead relief system. All of these individual pneumatic systems with the exception of the bulkhead relief system are interconnected and function as a single system.

The pressurization system consists of valves, tubing, and components for pressurizing the propellant tanks. The Digital Computer Unit (DCU) controls helium pressure before engine starts and for liquid oxygen tank pressurization during engine burns. The liquid hydrogen tank is pressurized during engine burns by gasesous hydrogen diverted from the main engines. Tank pressures are monitored by the DCU from outputs of redundant transducers in each tank. Preprogrammed logic defines the desired pressure level in the tanks and sequences for pressure changes throughout the mission. Before the Centaur is deployed, pressurization of the two propellant tanks is controlled by the five computer-control units located on the CISS.

Other pneumatic system functions include (1) a helium purge system to remove impurities from various system lines, especially during prelaunch and post-abort landings; (2) a vent system for redundant ground, ascent, and on-orbit venting of propellant tanks to maintain safe pressure levels; and (3) a fill/dump system to ensure Centaur compatibility with all Shuttle Orbiter abort modes that occur before deployment. Most of the pneumatic system functions are provided prior to deployment with interconnecting systems in the CISS.

Pyrotechnic devices are used to preclude premature RF antenna deployment. After Centaur separation, the RF antennas, which are spring-loaded and located on the equipment module, are released by the firing of redundant pyrotechnic pin pullers. Spacecraft separation involves a pyrotechnic deployment sequence, including firing of a Super Zip detonation cord and actuation of springs to accelerate the Spacecraft away from the Centaur.

4.3 Centaur-Integrated Support System (CISS)

4.3.1 CISS Mechanisms - The rotation system and umbilical detachment mechanisms are two mechanical system installations on the CISS consisting of identical primary and backup deployment adapter rotation systems (Figure B-26). Both the primary and the backup system are tolerant of single failures of the drive motors and clutches, and either can rotate the deployment adapter under maximum expected loading conditions. The rotation system rotates the Centaur from the stowed position in the cargo bay to 45° for deployment. To start deployment rotation, the latches retaining the forward portion of the Centaur are released, and electrical power is supplied to one of the primary motors. The motor clutches engage, and torque is transmitted to the crank which rotates the deployment adapter.

Should the primary motor fail to operate, the primary motor will be de-energerized, and power will be applied to the Primary Motor No. 2. If deployment is aborted, the Centaur is rotated back to the stored position by reversing the drive-unit direction. The forward latches of the Centaur are then relatched, and the Centaur cryogenic propellants are dumped prior to reentry and landing.

In the deployment sequence, the Super Zip is fired, and deployment springs accelerate the vehicle away from the deployment adapter at a Centaur-to-Orbiter separation velocity of 1 ft/s. This motor causes separation of the umbilical panel disconnects for the fill, drain, dump, and vent lines mounted on the panels. Following Centaur deployment, the payload bay may be closed with the deployment adapter in any position between 0 and 45°.

4.3.2 Centaur Support Structure (CSS) - The Centaur vehicle and spacecraft are supported within the Orbiter by the CSS, which remains within the Orbiter after deployment. The support structure consists of semicircular aluminum beams connected by longitudinal side beams and a keel beam. This construction supports the deployment adapter to which the aft end of the Centaur is bolted. Loads are transferred from the CSS to the Orbiter by steel trunnion pins. Additional CSS trunnions extend from the Centaur equipment module to latch the forward portion of the Centaur to the Orbiter. The CSS also provides support for helium storage bottles and the CISS avionics system.

The deployment adapter is a cylindrical structure 3 m (10 ft) in diameter and 1.1 m (44 in.) high, which supports the aft end of the Centaur. During the Centaur deployment sequence, the deployment adapter rotates about the CSS trunnion support pins to bring the Centaur to the desired separation attitude. The deployment adapter also includes the separation ring and also supports the fluid system and electric disconnect panels. The CISS electronics hardware is also supported on the deployment adapter.

The separation ring portion of the deployment adapter contains the pyrotechnic Centaur separation mechanisms designated as the Super Zip system. The separation ring is a 3 m (10-ft) diameter by 130 mm (5-in.) long aluminum cylinder. The ring bolts to the aft adapter of the Centaur and the CISS Deployment Adapter. When the pyrotechnic Super Zip explosive cord is fired, the separation ring is fractured, and the spring system pushes the

Centaur away from the Orbiter. Should both the primary and backup Super Zip pyrotechnic devices fail to operate, the Centaur would be lowered back into the payload bay.

4.3.3 CISS Avionics System - The CISS includes avionics support as well as structural support for the Centaur within the Orbiter. The CISS avionics system provides all Centaur-to-Orbiter electrical interfaces, computer control for all safety functions prior to deployment, electrical power, and power control, instrumentation and telemetry, pyrotechnic control for Centaur separation, and monitoring of propellant tanking levels.

The computer control avionics subsystem comprises five microprocessor control units, two control distribution units, and a digital computer unit. The function of the subsystem is to actively control all safety-related Centaur vehicle operations up through deployment such as sequencing, pressurization and vent control, abort operations, purge system control, pneumatics, system control, and deployment operations. Each of the safety functions, such as tank pressurization, is monitored independently by each control unit. Commands for a function by the control units are in majority agreement. A Centaur DCU located on the CISS provides Pulse Code Modulating (PCM) data from all control units to the ground and to the Orbiter. The DCU is also used in the instrumentation and telemetry subsystem.

Electrical power for the CISS and Centaur prior to deployment is supplied by the Orbiter. In the event of failure of Orbiter power, power could still be supplied from the two silver-zinc batteries located on the CISS. The electrical distribution unit provides power control for both Centaur and CISS loads.

The CISS instrumentation and telemetry systems record vehicle tank pressurization and avionics data and transmit these data to ground stations. Two remote multiplexing units and signal conditioners support CISS measurements. One Remote Multiplex Unit (RMU) provides a redundant path for channeling CISS measurements to the Centaur data tramsmitter.

4.3.4 CISS Pyrotechnic System - Pyrotechnic devices are employed in conjunction with certain CISS avionics functions. These devices are used for fluid isolation control and pyrotechnic deployment sequences. The CISS pyrotechnic functions include firing of the Super Zip detonation cord for Centaur separation and activation of helium purge pyrotechnic valves. The helium pyrotechnic valves provide the capability to supply tank insulation purge during an abort landing.

5.0 TRAJECTORY AND FLIGHT CHARACTERISTICS

The GLL launch will be from Pad 39B of the KSC Launch Complex. The sequence of events is provided in Table B-4. The reference trajectory upon leaving earth orbit is shown in Figure B-28 (Reference 18).

Table B-4: Typical Centaur/Planetary Mission Event Timeline with GLL's Specific Activities Identified

Event	Time (Hr:min:sec)
SSME Ignition	-0:00:06.6
SRB Ignition Command	0:00:00
First Motion/Lift-Off	0:00:00.24
Begin Roll Program (V=125 fps)	0:00:07
End Roll Program (V=300 fps)	0:00:15
Begin First Throttle Down (V=428 fps)	0:00:20
Begin Second Throttle Down (V=716 fps) (Optional)	0:00:31
Begin Throttle Up (V=1599 fps)	0:01:08
SRB Separation	0:02:05
3G Throttling Begins	0:07:39
MECO Command	0:08:34
Zero Thrust	0:08:40
ET Separation	0:08:52
OMS-1 TIG	0:10:34
OMS-1 Cuttoff	0:13:22
OMS-2 TIG	0:46:10
OMS-2 Cutoff	0:48:29
Open Payload Bay Doors (GLL)	1:23:00
Centaur Deploy (GLL)	6:41:24
Centaur Engine Ignition (GLL)	7:31:24

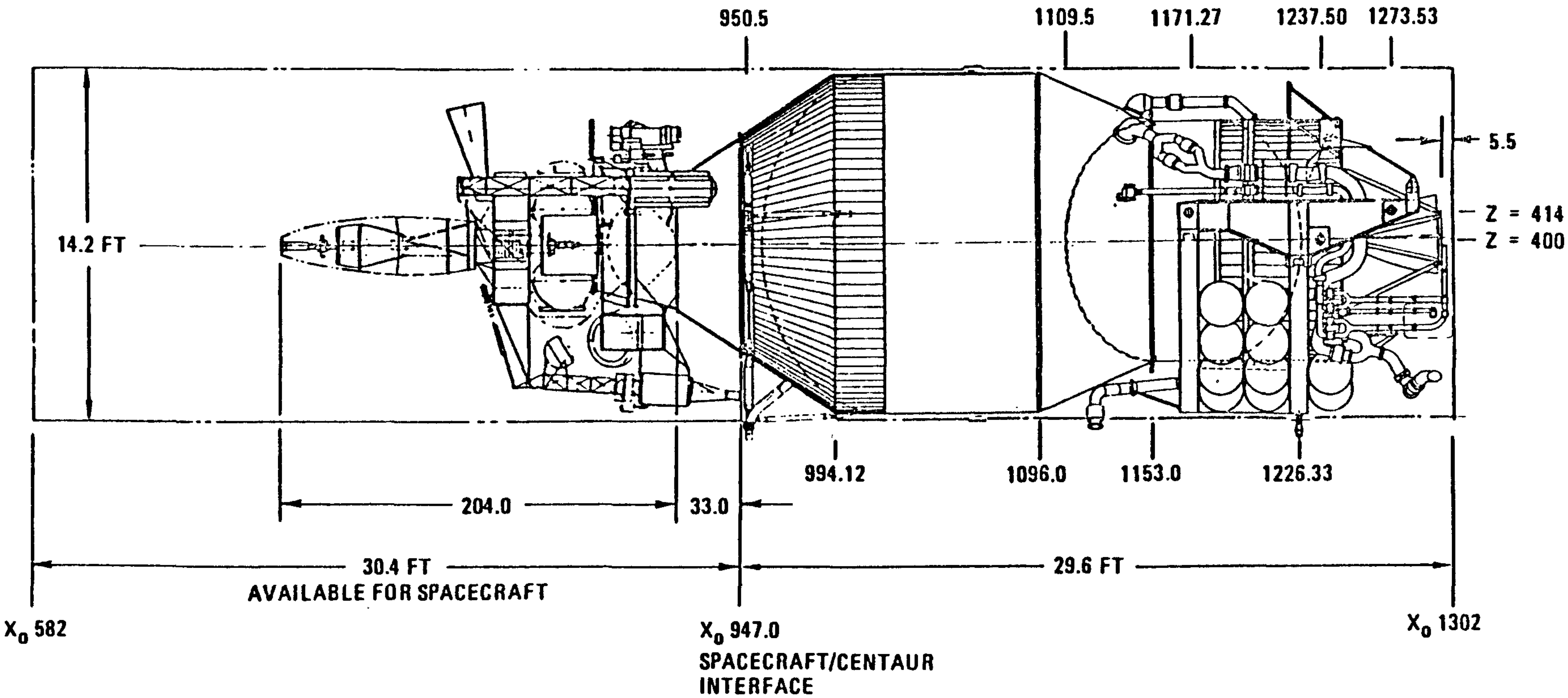


FIGURE B-27: Details of the Centaur G' vehicle are provided here. Highlighted are the LOX and LH_2 fuel tanks. Unless indicated differently, dimensions are given in inches.

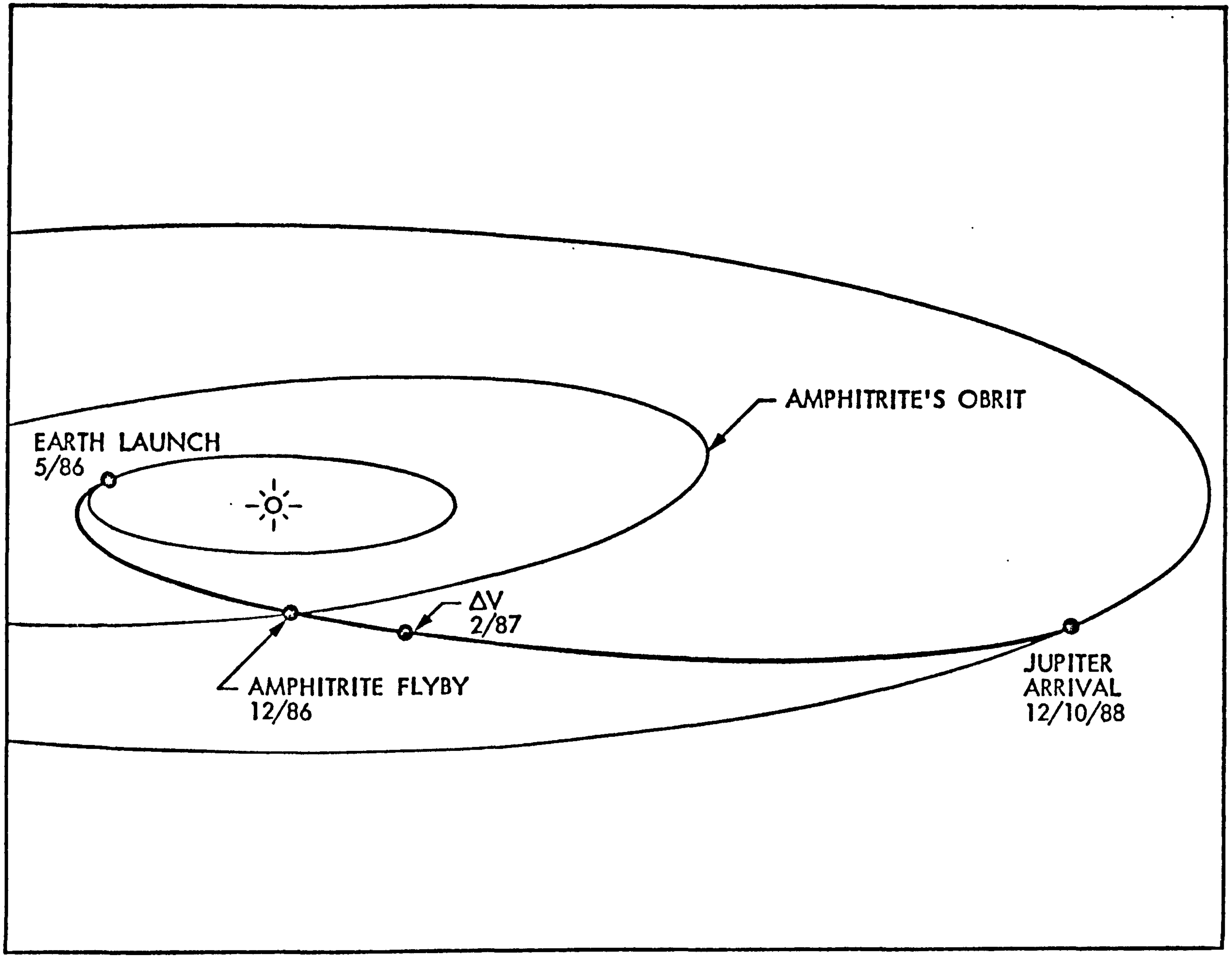


FIGURE B-28: The interplanetary trajectory of the GLL spacecraft with major milestones is shown above.

6.0 LAUNCH SITE

The Space Shuttle will be launched from Launch Pad 39B at the John F. Kennedy Space Center, Cape Canaveral, Florida, for the GLL Mission. Figure B-29 is a map of the Kennedy Space Center (KSC) and Cape Canaveral Air Force Base (CCAFB). The map shows the runway for orbiter landings, the vehicle assembly building area (where the shuttle will be mated on a mobile launch platform), and Launch Pad 39B (from which the Shuttle carrying the GLL payload will be launched).

Figure B-30 provides some added detail of the 39B Launch Pad.

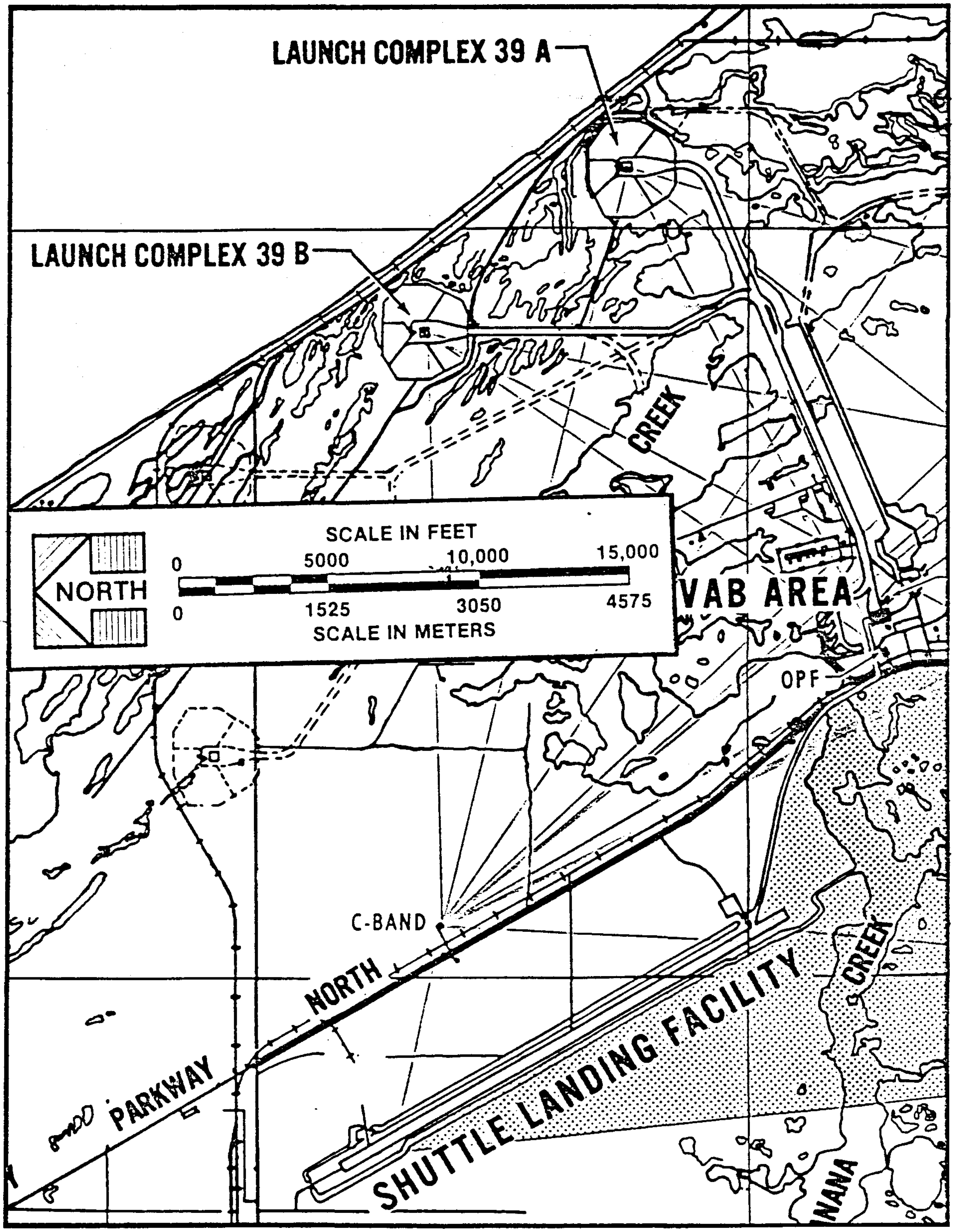


FIGURE B-29: The overall layout of Launch Complex 39B with respect to the VAB Complex and Shuttle Landing Facility is shown above.

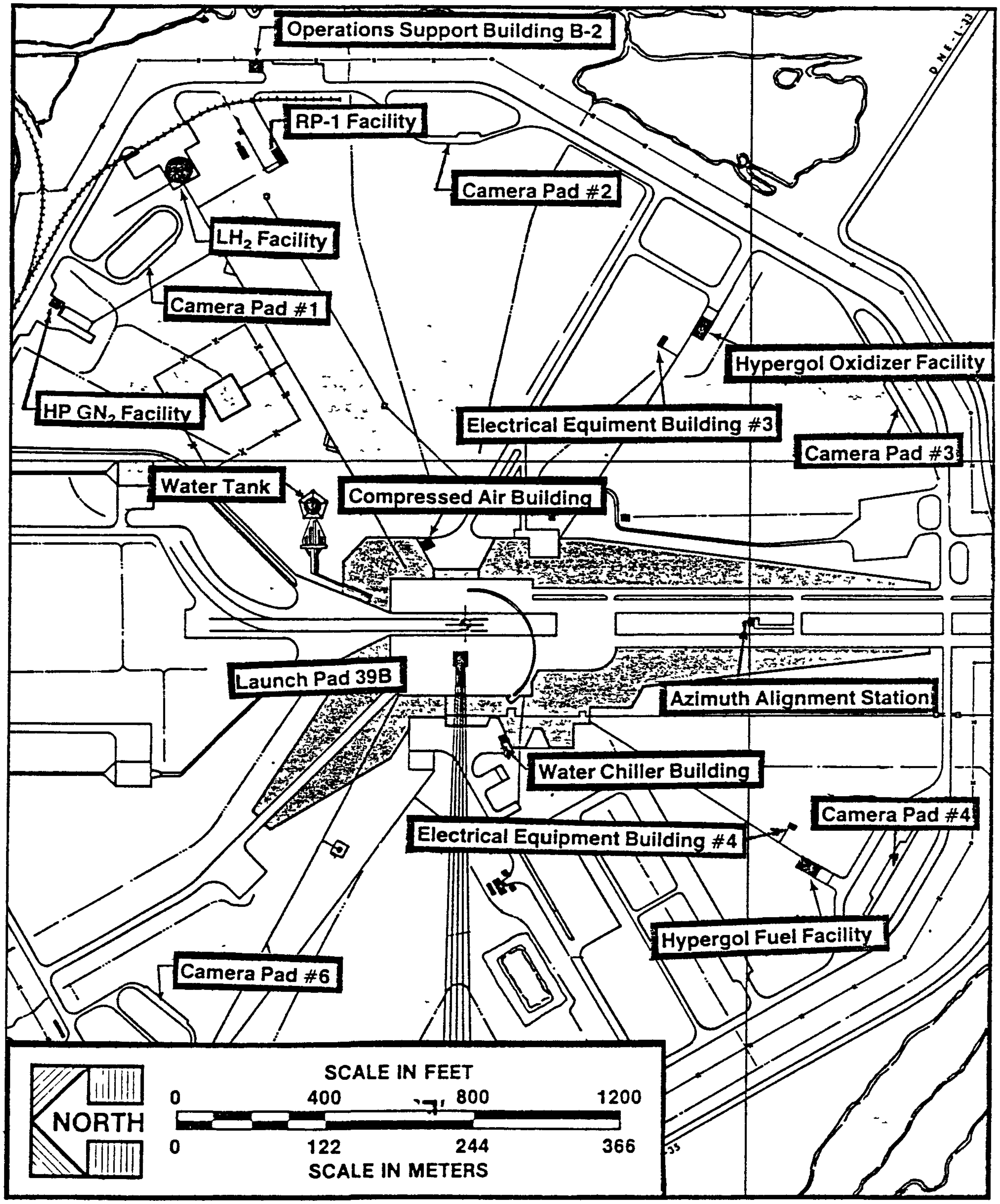


FIGURE B-30: Details of some of the significant attributes of Launch Complex 39B are shown above.

APPENDIX B-1

LWRHU DRAWINGS

The following are MRC's fabrication drawings for the LWRHU components:

AYC-790095, "Radioisotopic Heater Unit"

AYC-790096, "Clad Body Assy"

AYC-790097, "Clad Cap, Vent End Assy"

AYC-790098, "Frit"

AYC-790099, "Clad Cap, Vent End"

AYC-790100, "Clad Body"

AYC-790101, "Shim"

AYC-790105, "Clad Cap, Closure End"

AYC-790106, "Pellet, Fuel"

AYD-790379, "Aeroshell Assembly"

AYD-790380, "Aeroshell Body"

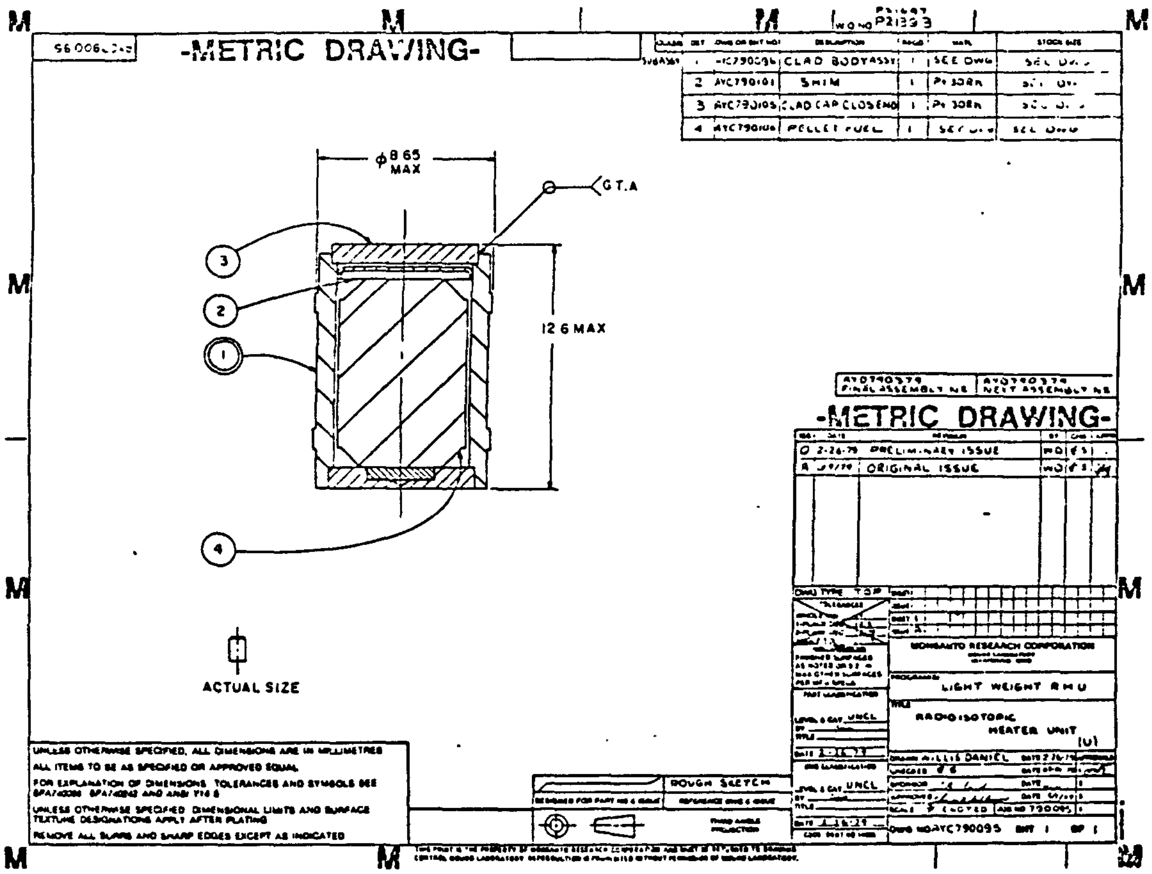
AYC-790381, "Insulator Cap"

AYC-790382, "Outer Insulator Body"

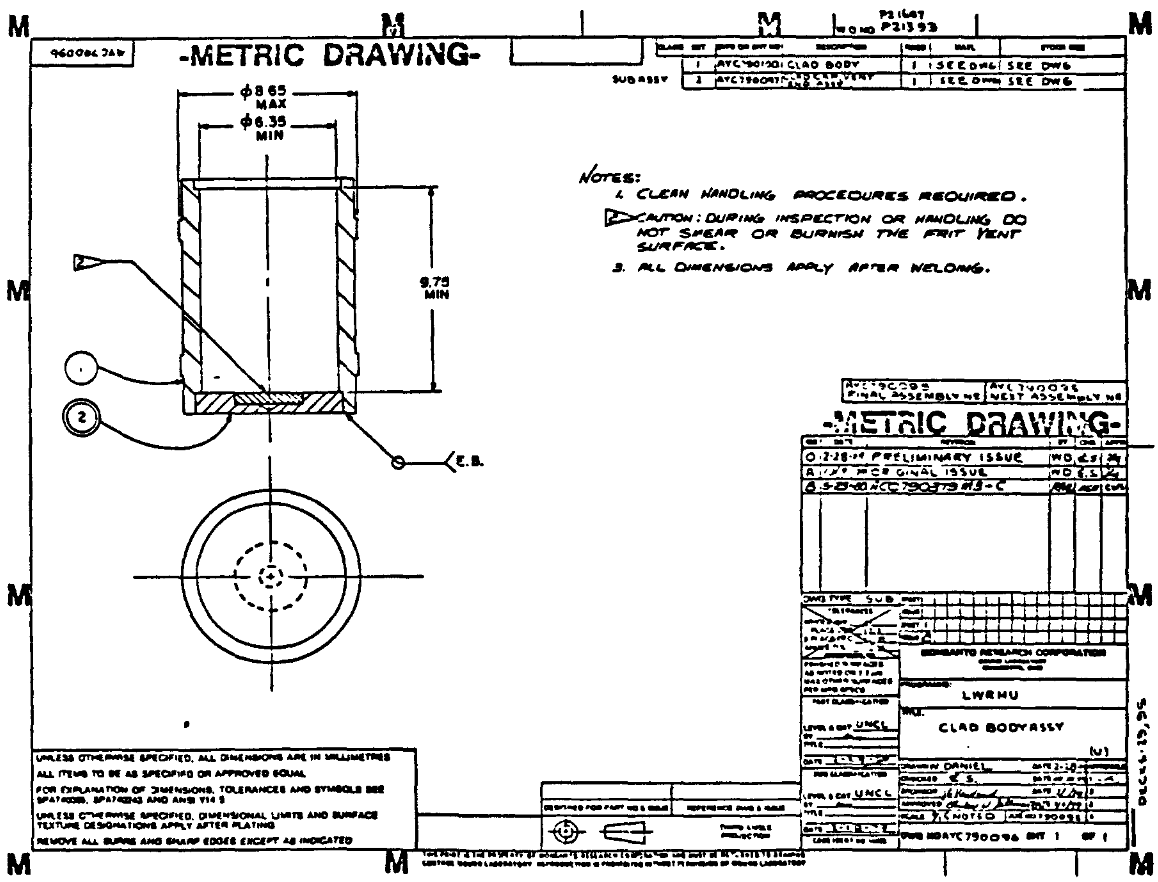
AYC-790383, "Middle Insulator Body"

AYC-790384, "Inner Insulator Body"

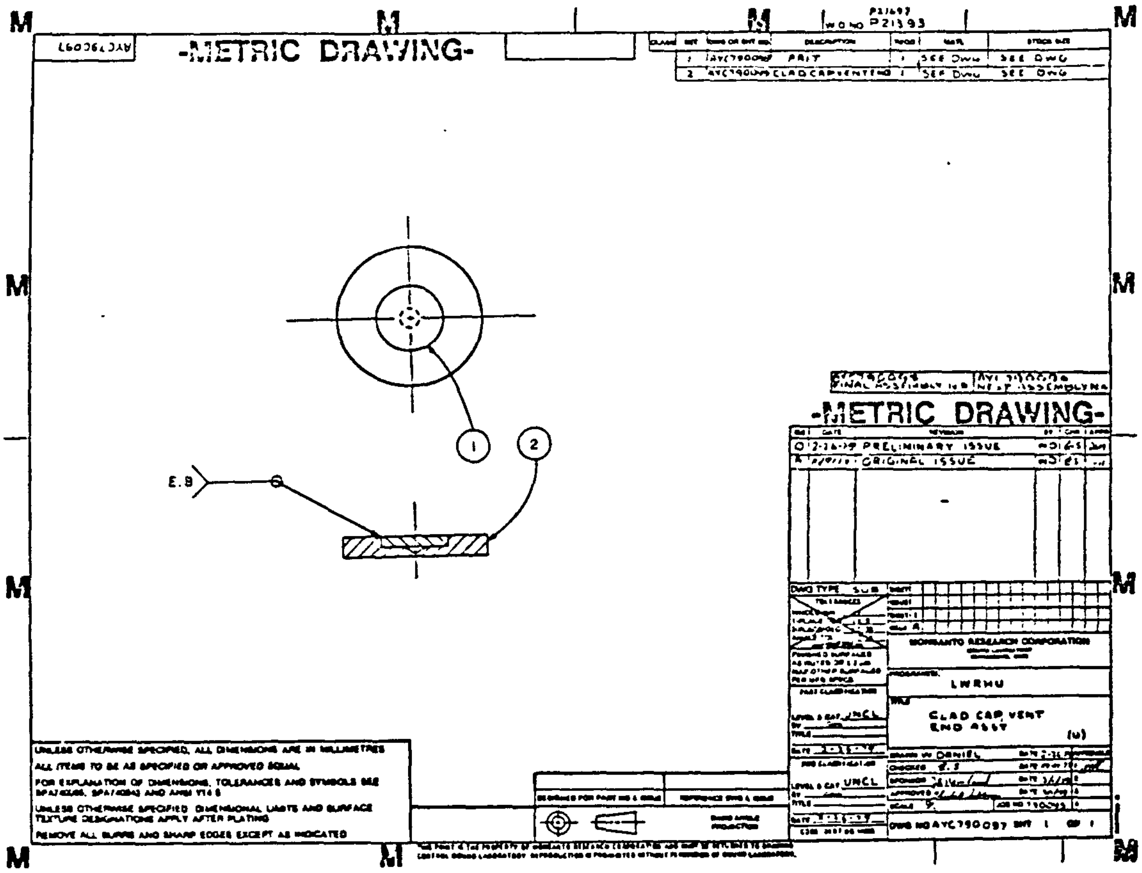
AYC-790385, "Aeroshell End Cap"

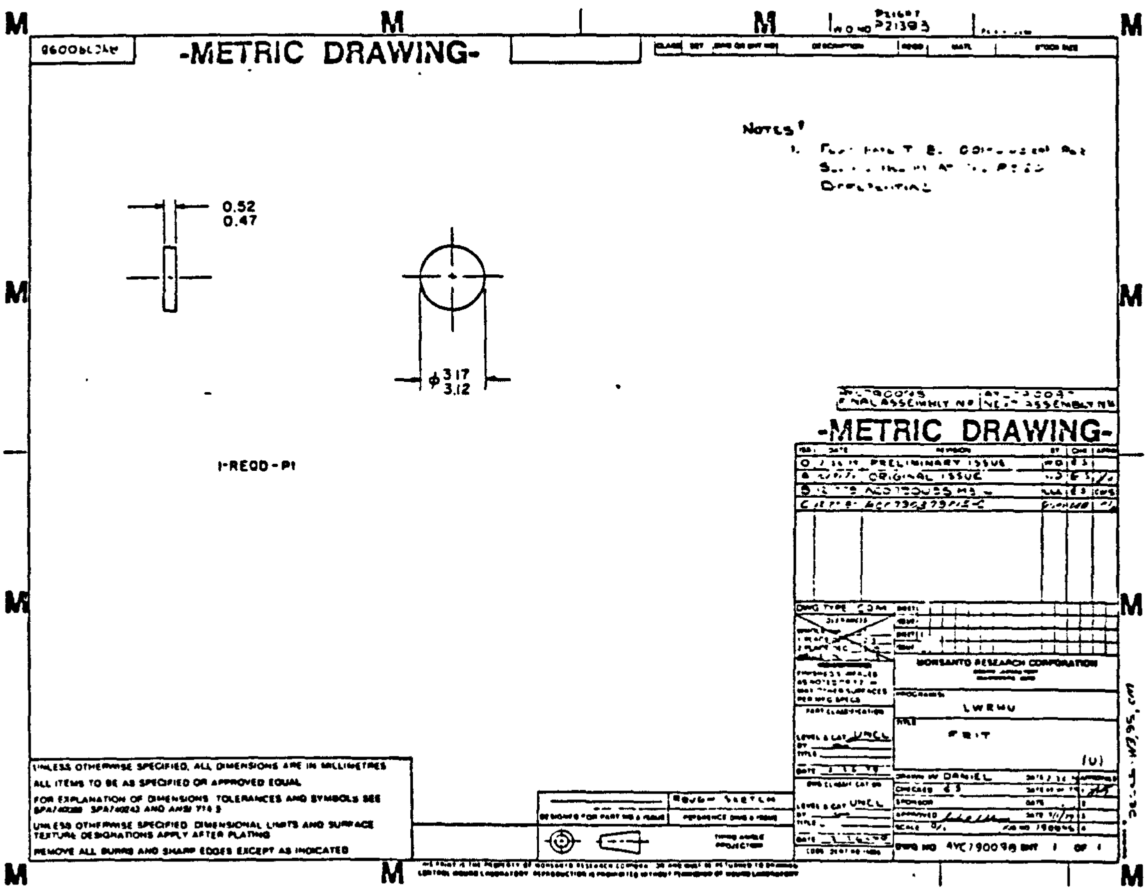


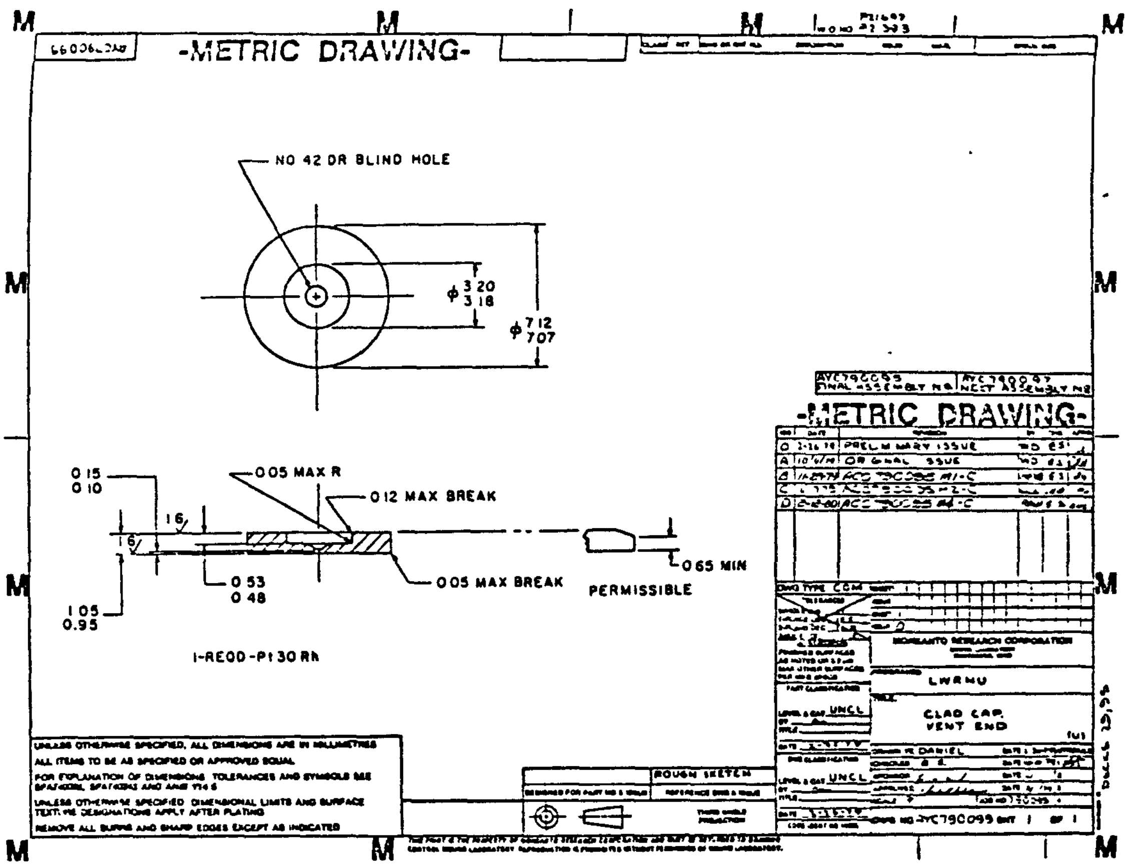
-METRIC DRAWING-
CLAD BODY ASSY
SHIM
CLAD CAP CLOSE END
PELLET FUEL
ϕ 8.65 MAX
G.T.A
12.6 MAX
ACTUAL SIZE
MONSANTO RESEARCH CORPORATION
LIGHT WEIGHT RHU
RADIOISOTOPIC HEATER UNIT (U)
ORIGINAL ISSUE
PRELIMINARY ISSUE
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED
ROUGH SKETCH
DWG NO AYC790095

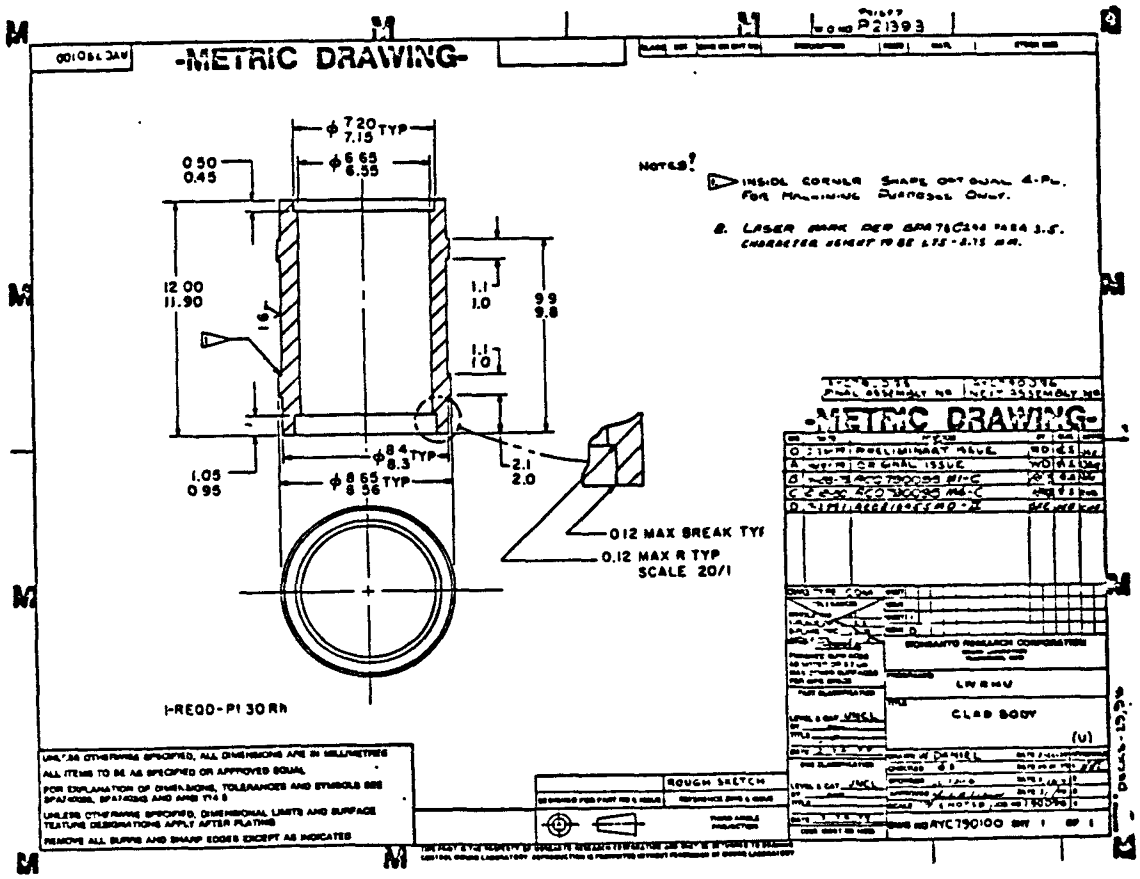


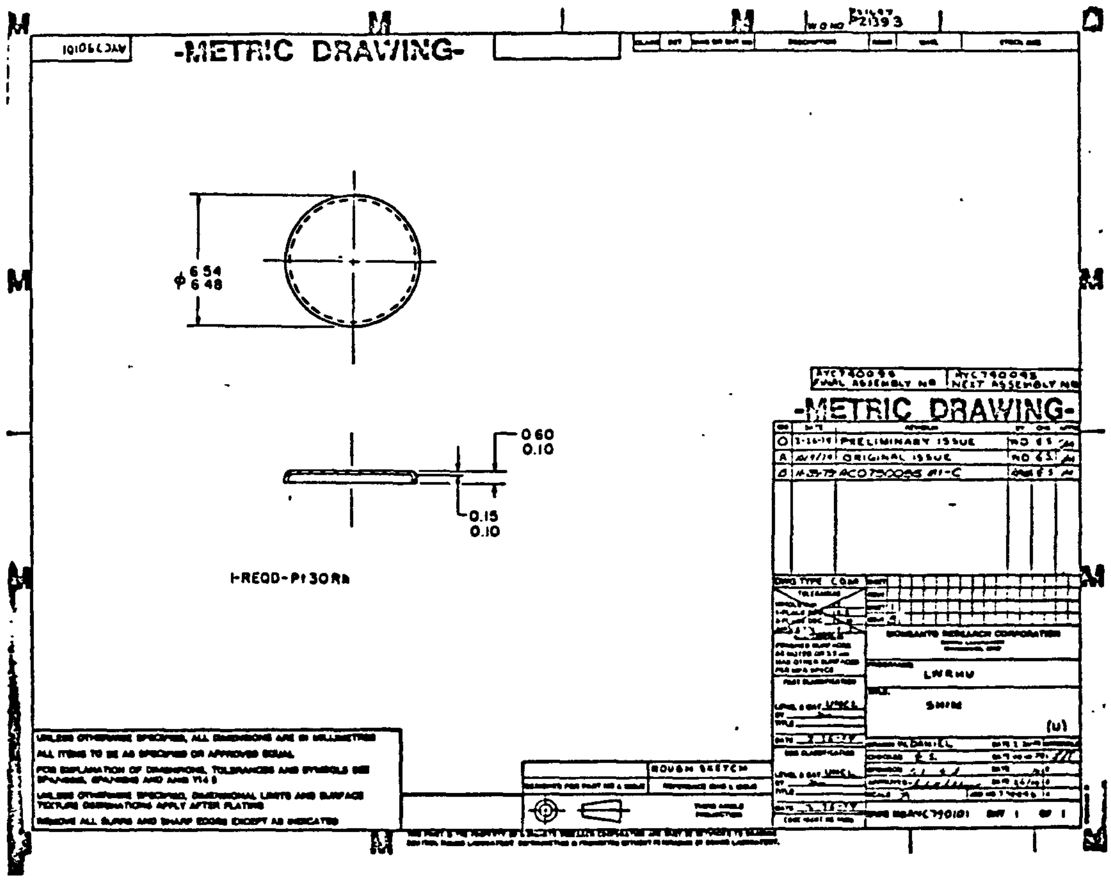
-METRIC DRAWING-
P21393
CLAD BODY
SEE DWG
ϕ 8.65 MAX
ϕ 6.35 MIN
9.75 MIN
E.B.
NOTES:
1. CLEAN HANDLING PROCEDURES REQUIRED.
2. CAUTION: DURING INSPECTION OR HANDLING DO NOT SMEAR OR BURNISH THE FRIT VENT SURFACE.
3. ALL DIMENSIONS APPLY AFTER WELDING.
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
CLAD BODY ASSY (U)
DANIEL
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED
DWG NO AYC790096

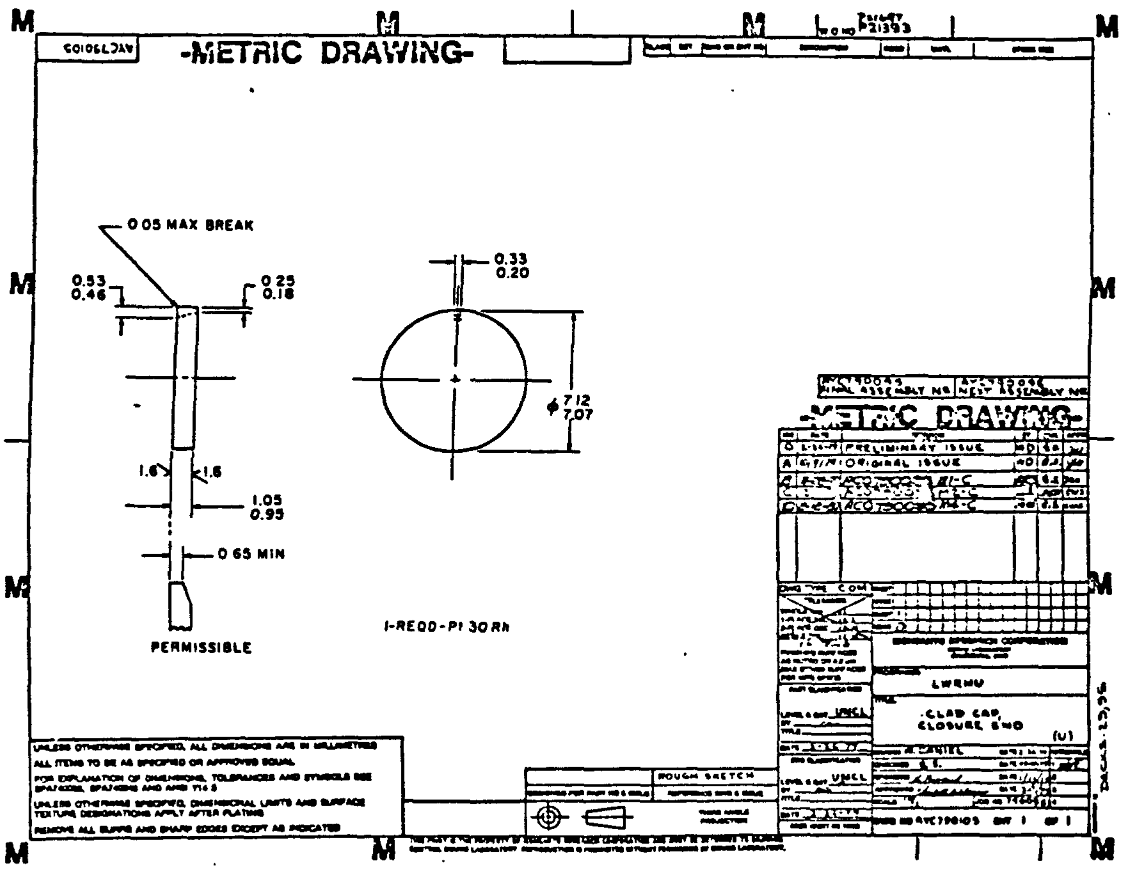

-METRIC DRAWING-
W.O. NO P21393
AYC790097 FRIT
AYC790097 CLAD CAP VENT END
SEE DWG
E.8
1
2
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
CLAD CAP VENT END ASSY (U)
DANIEL
UNCL
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
FOR EXPLANATION OF DIMENSIONS, TOLERANCES AND SYMBOLS SEE SPA740085, SPA740243 AND ANSI Y14.5
UNLESS OTHERWISE SPECIFIED DIMENSIONAL LIMITS AND SURFACE TEXTURE DESIGNATIONS APPLY AFTER PLATING
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED
DWG NO AYC790097 SHT 1 OF 1

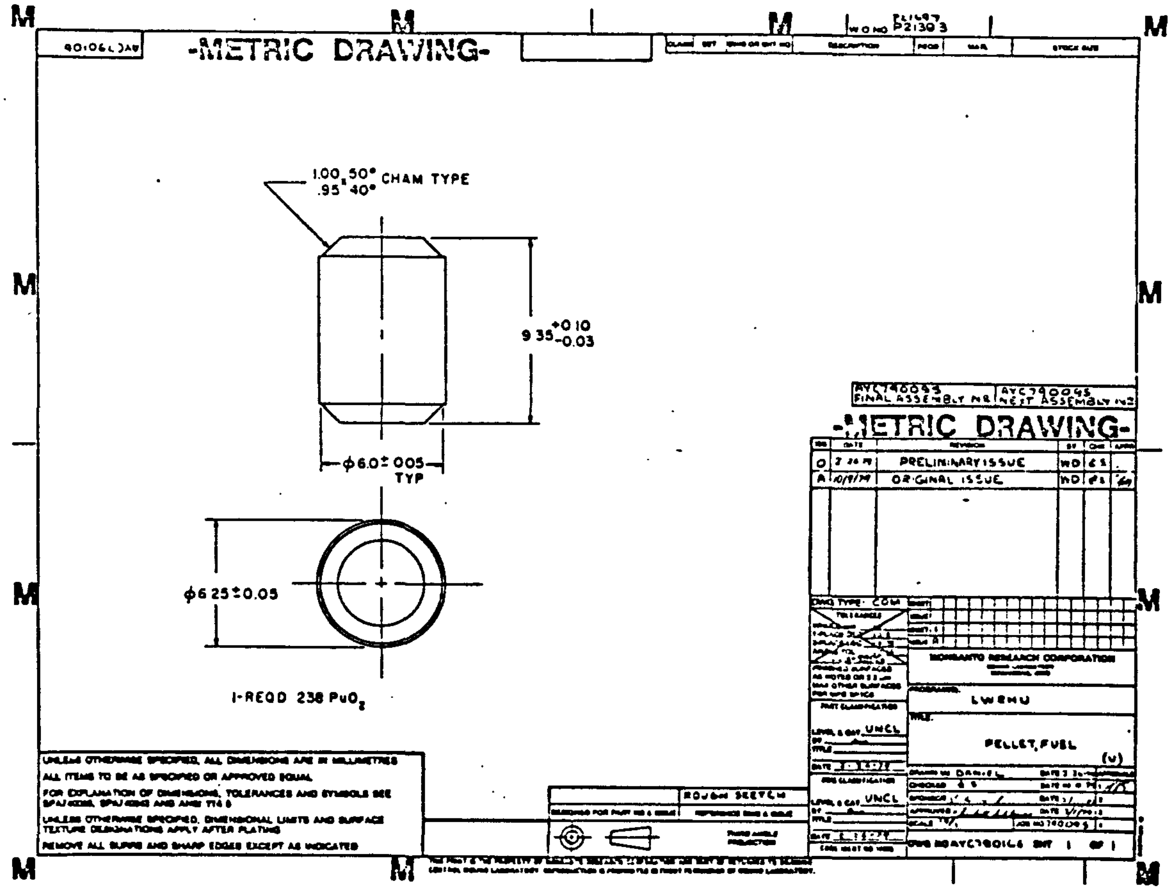

-METRIC DRAWING-
W.O. NO P21393
AYC790098
Notes:
1.
0.52
0.47
ϕ3.17
3.12
1-REQD-P1
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
FRIT (U)
DANIEL
UNCL
ROUGH SKETCH
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
FOR EXPLANATION OF DIMENSIONS, TOLERANCES AND SYMBOLS SEE SPA740085, SPA740243 AND ANSI Y14.5
UNLESS OTHERWISE SPECIFIED DIMENSIONAL LIMITS AND SURFACE TEXTURE DESIGNATIONS APPLY AFTER PLATING
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED
DWG NO AYC790098 SHT 1 OF 1

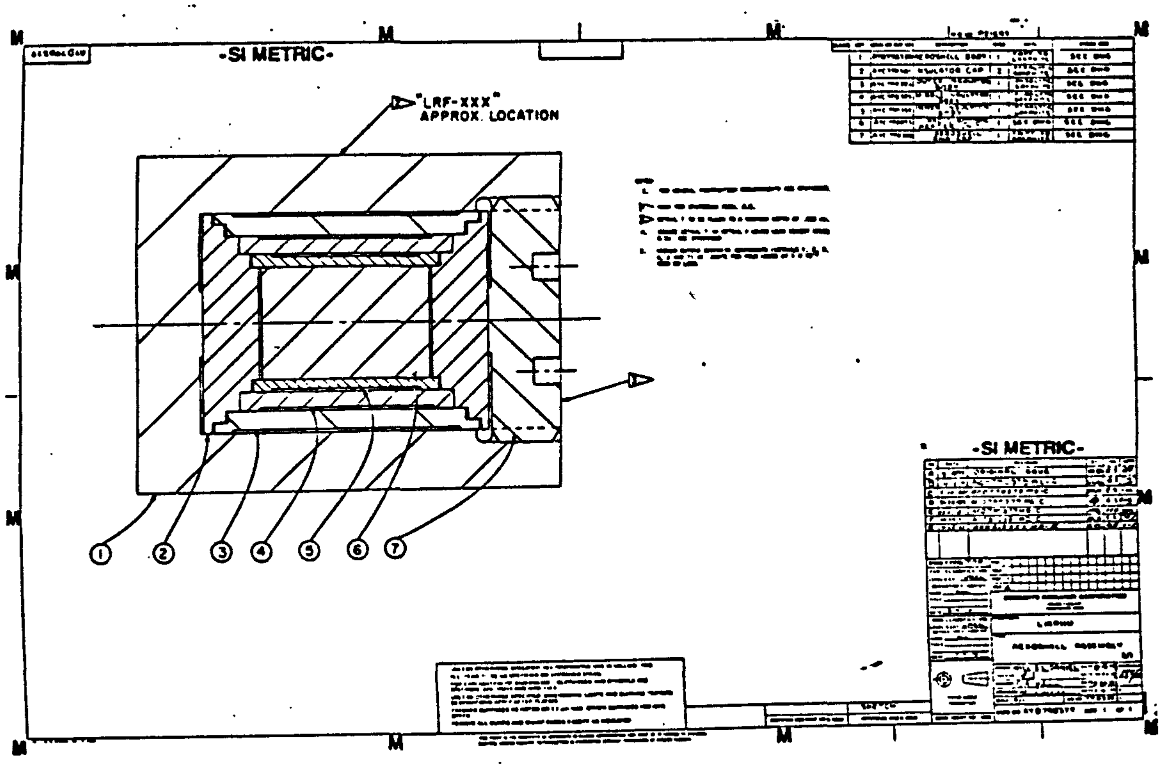

-METRIC DRAWING-
NO 42 DR BLIND HOLE
0.05 MAX R
0.12 MAX BREAK
0.05 MAX BREAK
0.65 MIN
PERMISSIBLE
1-REQD-P1 30 RH
MONSANTO RESEARCH CORPORATION
LWRHU
CLAD CAP, VENT END
RYC790099

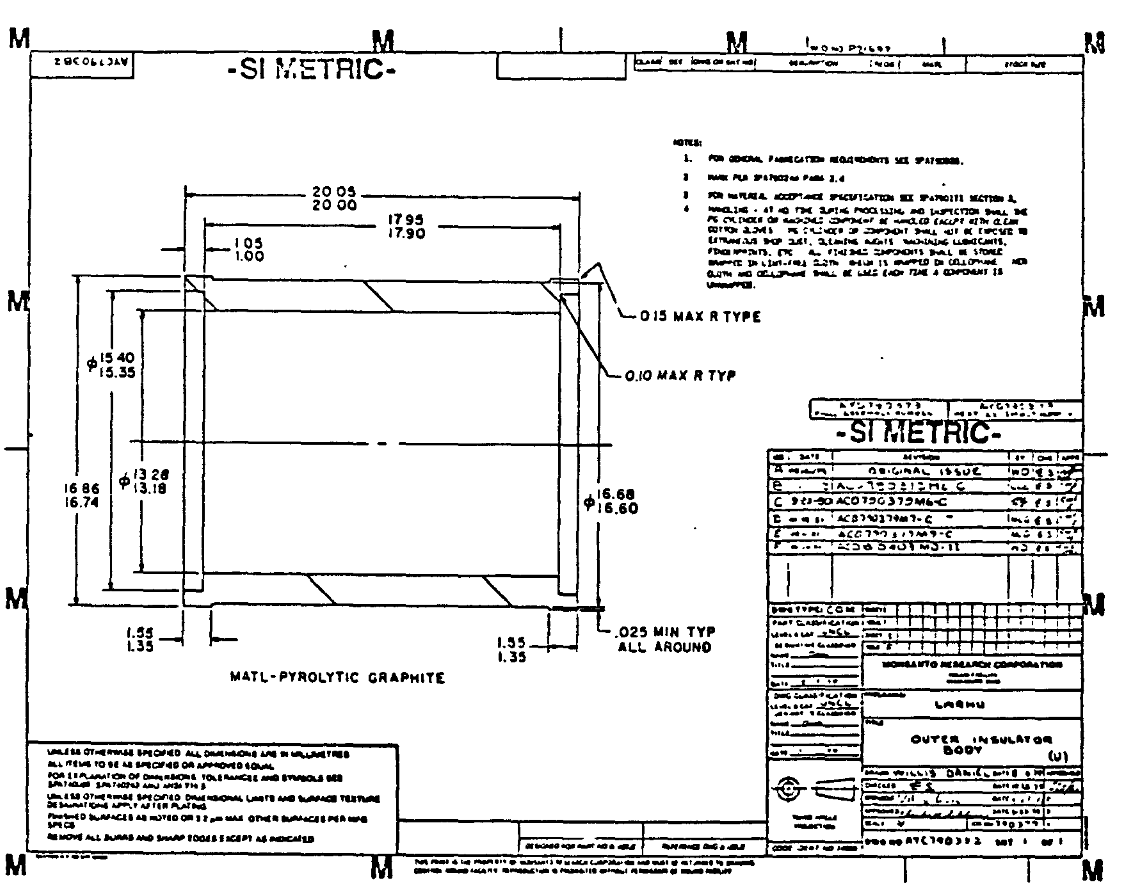

-METRIC DRAWING-
NOTES:
INSIDE CORNER SHAPE OPTIONAL 4-PL. FOR MACHINING PURPOSES ONLY.
2. LASER MARK PER SPA 76C294 PARA 3.5. CHARACTER HEIGHT TO BE 1.75 - 2.15 MM.
0.12 MAX BREAK TYP
0.12 MAX R TYP
SCALE 20/1
1-REQD-P1 30 RH
MONSANTO RESEARCH CORPORATION
LWRHU
CLAD BODY
RYC790100

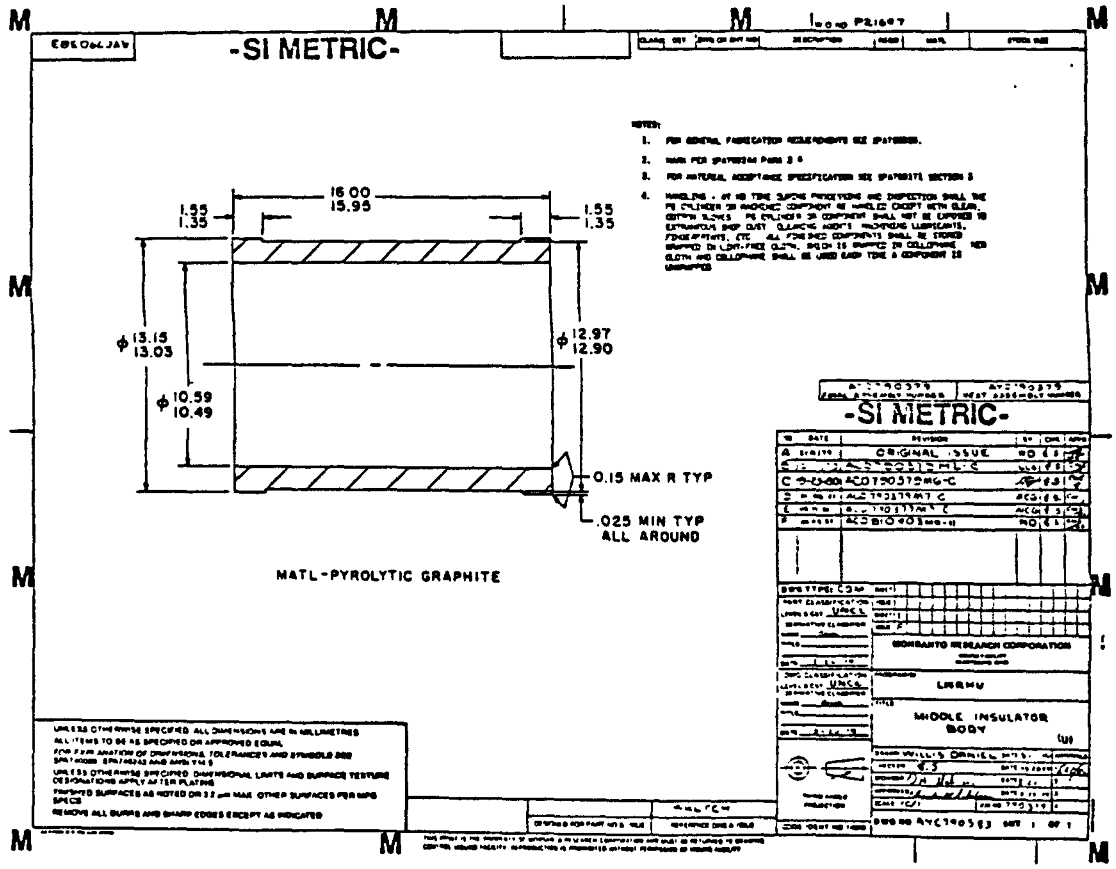

-METRIC DRAWING-
W.O. NO P21393
ϕ 6.54 6.48
0.60
0.10
0.15
0.10
1-REQD-Pt 30Rh
AYC790095
FINAL ASSEMBLY NO
AYC790095
NEXT ASSEMBLY NO
-METRIC DRAWING-
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
SHIM
(U)
UNCL
ROUGH SKETCH
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED

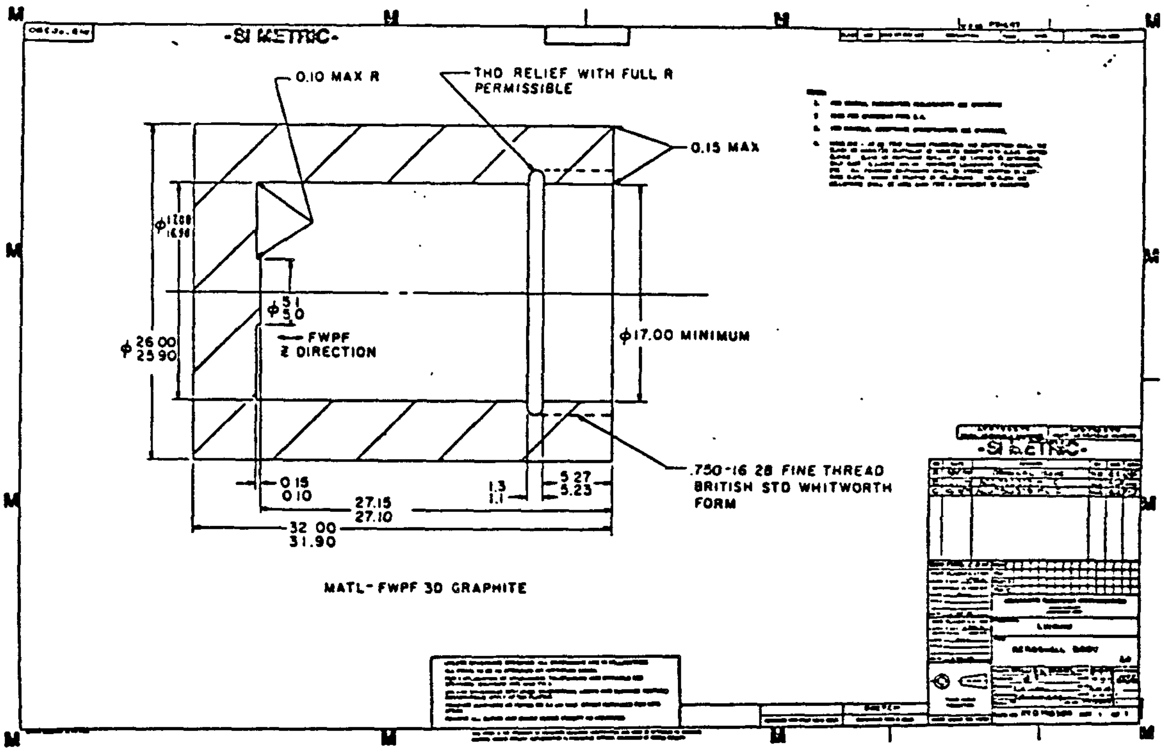

-METRIC DRAWING-
W.O. NO P21393
0.05 MAX BREAK
0.53
0.46
0.25
0.18
0.33
0.20
ϕ 7.12 7.07
1.6
1.6
1.05
0.95
0.65 MIN
PERMISSIBLE
1-REQD-Pt 30Rh
-METRIC DRAWING-
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
CLAD CAP,
CLOSURE END
(U)
UNCL
ROUGH SKETCH
UNLESS OTHERWISE SPECIFIED, ALL DIMENSIONS ARE IN MILLIMETRES
ALL ITEMS TO BE AS SPECIFIED OR APPROVED EQUAL
REMOVE ALL BURRS AND SHARP EDGES EXCEPT AS INDICATED

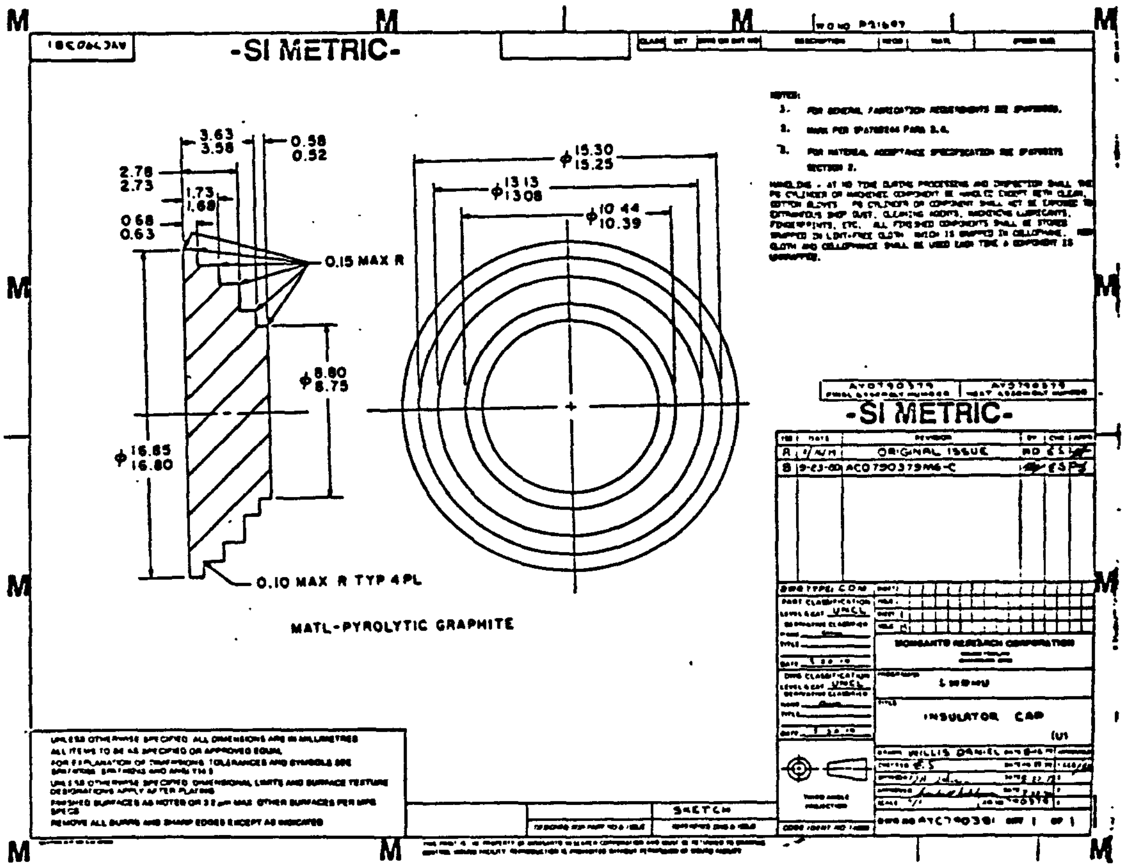

-METRIC DRAWING-
1.00 x 50° / .95 x 40° CHAM TYPE
9.35 +0.10 -0.03
φ6.0± 0.05 TYP
φ6.25±0.05
1-REQD 238 PuO2
-METRIC DRAWING-
PRELIMINARY ISSUE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
PELLET, FUEL
ROUGH SKETCH

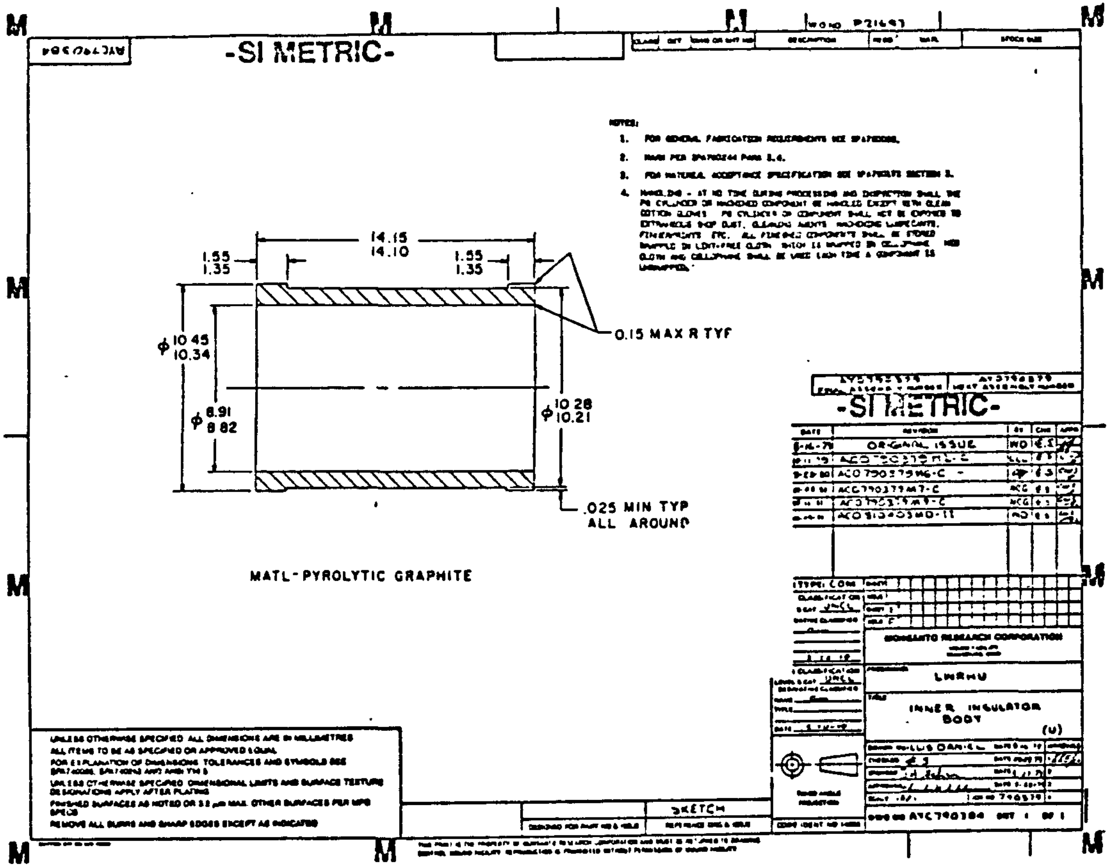

-SI METRIC-
"LRF-XXX" APPROX. LOCATION
1
2
3
4
5
6
7
-SI METRIC-

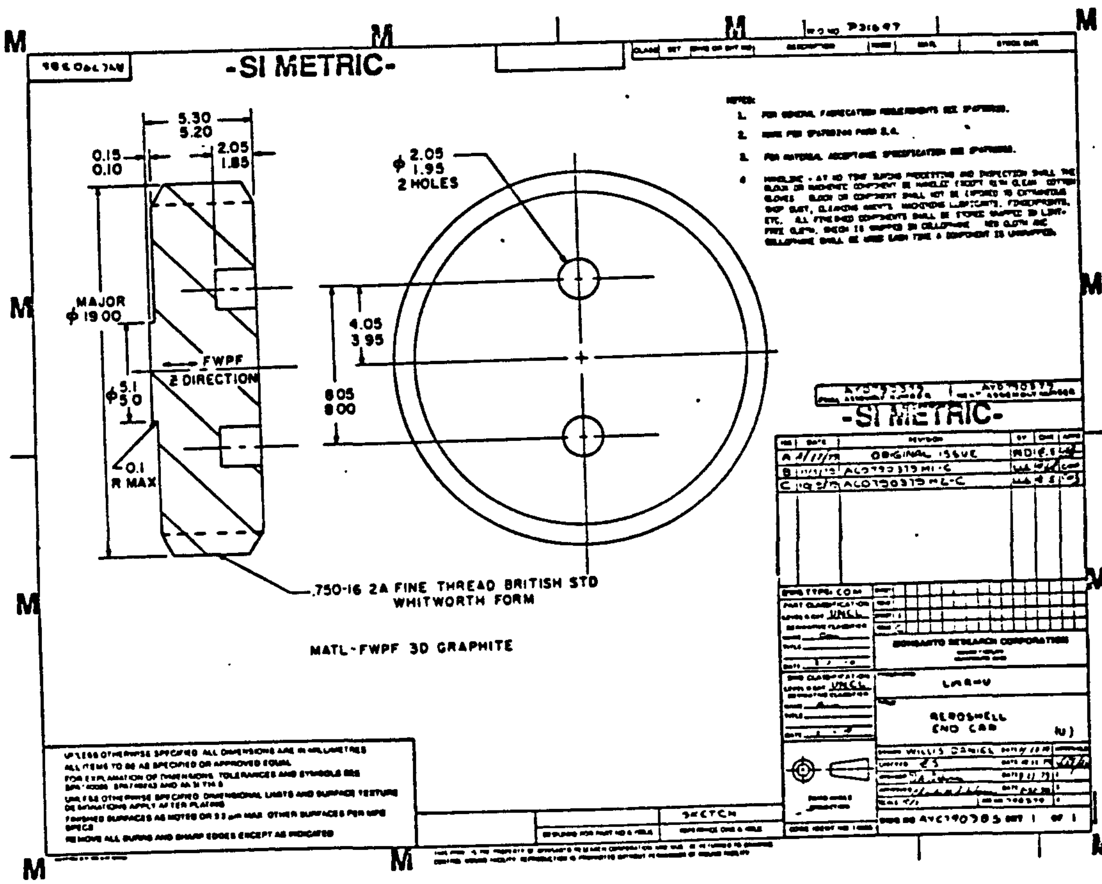

-SI METRIC-
20.05
20.00
17.95
17.90
1.05
1.00
0.15 MAX R TYPE
0.10 MAX R TYP
ϕ 15.40
15.35
ϕ 13.28
13.18
16.86
16.74
ϕ 16.68
16.60
1.55
1.35
.025 MIN TYP
ALL AROUND
MATL-PYROLYTIC GRAPHITE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
OUTER INSULATOR
BODY
(U)


-SI METRIC-
16.00
15.95
1.55
1.35
ϕ 13.15
13.03
ϕ 10.59
10.49
ϕ 12.97
12.90
0.15 MAX R TYP
.025 MIN TYP
ALL AROUND
MATL-PYROLYTIC GRAPHITE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
MIDDLE INSULATOR
BODY
(U)


-SI METRIC-
0.10 MAX R
THD RELIEF WITH FULL R PERMISSIBLE
0.15 MAX
FWPF
Z DIRECTION
ϕ17.00 MINIMUM
ϕ 26.00 25.90
.750-16 28 FINE THREAD BRITISH STD WHITWORTH FORM
0.15 0.10
27.15 27.10
32.00 31.90
1.3 1.1
5.27 5.23
MATL-FWPF 30 GRAPHITE


-SI METRIC-
3.63 3.58
0.58 0.52
2.78 2.73
1.73 1.68
0.68 0.63
0.15 MAX R
ϕ15.30 15.25
ϕ13.13 13.08
ϕ10.44 10.39
ϕ8.80 8.75
ϕ16.85 16.80
0.10 MAX R TYP 4PL
MATL-PYROLYTIC GRAPHITE
ORIGINAL ISSUE
INSULATOR CAP
SKETCH


-SI METRIC-
14.15
14.10
1.55
1.35
1.55
1.35
0.15 MAX R TYP
ϕ 10.45
10.34
ϕ 8.91
8.82
ϕ 10.28
10.21
.025 MIN TYP
ALL AROUND
MATL-PYROLYTIC GRAPHITE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
INNER INSULATOR BODY
SKETCH


-SI METRIC-
5.30
5.20
0.15
0.10
2.05
1.85
ϕ 2.05
1.95
2 HOLES
MAJOR
ϕ 19.00
4.05
3.95
FWPF
Z DIRECTION
ϕ 5.1
5.0
8.05
8.00
0.1
R MAX
.750-16 2A FINE THREAD BRITISH STD
WHITWORTH FORM
MATL-FWPF 3D GRAPHITE
ORIGINAL ISSUE
MONSANTO RESEARCH CORPORATION
LWRHU
AEROSHELL
END CAP
SKETCH

APPENDIX B-2

TABLE OF ACRONYMS AND ABBREVIATIONS

AMD - Accident Model Document

AOA - Abort Once-Around

ATO - Abort-to-Orbit

CBGS-1D - Confined-by-Ground Surface - One Diameter

CBGS-2D - Confined-by-Ground Surface - Two Diameter

CBM - Confined by Missile

CCAFB - Cape Canaveral Air Force Base

CISS - Centaur Integrated Support System

CSS - Centaur Support Structure

C_3 - Launch Energy

DCU - Digital Computer Unit

DOE - Department of Energy

DSN - Deep-Space Network

ET - External Tank

FSAR - Final Safety Analysis Report

FWPF - Fine-Weave Pierced Fabric

g - Gravity

GLL - Galileo Mission

GPHS - General Purpose Heat Source

GSE - Ground Support Equipment

INSRP - Interagency Nuclear Safety Review Panel

IRU - Inertial Reference Unit

JPL - Jet Propulsion Laborator

JSC - Johnson Space Center

KSC- Kennedy Space Center

LANL - Los Alamos National Laboratory

LH_2 - Liquid Hydrogen

LOX - Liquid Oxygen

LWRHU - Light-Weight Radioisotope Heater Unit

MECO - Main Engine Cut-Off

MILA - Merritt Island Launch Area

MRC - Monsanto Research Corporation

NASA - National Aeronautics and Space Administration

NRAD - Nuclear Risk Analysis Document

OMS - Orbital Maneuvering System

ORNL - Oak Ridge National Laboratory

OSNP - Office of Special Nuclear Projects

PCM - Pulse Code Modulation

PG - Pyrolytic Graphite

POSI OMS - Positive Thrust Oms

ppm - Parts per Million

PSAR - Preliminary Safety Analysis Report

RCS - Reaction Control System

RDD - Reference Design Document

RETRO OMS - Reverse Oms

RF - Radio Frequency

RHU - Radioisotope Heater Unit

RMU - Remote Multiplex Unit

rpm - Revolutions per Minute

RPM - Retropulsion Module

RTG - Radioisotope Thermoelectric Generator

RTLS - Return to Launch Site

S/C - Spacecraft

SAEF - Spacecraft Assembly and Encapsulation Facility

SAR - Safety Analysis Report

SMAB - Solid Motor Assembly Building

SRB - Solid Rocket Booster

STS - Space Transportation System (Shuttle)

SSME - Space Shuttle Main Engines

TAL - Trans-Atlantic Landing

TDRSS - Tracking and Data Relay Satellite System

3DCC - Three Dimensional Carbon/Carbon

USAR - Updated Safety Analysis Report

V - Velocity

VPF - Vertical Processing Facility

WX ALT - Cecil - Weather Alternate - Cecil Airport, Jacksonville, Florida

WX ALT - Moron - Weather Alternate, Moron, Spain

WX ALT - KSC - Weather Alternate, Kennedy Space Center

NOTE: Standard abbreviations for chemical symbols, English, and International System of Units (SI) are not included.

APPENDIX B-3

REFERENCES

1. E. W. Johnson, "LWRHU Hardware Summary," presented at the LWRHU Final Design Review, Los Alamos, New Mexico, August 28, 1980.

2. "Safety Analysis Summary Report for the MJS-Radioisotope Heater Unit," MLM-ML-76-48-0003, (August 31, 1976).

3. Updated Safety Analysis Report for the Galileo Mission and the International Solar-Polar Mission, Corporate Author, GESP-7186, (April, 1984).

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5. Verbal Presentation, C. T. Bradshaw, Valley Forge, Pennsylvania, January 23, 1985.

6. Verbal Presentation, R. W. Englehart, Valley Forge, Pennsylvania, January 23, 1985.

7. Teletype Communication, Schaefer, G. J. to Lombardo, J. J.

8. Space Shuttle Data for Planetary Mission Radioisotope Thermoelectric Generator (RTG) Safety Analyis, JSC-08116, (February 15, 1985).

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10. J. C. Uzzell, Jr., Handbook of Thermal Properties for Nuclear Power System Reentry Analysis, ANSP-169, J. C Uzzell, Jr., (1979).

11. Bockris, J.O' M., et al., Physichemical Measurements at High Temperatures, Butterworths Scientific Publications (1959).

12. NERVA Program, Report No. 2275, Vol. 2, Aerojet General (1966).

13. NS0060-01-20, "Acceptance Specification for FWPF Graphite," GE/AEPD, (1982).

14. Johnson, E. W., and Rinehart, G. H., Private Communication, November 12, 1984.

15. Campbell, A. R., and Hawley, E. A., Internal MRC Memorandum, October 16, 1984.

16. Anderson, M. E., "Neutron Flux, Spectrum and Dose Equivalent Measurements for a 4500-W (Th) $^{238}PuO_2$ General Purpose Heat Source," MLM-3248, Monsanto Research Corporation, (May 9, 1985).

17. MLM-2242, Rev.1, Safety Analysis Report for Packaging (SARP) USA/5790/BLF (DOE-AL) and USA/5791/BLF (DOE-AL) (January 25, 1980).

18. PRCBD No. 531583B, K. Hect (Change Request for Reference 7) (January 8, 1985).

NOTICE TO READER

Throughout the volumes comprising the LWRHU SAR, use is made as much as practicable of the SI (International System of Units) nomenclature for units. The four tables below are copied from ASTM E380-84, "Standard for Metric Practice."

TABLE 7 Units in Use with SI Temporarily

Quantity	Unit	Symbol	Definition
energy	kilowatthour	kWh	1 kWh = 3.6 MJ
cross section	barn	b	1 b = 10^{-28} m^2
pressure	bar	bar	1 bar = 10^5 Pa
activity (of a radionuclide)	curie	Ci	1 Ci = 3.7×10^{10} Bq
exposure (X and gamma rays)	roentgen	R	1 R = 2.58×10^{-4} C/kg
absorbed dose	rad	rd	1 rd = 0.01 Gy

TABLE 1 Base SI Units

Quantity	Unit	Symbol
length	metre	m
mass	kilogram	kg
time	second	s
electric current	ampere	A
thermodynamic temperature	kelvin	K
amount of substance	mole	mol
luminous intensity	candela	cd

TABLE 3 Derived SI Units with Special Names

Quantity	Unit	Symbol	Formula
frequency (of a periodic phenomenon)	hertz	Hz	1/s
force	newton	N	kg·m/s^2
pressure, stress	pascal	Pa	N/m^2
energy, work, quantity of heat	joule	J	N·m
power, radiant flux	watt	W	J/s
quantity of electricity, electric charge	coulomb	C	A·s
electric potential, potential difference, electromotive force	volt	V	W/A
electric capacitance	farad	F	C/V
electric resistance	ohm	Ω	V/A
electric conductance	siemens	S	A/V
magnetic flux	weber	Wb	V·s
magnetic flux density	tesla	T	Wb/m^2
inductance	henry	H	Wb/A
Celsius temperature	degree Celsius	°C	K−273.15
luminous flux	lumen	lm	cd·sr
illuminance	lux	lx	lm/m^2
activity (of a radionuclide)	becquerel	Bq	1/s
absorbed dose	gray	Gy	J/kg
dose equivalent	sievert	Sv	J/kg

TABLE 4 Some Common Derived Units of SI

Quantity[1]	Unit	Symbol
absorbed dose rate	gray per second	Gy/s
acceleration	metre per second squared	m/s^2
angular acceleration	radian per second squared	rad/s^2
angular velocity	radian per second	rad/s
area	square metre	m^2
concentration (of amount of substance)	mole per cubic metre	mol/m^3
current density	ampere per square metre	A/m^2
density, mass	kilogram per cubic metre	kg/m^3
electric charge density	coulomb per cubic metre	C/m^3
electric field strength	volt per metre	V/m
electric flux density	coulomb per square metre	C/m^2
energy density	joule per cubic metre	J/m^3
entropy	joule per kelvin	J/K
exposure (X and gamma rays)	coulomb per kilogram	C/kg
heat capacity	joule per kelvin	J/K
heat flux density irradiance	watt per square metre	W/m^2
luminance	candela per square metre	cd/m^2
magnetic field strength	ampere per metre	A/m
molar energy	joule per mole	J/mol
molar entropy	joule per mole kelvin	J/(mol·K)
molar heat capacity	joule per mole kelvin	J/(mol·K)
moment of force	newton metre	N·m
permeability (magnetic)	henry per metre	H/m
permittivity	farad per metre	F/m
power density	watt per square metre	W/m^2
radiance	watt per square metre steradian	$W/(m^2 \cdot sr)$
radiant intensity	watt per steradian	W/sr
specific heat capacity	joule per kilogram kelvin	J/(kg·K)
specific energy	joule per kilogram	J/kg
specific entropy	joule per kilogram kelvin	J/(kg·K)
specific volume	cubic metre per kilogram	m^3/kg
surface tension	newton per metre	N/m
thermal conductivity	watt per metre kelvin	W/(m·K)
velocity	metre per second	m/s
viscosity, dynamic	pascal second	Pa·s
viscosity, kinematic	square metre per second	m^2/s
volume	cubic metre	m^3
wave number	1 per metre	1/m

TABLE 5 SI Prefixes

Multiplication Factor	Prefix	Symbol
$1\,000\,000\,000\,000\,000\,000 = 10^{18}$	exa	E
$1\,000\,000\,000\,000\,000 = 10^{15}$	peta	P
$1\,000\,000\,000\,000 = 10^{12}$	tera	T
$1\,000\,000\,000 = 10^{9}$	giga	G
$1\,000\,000 = 10^{6}$	mega	M
$1\,000 = 10^{3}$	kilo	k
$100 = 10^{2}$	hecto[A]	h
$10 = 10^{1}$	deka[A]	da
$0.1 = 10^{-1}$	deci[A]	d
$0.01 = 10^{-2}$	centi[A]	c
$0.001 = 10^{-3}$	milli	m
$0.000\,001 = 10^{-6}$	micro	μ
$0.000\,000\,001 = 10^{-9}$	nano	n
$0.000\,000\,000\,001 = 10^{-12}$	pico	p
$0.000\,000\,000\,000\,001 = 10^{-15}$	femto	f
$0.000\,000\,000\,000\,000\,001 = 10^{-18}$	atto	a

[A] To be avoided where practical