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LOW ENERGY STAGE STUDY

VOLUME III

CONCEPTUAL DESIGN, INTERFACE ANALYSIS, FLIGHT AND GROUND OPERATIONS

FOR NASA
MARSHALL SPACE FLIGHT CENTER

VOUGHT CORPORATION
an LTV company

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STAGE STUDY

VOLUME III

CONCEPTUAL DESIGN, INTERFACE ANALYSIS, FLIGHT AND GROUND OPERATIONS

FOR NASA
MARSHALL SPACE FLIGHT CENTER

CONTRACT NAS8- 32710
DPD 553 MA- 04

VOUGHT CORPORATION
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F O R E W O R D

This $236,000 Low Energy Stage Study was performed by Vought Corporation under NASA Contract NAS8-32710 for Marshall Space Flight Center from September 1977 through August 1978. The prime objective of the study was to determine the most cost effective approaches for placing automated payloads into low energy Earth orbits. These payloads are injected into circular or elliptical orbits of different inclinations with energy requirements in the range of capability between that of the Space Shuttle standard orbit altitude (296 km) and of the Shuttle with a Spinning Solid Upper Stage - D (SSUS-D). The study results are documented in five volumes:

I. Executive Summary

II. Requirements and Candidate Propulsion Modes

III. Conceptual Design, Interface Analyses, Flight and Ground Operations

IV. Cost Benefit Analysis and Recommendations

V. Program Study Cost Elements and Appendices

The Vought Corporation study manager was Mr. J. M. Bean. Other key Vought participants were H. I. Knight, J. J. Banchetti, B. H. Fuller, B. J. Cathey, and C. D. Stephens.

The study was performed under the technical direction of C. C. Priest, Marshall Space Flight Center. Mr. M. Kitchens was the overall program manager at NASA Headquarters, Office of Space Transportation Systems.

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ABBREVIATIONS AND ACRONYMS

ACS	-	Attitude Control System
A/D	-	Analog to Digital
AGE	-	Aerospace Ground Equipment
AK	-	Apogee Kick
AKM	-	Apogee Kick Motor
AMPTE	-	Active Magneto Spinning Particle Tracer
ANT	-	Antenna
ASE	-	Airborne Support Equipment
BER	-	Bit Error Rate
BITE	-	Built-In Test Equipment
C&DH	-	Communication and Data Handling
CER	-	Cost Estimating Relationship
c.g.	-	Center of Gravity
CITE	-	Cargo Integrated Test Equipment
cm	-	Centimeter
COR	-	Contracting Officers Representative
C/O	-	Checkout
C&W	-	Caution and Warning
CWE	-	Caution and Warning Electronics
db	-	Decibel
DDT&E	-	Design Development Test and Evaluation
DMU	-	Deployment Mechanism Unit
DOD	-	Department of Defense
DVT	-	Development Test
EDS	-	Empirical Data Sets
EED	-	Electro-explosive Device
EIRP	-	Effective Isotrophic Radiated Power
ELV	-	Expendable Launch Vehicle
EMI	-	Electro-Magnetic Interference
ERBS	-	Earth Radiation Budget Satellite
ESH	-	Explosive Safe Area
ETR	-	Eastern Test Range
EVA	-	Extravehicular Activity
FSS	-	Flight Support System for MMS
GCU	-	Guidance and Control Unit
GHe	-	Gaseous Helium
GPC	-	General Purpose Computer
GPS	-	Global Positioning Satellite
GSE	-	Ground Support Equipment
GSFC	-	Goddard Space Flight Center
HMF	-	Hypergolic Maintenance Facility
Hz	-	Hertz (cycles)

ABBREVIATIONS AND ACRONYMS (CONT'D)

ICD	-	Interface Control Document
ICU	-	Ignition Control Unit
I/O	-	Input/Output
IRIG	-	Inter-Range Instrumentation Group
ISU	-	Inertial Stabilization Unit
IUS	-	Inertial Upper Stage
JSC	-	Johnson Space Center
Kbps	-	Kilo bits per second
Kg	-	Kilogram
KSC	-	Kennedy Space Center
km	-	Kilometers
Lbf	-	Pound-Force
Lbs	-	Pounds
L/D	-	Length/Diameter
LES	-	Low Energy Stage
LH	-	Left Hand
MCDS	-	Multifunction CRT Display System
MCEM	-	Mechanical Cost Evaluation Methodology
MDM	-	Multiplexer-Demultiplexer
MFBA	-	Mobile Flat Bed Assembly
MHz	-	Mega hertz
MLI	-	Multilayer Insulation
m	-	Meters
MMH	-	Monomethylhydrazine
MMS	-	Multimission Modular Spacecraft
MMSE	-	Multimission Support Equipment
mps	-	Meters per second
MSDP	-	Mission Station Distribution Panel
MSFC	-	Marshall Space Flight Center
N	-	Newtons
nm	-	Nautical Mile
NSI	-	NASA Standard Initiator
O&CF	-	Operations and Control Facility
OMS	-	Orbiter Maneuvering Subsystem
OPEN	-	Origin of Particles in the Earth Neighborhood
OPAF	-	Ordnance Payload Assembly Facility
OPF	-	Orbiter Processing Facility

ABBREVIATIONS AND ACRONYMS (CONT'D)

PCM	-	Pulse Code Modulation
PCR	-	Payload Changeout Room
PCU	-	Power Control Unit
PDI	-	Payload Data Interleaver
PGHM	-	Payload Ground Handling Mechanism
PK	-	Perigee Kick
PKM	-	Perigee Kick Motor
P/L	-	Payload
PRICE	-	Programmed Review of Information for Costing and Evaluation
PSDP	-	Payload Station Distribution Panel
PSI	-	Pressure Systems, Inc.
psi	-	Pounds per square inch
psig	-	Pounds per square inch, gage
RCS	-	Reaction Control System
RCVR	-	Receiver
RF	-	Radio Frequency
RFI	-	Radio Frequency Interference
RHCP	-	Right Hand Circular Polarization
RMS	-	Remote Manipulator System
R	-	Retrieval
rpm	-	Revolution per minute
SAEF	-	Spacecraft Assembly Encapsulating Facility
SCOOP	-	System Technology Office Confirmation of Optical Phenomenology
S/DIU	-	Signal/Data Interface Unit
SDP	-	Special Defense Program
Sec	-	Second
SIO	-	Serial Input/Output
SOP	-	Standard Operating Procedure
SPST	-	Single Pole Single Throw
SRB	-	Solid Rocket Booster
SRM	-	Solid Rocket Motor
SSUS-A	-	Spinning Solid Upper Stage - Atlas Class
SSUS-D	-	Spinning Solid Upper Stage - Delta Class
STDN	-	Space Tracking and Data Network
STS	-	Space Transportation System
TCD	-	Technical Characteristics Data
Td	-	The Time (months) to design and develop or produce a WBS item
TIG	-	Tungsten Inert Gas
TM	-	Telemetry

ABBREVIATIONS AND ACRONYMS (CONT'D)

TRS	-	Teleoperator Retrieval System
Ts	-	The Lead Time (months) measured from the start of cost accured for the item to the launch milestone date, for the initial item
TSP	-	Twisted Shielded Pair
Tx	-	Transmitter
UARS	-	Upper Atmosphere Research Satellite
UDS	-	Universal Documentation System
VPF	-	Vertical Processing Facility
VPHD	-	Vertical Payload Handling Device
V	-	Revisit
WBS	-	Work Breakdown Structure
WTR	-	Western Test Range
ΔV	-	Delta Velocity

INTRODUCTION

This volume describes the work in Task 3, Conceptual Design of Selected Propulsion Modes, Task 4, Interface Analyses and Task 5, Ground and Flight Operations. In Task 3, low energy conceptual stage designs were developed and performance established. Additionally, adaptation to existing/planned Shuttle upper stages were developed and performance established. Task 3 also includes a discussion of integral propulsion stages. In Task 4, selected propulsion modes and subsystems were used as a basis to develop ASE design concepts. Orbiter installation and integration (both physical and electrical interfaces) were defined. Task 4 also included development of adaptations of LES using existing/planned ASE and Orbiter interfaces. In Task 5, selected low energy stages from Task 3 and ASE from Task 4 were used to define and describe typical ground and flight operations for a LES program.

The report is contained in five volumes and organized as follows:

VOLUME	TASKS	CONTENTS
I	-	Executive Summary
II	1	Requirements Definition
	2	Candidate Propulsion Modes
III	3	Conceptual Design
	4	Interface Analysis
	5	Ground and Flight Operations
IV	6	Cost Benefit Analysis
	7	Recommendations
V	-	Program Study Cost Elements

A listing of references applicable throughout the report is included at the end of each volume.

4.0 TASK 3: CONCEPTUAL DESIGN OF SELECTED PROPULSION MODES

In this task the bipropellant and monopropellant concepts selected in Task 2 were refined and conceptual designs established. Four primary efforts were addressed: evaluation of the requirements of the revised payload mission model, refinement of each of the selected concepts, selection of propulsion subsystem component and selection of other subsystems components. The feasibility of using the propulsion system in an integrated payload/propulsion mode was investigated and the impact of the low energy stage on payload design trends assessed. Outputs of this task used in subsequent tasks were the mass properties, performance and conceptual designs of the selected propulsion modes.

4.1 REVISED PAYLOAD MISSION MODEL

As stated in Volume II, Paragraph 2.0, a revised LES Payload Mission Model was provided for use in Tasks 3 through 6 which incorporated the payloads of the Space Transportation System 487 Model of 1978. This payload mission model, Reference 22, was developed by Battelle Columbus Laboratories and was provided by the NASA in April 1978. The LES Mission Model covers the time period 1980 - 1991.

Like the original payload mission model used in Tasks 1 and 2, the new model included a variety of payloads from small, Scout-class automated spacecraft to large free-flying laboratories and observatories. Destination orbits range from altitudes of a few hundred kilometers to a few thousands kilometers with inclinations from 2.9 to more than 100 degrees. Likewise, geosynchronous transfer orbits were not included in this new model.

The primary data base for the LES Payload Model was the NASA 487 Payload Model. However, some of the data in the 487 model required clarification and/or changes for use in this study. Battelle contacted cognizant individuals within the NASA and other appropriate organizations and reviewed the mission data. Since Scout payloads are not included in the 487 model, a list of planned Scout missions and supporting data was obtained from the NASA Scout Program Manager and these missions were included in the Battelle Low Energy Mission Model. Information relative to unclassified DOD payload missions (included in the 487 model only in general terms) was provided to Battelle by the Aerospace Corporation.

Table 4-I compares the revised LES Payload Mission Model with the original model used for Tasks 1 and 2. Launch schedules by year (1980 - 1991) are shown for ETR, WTR, and Scout launches. The principal differences between the models is a higher WTR launch rate in the revised LES model in the mid-to-late 1980's. This results from a higher projected number of polar and sun-synchronous missions sponsored by non-NASA/non-DOD users in the 487 model.

4.1.1 Model Description

The payload/mission model, Table 4-II, covers a period extending from 1980 through 1991 and includes missions sponsored by NASA, U.S. Government/civil organizations, DOD, and foreign organizations. Planned and potential Scout missions, both NASA and DOD, are listed. The payloads are identified by their mission names, sponsoring organizations, and by their STS launch configurations, payload codes, and classes. The mission line items are numbered sequentially and are referred to by these numbers in Tasks 3 through 6. The data shown for the mission line items are annual launch rate and schedule, mass and size of the payloads, the currently planned launch system and the destination orbit (perigee, apogee, and inclination).

The launch schedules also were primarily taken from the NASA 487 Model. This fact produced some problems in that the schedules in some instances reflect the expected need for retrievals and revisits of spacecraft. These requirements are reflected in Table 4-II launch schedules as "R" (retrieval) or "V" (revisit) where appropriate. Since the Low Energy Stage Study groundrules specify that the low energy stage be baselined as an expendable stage, retrieval missions were excluded from the analysis. Revisits, on the other hand, were assumed to represent additional LES missions equivalent to the original spacecraft launch.

The above assumptions regarding retrieval missions and revisits are reflected in the payload totals in Table 4-II. Thus, in the case of line item #12 ("Gamma Ray Observatory") the three primary launches, two retrievals and three revisits shown produce a total of only 6 LES payloads - the retrievals not being counted.

TABLE 4-I COMPARISON OF LES PAYLOAD MODELS (BY LAUNCH SITE)

	LAUNCH SCHEDULE														
	80	81	82	83	84	85	86	87	88	89	90	91	92	Sub Total	Total
487 LES MODEL															
ETR	1	1	3	3	3	4	3	8	3	7	4	6		46	
WTR	–	–	–	4	6	7	8	9	11	8	10	11		74	131
SCOUT CLASS	3	3	4	1	–	–	–	–	–	–	–	–		11	
ORIG. LES MODEL															
ETR	1	3	3	3	7	4	5	5	6	5	5	7		54	
WTR	1	1	2	5	5	6	8	5	6	4	9	6		58	129
SCOUT CLASS	5	3	4	1	—	1	—	1	—	1	—	1		17	

TABLE 4-II LES PAYLOAD MODEL (FROM NEW 487 MODEL)

			LAUNCH SCHEDULE														SPACECRAFT PARAMETERS			DELIVERY ORBIT				
	MISSION NAME	SPONSOR	80	81	82	83	84	85	86	87	88	89	90	91	92	LES(b) PAYLOAD TOTAL	MASS kg	LENGTH/ DIA. m	STS CONFIG-URATION	APOGEE/ PERIGEE km	INCL. deg.	LAUNCH SITE	PAYLOAD CODE	PAYLOAD CLASS
1.	Extreme UV Explorer	NASA-OSS			1											1	310	0.9/ 4.6	FF	550/ 550	28.5	ETR	AAAA01	P+A EX
2.	High Energy Explorer	NASA-OSS				1		1		1		1				4	2270	4.6/ 4.6	FF	463/ 463	28.5	ETR	AAAA02	P+A EX
3.	Low Energy Explorer	NASA-OSS								1		1				2	1000	1.8/ 1.4	FF	556/ 556	44.9	ETR	AAAA03	P+A EX
4.	Cosmic Background Explorer (COBE)	NASA-OSS					1				1		1			3	816	2.9/ 4.4	FF	900/ 900	99	WTR	AAAF01	P+A EX
5.	IR Astronomy Explorer	NASA-OSS									1		1			2	900	2.5/ 1.5	FF	700-900 Circ.	98-99[a]	WTR	AAAF02	P+A EX
6.	Electrodynamic Explorer A	NASA-OSS							1					1		2	680	1.8/ 1.4	FF	500/ 204	90	WTR	AAAF03	P+A EX
7.	Gravity Probe B (Relativity)	NASA-OSS								1						1	910	3.6/ 2.2	FF	520/ 520	90	WTR	AGAF02	SM/ P+A/FF
8.	Advanced Relativity	NASA-OSS												2		2	910	3.6/ 2.2	FF	520/ 520	90	WTR	AGAF03	SM/ P+A/FF
9.	Plasma Probe B	NASA-OSS						1								1	300	3.0/ 4.6	S-D	29,600/ 370	90	WTR	AGAF04	SM/ P+A/FF
10.	Solar Maximum	NASA-OSS			R		1		R	1	R	1	R	1		4	2047	4.0/ 2.2	MMS	463/ 463	28.5	ETR	AEAE01	SM/ P+A OSS
11.	Upper Atmosphere Research Sat (UARS)	NASA-OSS						1								1	2400	5.0/ 4.0	MMS	400-625 circ	52	ETR	AEAE02	SM/ P+A OSS
12.	Gamma Ray Observatory	NASA-OSS				1		R	1	V	V	R	1	V		6	10000	7.3/ 4.3	FF	400/ 400	28.5	ETR	ACAA01	MED P+A OBS
13.	1.2M X-Ray Observatory	NASA-OSS						1		R1	V	V	R	1		5	10000	12.4/ 4.3	FF	500/ 500	28.5	ETR	ADAA01	LG P+A OBS
14.	Space Telescope	NASA-OSS					1		R	1		V		V		4	9400	12.97/ 4.57	FF	500/ 500	28.5	ETR	ABAA01	LG P+A OBS
15.	Large Solar Obs.	NASA-OSS											1			1	9825	16.2/ 4.6	FF	350/ 350	28.5	ETR	ADAA02	LG P+A OBS
	TOTALS (LES PAYLOADS)(b)				1	2	3	4	2	7	4	5	4	7		39								

(a) Sun-synchronous orbit
(b) Retrievals not included.

TABLE 4-II LES PAYLOAD MODEL (CONT'D)

	MISSION NAME	SPONSOR	80	81	82	83	84	85	86	87	88	89	90	91	92	LES[c] PAYLOAD TOTAL	SPACECRAFT PARAMETERS MASS kg	LENGTH DIA. m	STS CONFIG-URATION	DELIVERY ORBIT APOGEE PERIGEE km	INCL. deg.	LAUNCH SITE	PAYLOAD CODE	PAYLOAD CLASS
	NASA-OA																							
16.	Earth Radiation Budget Sats (ERBSS)	NASA-OSTA			1											1	454	4.3/2.1	FF	704/704	56	ETR	APCA01	LG/APP/FF
17.	Halogen OCC Sat	NASA-OSTA				1										1	60	0.8/0.8	FF	593/593	56	ETR	ARCA01	SM/APP/FF
18.	LANDSAT D	NASA-OSTA				R1		R		1			R1			3	1700[a]	4.3/2.2	MMS	704/704	98.2	WTR	BBCB01	MED/APP/OBS
19.	Soil Moisture	NASA-OSTA					1									1	1400	4.6/1.8	FF	570/570	98	WTR	APCA02	LG/APP/FF
20.	MAGSAT	NASA-OSTA					1									1	270	0.9/0.9	FF	344/344	99	WTR	ARCA02	SM/APP/FF
21.	SEASAT B	NASA-OSTA				1										1	3400	8.0/3.0	MMS	740/740	85	WTR	ACQF03	MED/APP/FF
22.	TIROS O	NASA-OSTA						1								1	1100/1600	7.0/3.6	FF or MMS	830 or 1700[b]	98 or 103	WTR	ACQB01	MED/APP/FF
23.	Environmental Monitor Sat (LOW)	NASA-OSTA						1	1				1			3	1600-2000	5.2/2.3	MMS	590/590	56	ETR	ACQB02	MED/AP FF
24.	Earth Survey	NASA-OSTA								1		1				2	772	3.0/1.5	FF	910/910	100	WTR	ARCF02	SM/APP/FF
25.	Global Resources Mon Info System	NASA-OSTA						1								1	1600-2000	6.1/4.0	MMS	700/700	98.2	WTR	APCB02	LG/APP/FF
26.	GRAVSAT	NASA-OSTA					1				1					2	2000	2.2/4.0	MMS	300/300	90	WTR	APCB01	LG/APP/FF
	TOTAL				1	3	3	3	1	2	1	1	2			17								
	NASA-OAST																							
27.	Space Technology Research Satellite	NASA-OAST								1		R1		R1		3	4500	3.6/2.4	FF	426/426	28.5	ETR	AMBA01	MED/TECH OBS
	TOTAL									1		1		1		3								
	NASA SUMMARY																							
	OSS Total				1	2	3	4	2	7	4	5	4	7		39								
	OA Total				1	3	3	3	1	2	1	1	2			17								
	OAST Total									1		1		1		3								
	NASA TOTAL				2	5	6	7	3	10	5	7	6	8		59								

(a) Includes PM-1 propulsion module to be used for on-orbit attitude control and stationkeeping only.

(b) Circular orbit.

(c) Retrievals not included.

TABLE 4-II LES PAYLOAD MODEL (CONT'D)

	MISSION NAME	SPONSOR	LAUNCH SCHEDULE 80	81	82	83	84	85	86	87	88	89	90	91	LES[c] PAYLOAD TOTAL	SPACECRAFT PARAMETERS: MASS kg	SPACECRAFT PARAMETERS: LENGTH/DIA. m	STS CONFIG-URATION	DELIVERY ORBIT: APOGEE/PERIGEE km	DELIVERY ORBIT: INCL. deg.	LAUNCH SITE	PAYLOAD CODE	PAYLOAD CLASS
28	Tiros Operational	NOAA							1	1	1	1	1	1	6	1100/1600	7.0/3.6	FF or MMS	830 or 1700[d]	98 or 103[a]	WTR	BCQB03	MED/G/OPS
29	Govt Earth Resources - A (low)	U.S. Govt.						1		R	1		R	1	3	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BCRB01	MED/G/OPS
30	Govt Earth Resources - B (low)	U.S. Govt.							1		R	1		R	2	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BCRB02	MED/G/OPS
31	Govt Earth Resources - C	U.S. Govt.								1		R	1		2	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BCRB03	MED/G/OPS
32	OPERATIONAL SEASAT A	U.S. Govt.							1		R1		R1		3	3400[b]	8.0/3.0	MMS	740 circ	85	WTR	BCRB04	MED/G/OPS
33	OPERATIONAL SEASAT B	U.S. Govt.							1		R1		R1		3	3400[b]	8.0/3.0	MMS	740 circ	85	WTR	BCRB05	MED/G/OPS
34	OPERATIONAL SEASAT C	U.S. Govt.								1		R1		R1	3	3400[b]	8.0/3.0	MMS	740 circ	85	WTR	BCRB06	MED/G/OPS
35	INRESAT A	International							1		R		1		2	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BDHB01	MED/C/OPS
36	INRESAT B	International									1		R		1	1700[b]	4.3/2.2	MMS	500-700 circ	97-95[a]	WTR	BDHB02	MED/C/OPS
37	INRESAT C	International										1		R	1	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BDHB03	MED/C/OPS
38	PRIVATE EARTH RESOURCES - A (low)	Commercial							1		R		1		2	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BDHB04	MED/C/OPS
39	PRIVATE EARTH RESOURCES - B (low)	Commercial									1		R		1	1700[b]	4.3/2.2	MMS	500-700 circ	97-98[a]	WTR	BDHB05	MED/C/OPS
	TOTAL							1	6	3	6	4	6	3	29								

(a) Sun-synchronous orbit
(b) Includes MMS PM-1
(c) Retrievals not included.
(d) Circular orbit

TABLE 4-II LES PAYLOAD MODEL (CONCLUDED)

	MISSION NAME	SPONSOR	LAUNCH SCHEDULE 80	81	82	83	84	85	86	87	88	89	90	91	92	LES (c) PAYLOAD TOTAL	SPACECRAFT PARAMETERS MASS kg	LENGTH DIA. m	STS CONFIG-URATION	DELIVERY ORBIT APOGEE PERIGEE km	INCL. deg.	LAUNCH SITE	PAYLOAD CODE	PAYLOAD CLASS
	FOREIGN																							
40.	Canadian Scientific	Canada					1							1		2	400	1.5/ 1.2	FF	550/ 550	90	WTR	AZLA01	SM/F/FF
41.	European Scientific	Europe								1		1		1		3	400	1.5/ 1.2	FF	550/ 550	28.5	ETR	AZLA02	SM/F/FF
42.	Canadian Microwave	Canada						1						1		2	1523	4.0/ 1.5	FF	909/ 909	99	WTR	BAPF01	MED/F/FF
43.	Sea Monitor	Canada									1					1	3110	9.0/ 4.0	FF	800/ 800	98.5	WTR	BAPG01	MED/F/FF
44.	Earth Resources Foreign (low)	Foreign								1		1		1		3	1042	8.2/ 1.5	FF	910/ 910	99	WTR	AZNF03	SM/F/FF
	TOTAL						1	1		2	1	2		4		11								
	DoD (a)																							
45.	USAF Space Test Program	DoD	1	1	1	1	1	1	1	1	1	1	1	1		12	910-1000	1.0/ 4.0	FF	600-1000 Circ.	28.5-100	ETR-WTR[d]	BT	DoD
46.	USAF Meteorological Satellite	DoD				1	1	1	1	1	1	1	1	1		9	1150	6.0/ 3.0	FF	750/750	98.4	WTR	BT	DoD
	TOTAL		1	1	1	2	2	2	2	2	2	2	2	2		21								
	PLANNED SCOUT																							
47.	San Marco D_m	OSS	1													1	50-60	1.5/ 0.8	Scout	27,000/ 420	2.9	SM		Scout
48.	San Marco D_1	OSS	1													1	200	1.5/ 0.8	Scout	800/230	2.9	SM		Scout
49.	Solar Mesosphere Explorer	OSS		1												1	165	1.5/ 0.8	Scout	500/500	97	WTR		Scout
50.	AMPTE A	OSS		1												1	54	1.5/ 0.8	Scout	20 r_e/(b) 200	2.9	SM		Scout
51.	AMPTE B	OSS			1											1	66	1.5/ 0.8	Scout	8 r_e/(b) 200	2.9	SM		Scout
52.	Transit	DoD	1	1	1	1										4	170-200	1.5/ 0.8	Scout	1000/ 1000	90	WTR		DoD
	TOTAL		3	3	2	1										9								
	POTENTIAL SCOUT																							
53.	Canadian Scientific				2											2	145	1.5/ 0.8	Scout	550/ 550	90	WTR	AZLA01	Scout

(a) Battelle 3/78 best estimate of unclassified low energy DoD missions.
(b) 296 if Shuttle launched.
(c) Retrievals not included.
(d) Launches in 1980-1982, 1984, 1986, 1988, 1990 were assumed to be from ETR. The remainder were assumed to be from WTR.

The NASA portion of the LES Model includes a total of 27 STS line items and 5 Scout line items for a total of 64 payloads (59 STS payloads, 5 Scout). Forty-five are small-to-intermediate size automated spacecraft and observatories (less than 3000 kg) and 19 are larger observatories.

The Solar Maximum Mission (item #10) is to be the first use of NASA's MultiMission Modular Spacecraft (MMS), which is currently under development. The first spacecraft in the Solar Maximum program will be launched on a Delta expendable launch vehicle in 1979 (no ELV launches except Scout are shown in Table 4-II). All subsequent missions are to be performed by the STS and are included in Table 4-II, beginning with the retrieval of the Delta launched spacecraft. The MMS is a modular spacecraft bus intended to be adaptable to a variety of applications. In addition to the Solar Maximum mission, use of the MMS is planned or is being considered as an option on a number of NASA missions: Upper Atmosphere Research Satellite (item #11), LANDSAT D (#18), SEASAT B (#21), TIROS (#22), Environmental Monitor Satellite (#23), Global Resources Monitoring Information (#25), and GRAVSAT (#26). In addition, MMS usage is projected in the 487 Model (and the LES Model) to carry over to non-NASA missions evolving as follow-ons to NASA programs such as LANDSAT, TIROS, and SEASAT.

The development of one or more optional propulsion modules is included in MMS planning. A small propulsion module known as the PM-I is already being developed and a larger module, PM-II is under consideration. PM-I will be used on LANDSAT and is intended to provide only in-orbit attitude control, orbit maintenance, and the capability for minor orbit adjustments. PM-II would be intended to perform orbit transfer from the Shuttle to spacecraft operational altitude (and perhaps return). The PM-II was considered a candidate for the launch of MMS payloads in the LES Study.

In some instances, MMS spacecraft definitions shown in Table 4-II include the PM-I. Those cases are identified in the table with a footnote. The inclusion of the PM-I has no impact on the LES Study, since it is not being considered for orbital transfers. The PM-II is not included in any spacecraft definitions in Table 4-II.

The non-NASA portion of the model includes 19 STS line items and one Scout line item (Transit, item #52) for a total of 67 payloads (61 STS payloads, 6 Scout). Twenty-nine of these payloads are in the U.S. Government/Civil User category. These are all envisioned as operational follow-ons to the NASA TIROS, LANDSAT, and SEASAT programs. Consequently, the mission definitions shown (spacecraft parameters, orbit parameters) correspond to those specified for the precursor NASA programs.

The TIROS Operational Spacecraft (item #28) represents the only case in Table 4-II where the LES model launch schedule differs from that shown in the NASA 487 Model. The 487 Model contains three line items (ITOS A, ITOS B, and ITOS C) which have been replaced by the single TIROS Operational line item in Table 4-II. The ITOS A-C series included STS launches beginning in 1983. Current NOAA plans include one series of spacecraft beginning in 1978 and continuing through 1984 or 1985, and a second series to be launched in the 1988-1991 time frame. The first series (now in production) will be launched on Atlas F boosters and were therefore omitted from Table 4-II. The second series will be STS launched and have been included in Table 4-II.

DOD programs in Table 4-II include two representative line items (#45 and #46). In addition, four Scout "Transit" launches are shown in (item #52).

There are a total of 5 STS line items (items #40-44) and one Scout line item (#53) representing foreign payloads in the model. These items produce a total of 13 payloads (11 STS payloads, 2 Scout). Actually, items 40 and 53 could be merged as they represent two phases of the same program, "Canadian Scientific". Table 4-III presents a summary of payload launches by STS user and Scout launches furnished by Battelle for the LES Study.

4.1.2 Model Review

The revised LES Mission Model (Table 4-II) was reviewed to establish a more precise definition of line items mass and orbital characteristics, since some payloads had a range of mass values and/or a range of orbit characteristics. The following changes were made to arrive at a single payload weight and/or orbit characteristics:

TABLE 4-III LES PAYLOAD MODEL SUMMARY - BATTELLE

MISSION CATEGORY	LAUNCH SCHEDULE												TOTAL
	80	81	82	83	84	85	86	87	88	89	90	91	
NASA			2	5	6	7	3	10	5	7	6	8	59
U.S. Govt/Civil						1	6	3	6	4	6	3	29
Foreign					1	1		2	1	2		4	11
DoD	1	1	1	2	2	2	2	2	2	2	2	2	21
Scout Class	3	3	4	1									11
TOTAL	4	4	7	8	9	11	11	17	14	15	14	17	131

TABLE 4-IV LES PAYLOAD MODEL SUMMARY - USED IN STUDY

MISSION CATEGORY	LAUNCH SCHEDULE												TOTAL
	80	81	82	83	84	85	86	87	88	89	90	91	
NASA			2	5	5	7	3	10	4	7	6	8	57
U.S. Govt/Civil						1	6	3	6	4	6	3	29
Foreign					1	1		2	1	2		4	11
DoD			2	3	2	2	2	2	2	2	2	2	21
Scout Class	3	3	4	1									11
TOTAL	3	3	8	9	8	11	11	17	13	15	14	17	129

- Line Items #22 and #28 (TIROS O or TIROS Operational) - 1100 kg and 830 km X 830 km X sun-synchronous inclination was selected to reflect a Free-Flyer STS configuration.
- Line Items #23 and #25 (Environmental Monitor SAT (Low) and Global Resources Monitor Inforamtion System) - An average weight of 1800 kg was selected to reflect a typical mass.
- Line Items #45, #47 and #52 (USAF Space Test Program, San Marco D_M, and Transit) - Since a small weight spread was indicated, the larger was selected for each case.
- Line Items #29, #30, #31 and #35 through #39 - An average orbit of 600 km at sun-synchronous inclination was selected, since all of these payload missions were characterized by the same 500 to 700 km circular orbit.

Two additional modifications were made to the Battelle model to arrive at a representative LES mission model. These modifications were made to reflect a more realistic LES requirement.

- Line Item #45 (USAF Space Test Program) - Since this line item represents a spectrum of DOD space test payloads that are not clearly defined, the launches in 1980 and 1981 were moved to 1982 and 1983 respectively to be consistent with the rest of the LES Payload/Model.
- Line Item #26 (GRAVSAT) - An orbit requirement of 300 km (162 nm) circular was indicated which is essentially the same orbit as the standard Shuttle orbit - 296 Km (160 nm). Therefore, the two launches were not included in the LES model since a low energy stage system would not be required.

The LES Payload Model is summarized in Table 4-IV showing the launch schedule by year, 1980-1991, for the various sponsors including DOD and Scout. The resulting revised LES Payload Model includes a total of 129 individual payload launches.

4.1.3 Performance Requirements

The LES Study Mission Model derived from the Battelle Model provided the payload mass for each of the missions as well as the destination orbits. This information, together with standard Shuttle mission destination orbits, provided the basis for establishing the energy required to effect an orbit transfer for each payload mission.

The standard Shuttle mission destinations, defined in Reference 23 for Tasks 3 through 6 are shown in Table 4-V.

TABLE 4-V STANDARD SHUTTLE ORBITS

LAUNCH SITE	CIRCULAR ORBIT ALTITUDE	INCLINATION
ETR	296 km (160 nm)	28.5° 56.0°
WTR	296 km (160 nm)	90.0° 98.0°

The only difference between these standard Shuttle orbits and those presented in Volume II, paragraph 2.1.3 is the change from 104 degrees WTR inclination to 98 degrees. This change was the result of the large number of payload missions requiring sun-synchronous inclination orbits in the new LES Model. The 98 degree inclination standard Shuttle orbit is approximately in the center of the LES mission model sun-synchronous payload missions and therefore is more representative of the expected Shuttle operation for these missions.

As in Task 1, the payload energy requirements were computed based on the four standard Shuttle inclinations and the initial operational date for the WTR launch site (1983). The required velocity increment (ΔV) was computed for orbit transfer from the Shuttle orbit altitude (296 km) and standard inclination to the payload destination orbit altitude and inclination. Again the inclination change requirements were held to a minimum by assuming that the Shuttle orbit inclination was the one closest to the payload destination orbit inclination. The velocity increment for each payload of the LES Mission Model is presented in Figure 4.1, where the data points are identified according to the Line Item number of Table 4-II.

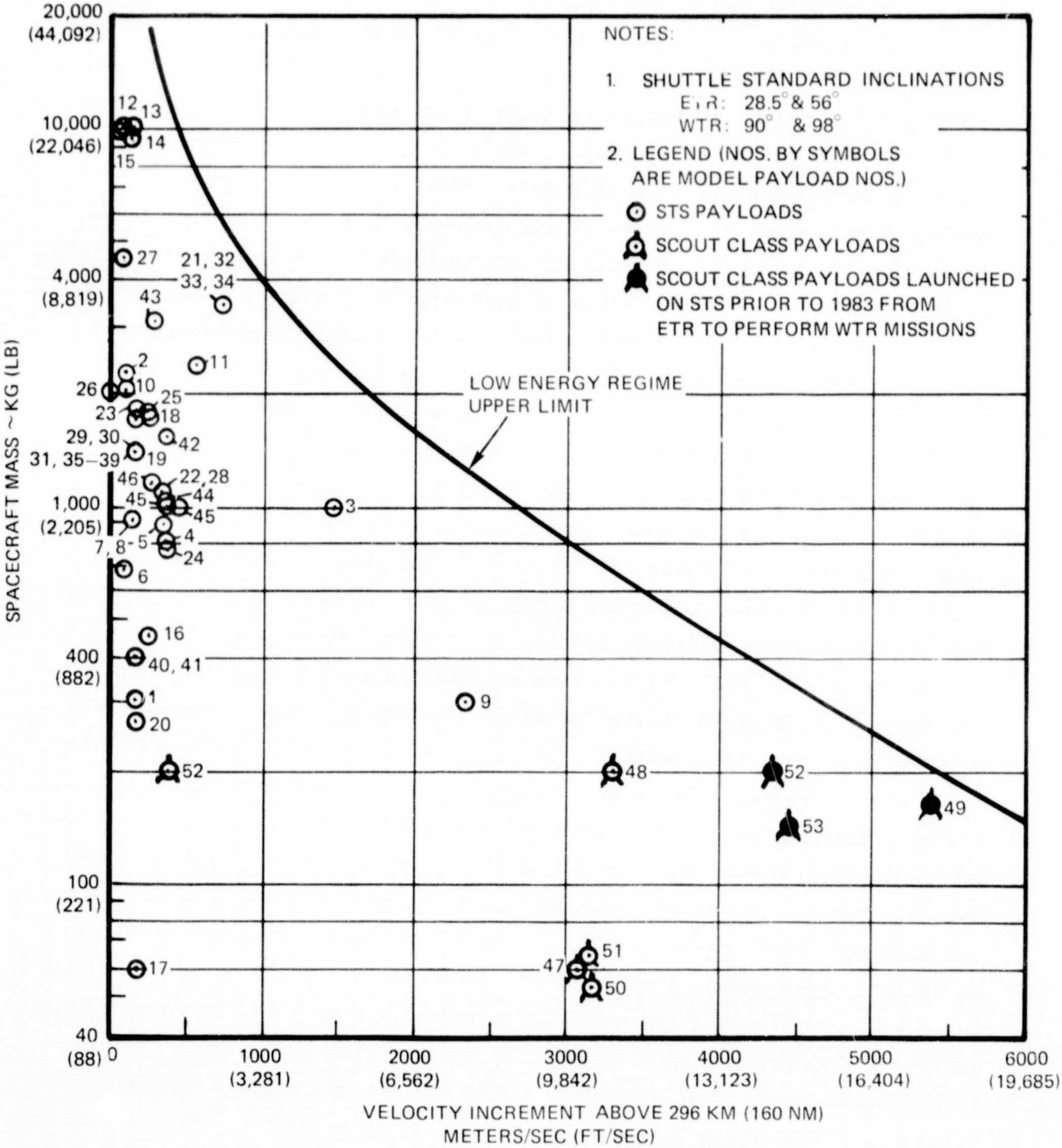


FIGURE 4.1 REVISED LOW ENERGY PAYLOAD MODEL MASS ENERGY REQUIREMENTS

A review of the velocity increments required for the revised LES mission model as compared to the original mission model shows that there is a tighter grouping of payloads in the 1000 m/sec or less range. This grouping tends toward a smaller velocity requirement for the revised model in the range of 500 m/sec or less. Additionally, there are fewer payloads requiring very large velocity increments from 4300 to 5400 m/sec in the revised model, since the number of identified Scout-class payloads is reduced. However, three Scout-launched payloads remain in the revised model to be launched prior to 1983 (Shuttle operational from WTR). Four Scout payloads are in the 3100 to 3300 m/sec range with the energy requirement for one slightly higher due to a weight increase from 64 kg to 200 kg.

A further examination of these velocity increment requirements presented in Figure 4.1 led to a more realistic definition of the Low Energy Regime for purposes of LES concept refinements in Task 3. A generalized velocity increment requirement was superimposed on the LES mission model requirements of Figure 4.1 reflecting the velocity increment required to provide an inclination change of 28.5 degrees as well as to provide a typical orbit altitude of 1111 km (600 nm) representing the lower fringes of the Van Allen belt. This velocity increment, approximately 3650 m/sec, is shown in Figure 4.2 along with several other generalized velocity increment requirements: to change altitude only (up to 1000 km), to achieve a 1000 km orbit with 12 degree inclination change, and to earth escape from the Shuttle orbit with no inclination change. The region bounded by the solid line was taken as the new "Low Energy Regime" definition for Task 3. The energy requirements for the three payloads which appear outside the boundary (Payloads number 49, 52 and 53) reflect the energy required to deliver these paylodas with the STS from ETR into polar or sun-synchronous orbits. After 1982 (when STS is operational from WTR) the velocity requirement for payloads of this category is less than 1000 m/sec. The payload missions are coded, in Figure 4.2 to reflect ETR and WTR launches.

4.1.4 Payload Characteristics and Requirements

A review of the LES payload model was conducted to determine if the payload size, mass and destination accuracy requirements should be changed from those established in Task 1. The review revealed a trend to

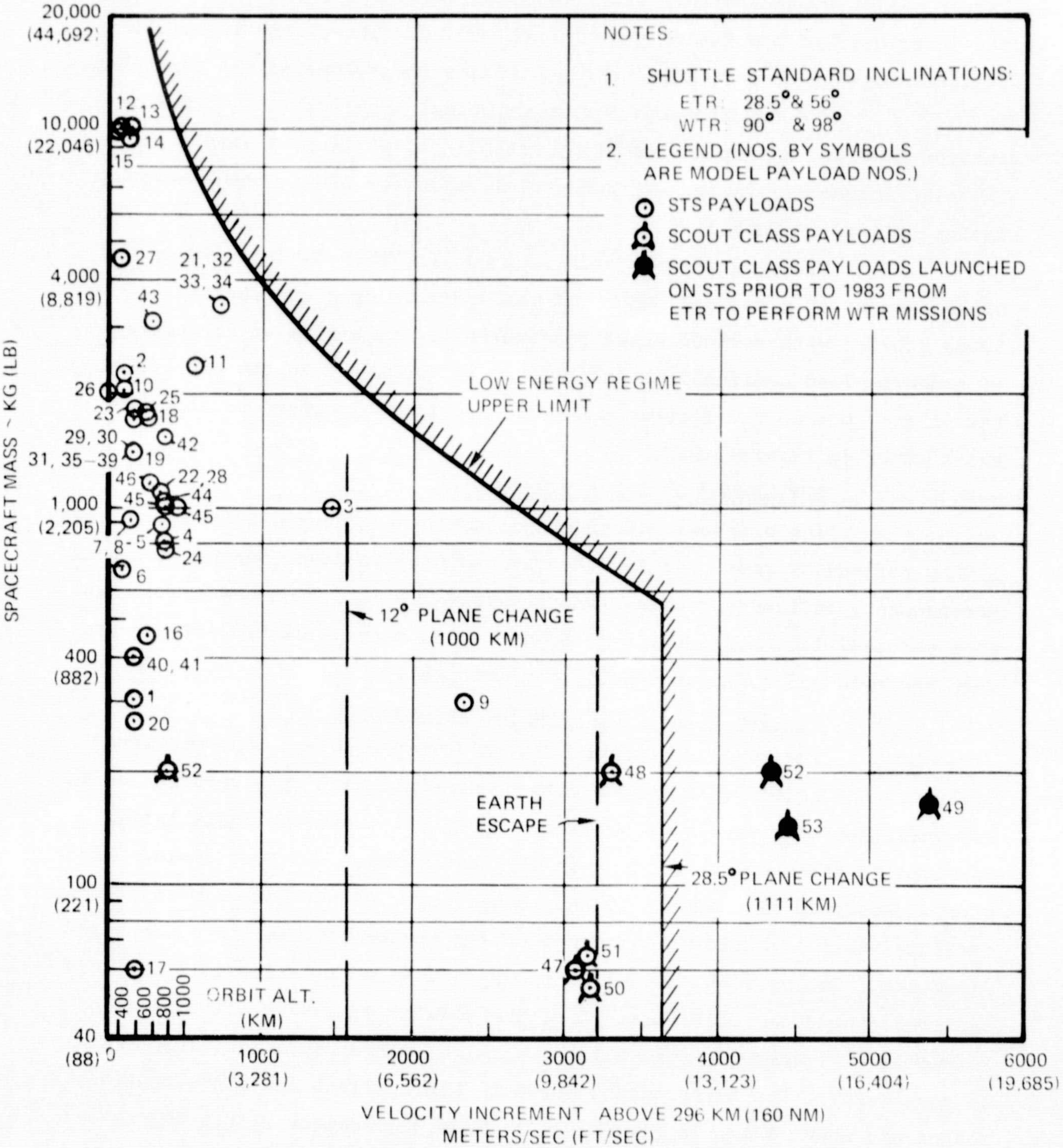


FIGURE 4.2 REFINEMENT OF LOW ENERGY REGIME DEFINITION

larger diameter and shorter length payloads for those payloads planned for the later half of the twelve year launch period. This trend is toward a payload shape which more efficiently utilizes the volume of the cargo bay.

The requirements for longitudinal accelerations, orbit insertion accuracy and spin capability established earlier in Task 1 and as shown in Volume II, Table 2-VI were still valid for use in the conceptual design efforts of Task 3.

Grouping the payloads as shown in Table 4-VI permitted the establishment of a reduced number of payload envelopes and masses for the trade studies with various stage configurations to arrive at arrangements of stage/payload combinations which could be installed in the Orbiter cargo bay in the most cost effective manner. A composite payload envelope is shown later in Figure 5.3.

4.2 REFINEMENT OF SUBSYSTEM CONCEPTS

The bipropellant and monopropellant concepts selected in Task 2 were refined by re-sizing for the revised mission model, propellant tank arrangement, refined structural and subsystem weights, and the consideration of a vertical launch capability. Guidance and control system refinements were reviewed and a guidance subsystem selected.

4.2.1 Shuttle Cargo Bay Packaging Arrangement

The modular bipropellant and monopropellant concepts were evaluated and selected in Task 2 without consideration of a vertical cargo bay installation capability for either approach. Vertical installation versions of these selected concepts were evaluated. Net cost savings of approximately 20 $M compared to the launch costs of the Task 2 mission model indicated that a vertical launch capability be included in the concept refinements. Two vertical installation approaches were selected. The first centered around the rearrangement of the modular structure and tankage of Volume II - Figures 3.14 and 3.15 for minimum cargo bay length when installed vertically. The arrangements for both bipropellant and monopropellant stages are shown later in Figures 4.26 and 4.32, respectively. The second approach was centered around the modular bipropellant stage and its capability to generate a very high velocity change for equatorial orbit payloads weighing up to 150 kg (331 lbs.). The higher weight and lower specific impulse

TABLE 4-VI LES PAYLOADS ORDERED BY PAYLOAD LENGTH

MISSION NUMBER	LENGTH M (ft)	DIA M (ft)	MASS Kg (lb)	NO. OF LAUNCHES 131	VERT. CAND. 26	FIRST YEAR
17	0.8 (2.6)	0.8 (2.6)	60 (132)	1	1	83
20	0.9 (3.0)	0.9 (3.0)	270 (595)	1	1	84
1	0.9 (3.0)	4.0 (13.1)	310 (683)	1		82
45	1.0 (3.3)	4.0 (13.1)	1000 (2205)	12		80
* 53, 47, 48, 49, 50, 51, 52 {*145, 60, 200, 165, 54, 66, 200} {320} {132} {441} {364} {119} {146} {441}	1.5 (5.0)	0.8 (2.6)	*	11	11	80
40,41	1.5 (5.0)	1.2 (3.9)	400 (882)	5	5	82
3, 6	1.8 (6.0)	1.4 (4.6)	1000 (2205)	4	4	86
26	2.2 (7.2)	4.0 (13.1)	2000 (4410)	2		84
5	2.5 (8.2)	1.5 (5.0)	900 (1984)	2	2	88
24	3.0 (9.8)	1.5 (5.0)	772 (1702)	2	2	87
4	2.9 (9.5)	4.4 (14.4)	816 (1800)	3		84
9	3.0 (9.8)	4.6 (15.0)	300 (662)	1		85
7, 8	3.6 (11.8)	2.2 (7.2)	910 (2007)	3		87
27	3.6 (11.8)	2.4 (7.9)	4500 (9921)	3		87
42	4.0 (13.1)	1.5 (5.0)	1523 (3358)	2		85
16	4.3 (14.1)	2.1 (6.9)	454 (1001)	1		82
10	4.0 (13.1)	2.2 (7.2)	2047 (4513)	4		84
18,29,30,31,35,36,37,38,39	4.3 (14.1)	2.2 (7.2)	1700 (3748)	17		83
19	4.6 (15.1)	1.8 (6.0)	1400 (3086)	1		84
2	4.6 (15.1)	4.6 (15.0)	2270 (5004)	4		83
11	5.0 (16.4)	4.0 (13.1)	2400 (5291)	1		85
23	5.2 (17.1)	2.3 (7.5)	1800 (3968)	3		85
46	6.0 (19.7)	3.0 (9.8)	1150 (2535)	9		83
25	6.1 (20.0)	4.0 (13.1)	1800 (3968)	1		85
22, 28	7.0 (23.0)	3.6 (11.8)	1100 (2425)	7		85
12	7.3 (24.0)	4.3 (14.1)	10000 (22046)	6		83
21, 32, 33, 34	8.0 (26.2)	3.0 (9.8)	3400 (7496)	10		83
44	8.2 (26.9)	1.5 (5.0)	1042 (2297)	3		87
43	9.0 (29.5)	4.0 (13.1)	3110 (6856)	1		88
13	12.4 (40.7)	4.3 (14.1)	10000 (22046)	5		85
14	12.9 (42.3)	4.5 (15.0)	9400 (20723)	4		84
15	16.2 (53.2)	4.6 (15.0)	9825 (21660)	1		90

of the monopropellant concept precluded its consideration in this application. Propulsion and other subsystem components for this modular configuration were the same as the horizontal bipropellant stage except for a rearrangement for vertical installation. A twelve tank version of this approach is shown later in Figure 4.27. A four tank version of this arrangement, through removal of the upper and lower rows of tank pairs, uses the same primary structure.

4.2.2 Structural Arrangement and Weight Refinement

Figure 4.3 shows the modular LES structural arrangements evaluated in Task 3. Each stage configuration was analyzed for a critical payload/stage combination selected from Table 4-II. The structure was designed to have strength and stiffness to withstand the environment during prelaunch ground operations, Orbiter boost and shutdown, and deployment from the cargo bay followed by free flight boost. Normal and emergency landing conditions were considered for abort or options where LES was not deployed from the Orbiter.

4.2.2.1 Structural Criteria - Structural design and environmental criteria used to size the structure are found in Johnson Space Center documents, References 24 and 37. The critical design condition is for emergency landing, a load condition where the ultimate factor of safety is 1.0. Table 4-VII shows the ultimate linear and angular accelerations for this condition as compared to the boost (max), a critical normal operating condition where the ultimate factor of safety is 1.4. The Gx acceleration for boost is slightly higher (4.62 vs 4.5) yet the combined loads are greater for emergency landing. Shipping and ground handling conditions were not critical for the stage design.

TABLE 4-VII LES DESIGN LOADS - ULTIMATE

	LINEAR - g			ANGULAR-RAD/SEC^2		
CONDITION	X	Y	Z	X-X	Y-Y	Z-Z
Boost (max)	-4.62	-0.28	-1.05	-0.28	-0.35	-0.35
Emergency Landing	+4.5	+1.5	+4.50	+4.5	+0.738	+2.215

COORDINATE SYSTEM: UP, RIGHT, AFT ACCELERATIONS ARE POSITIVE (RIGHT HAND RULE APPLIES)

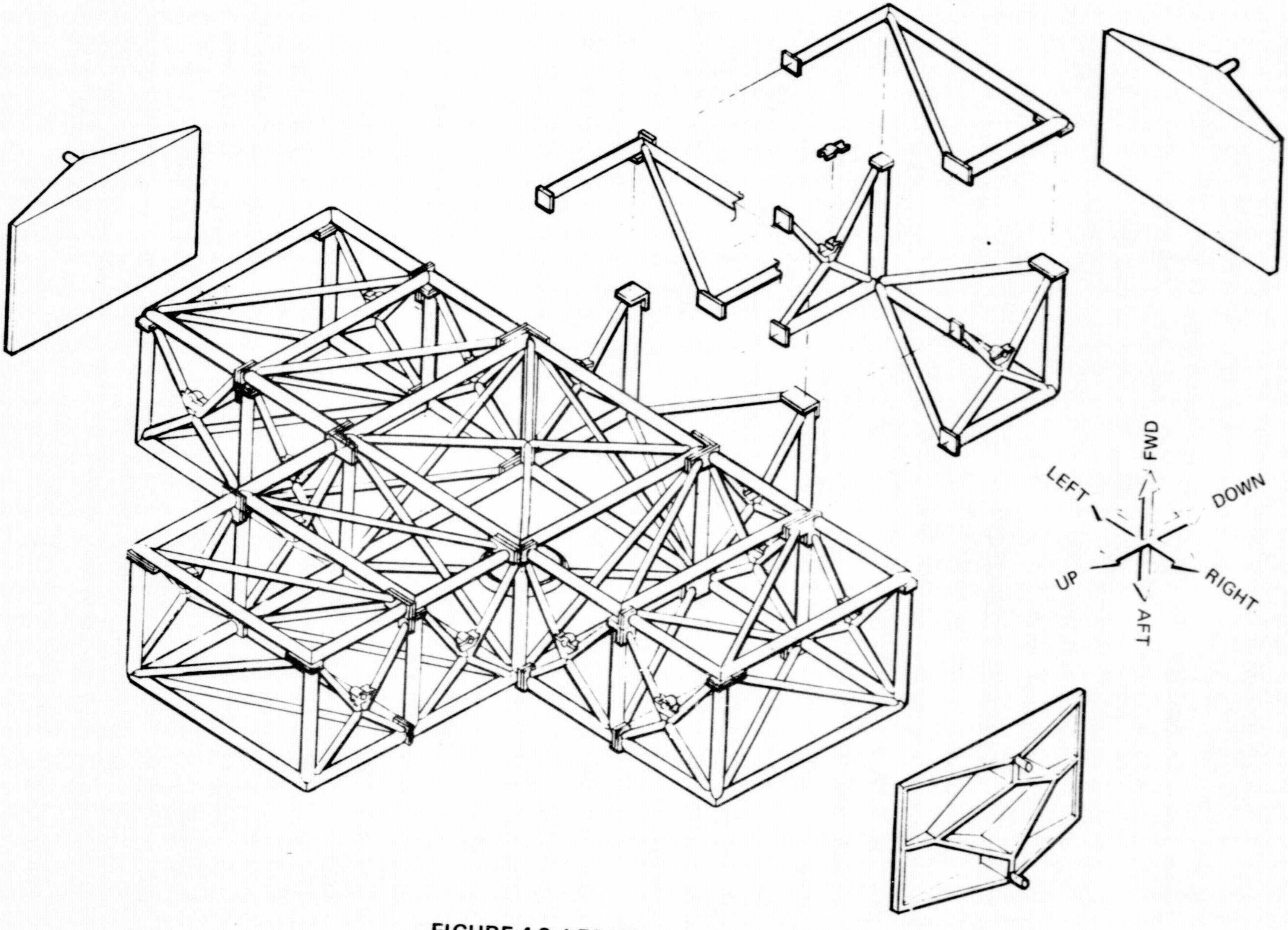


FIGURE 4.3 LES MODULAR STRUCTURE

Qualification structural static and dynamic tests were assumed to require two test articles. The static test article would be tested to ultimate loads to qualify the structure for handling, flight, and landing conditions. This article would be tested to failure. The structural article would have actual or simulated systems for performing dynamic, vibro-acoustic, and fatigue testing. This article would be refurbished and used later as an operational unit.

4.2.2.2 Structural Arrangement and Weight Refinement - Parametric weight variations, based on empirically derived equations, were used in Task 2 for the comparison of the many propulsion approaches under consideration. In Task 3 a structural loads analysis and structural member sizing was performed to determine the detail structural weights for the selected concepts. Additionally, structural arrangement versus weight trade studies concluded that an all truss structure as shown in Figure 4.3 was the optium structural configuration. The parametric stage structural weight comparisons of four structural concepts are shown in Figure 4.4. The flat cross (concept 1) was selected for the 8-tank horizontally mounted configurations. Modules of these stage structures are removed to form the 2-and 4-tank stage vertical and horizontal configurations. This modularity is shown in Figure 4.3. The weight data of Figure 4.4 includes a 10% contingency for additional stiffness requirements for dynamic loads which were not considered in the analysis.

The all truss was selected for its structural efficiency to support a wide variety of stage components requiring a large volume structural system. In addition, the truss structure provides access to fuel tanks and system components without weight penalties associated with access panels for monocoque and semi-monocoque structure.

Aluminum material (2219) was selected for the swaged tube truss members because of its weldability, and high specific stiffness and strength. Joints and mechanical parts use the same material so that the joints can be welded or mechanically joined as dictated by fabrication assembly requirements. Approximately 70% of the joints are considered to be welded to take advantage of weight savings. Weight efficiencies for six welded and mechanical joint configurations are presented in Figure 4.5. From

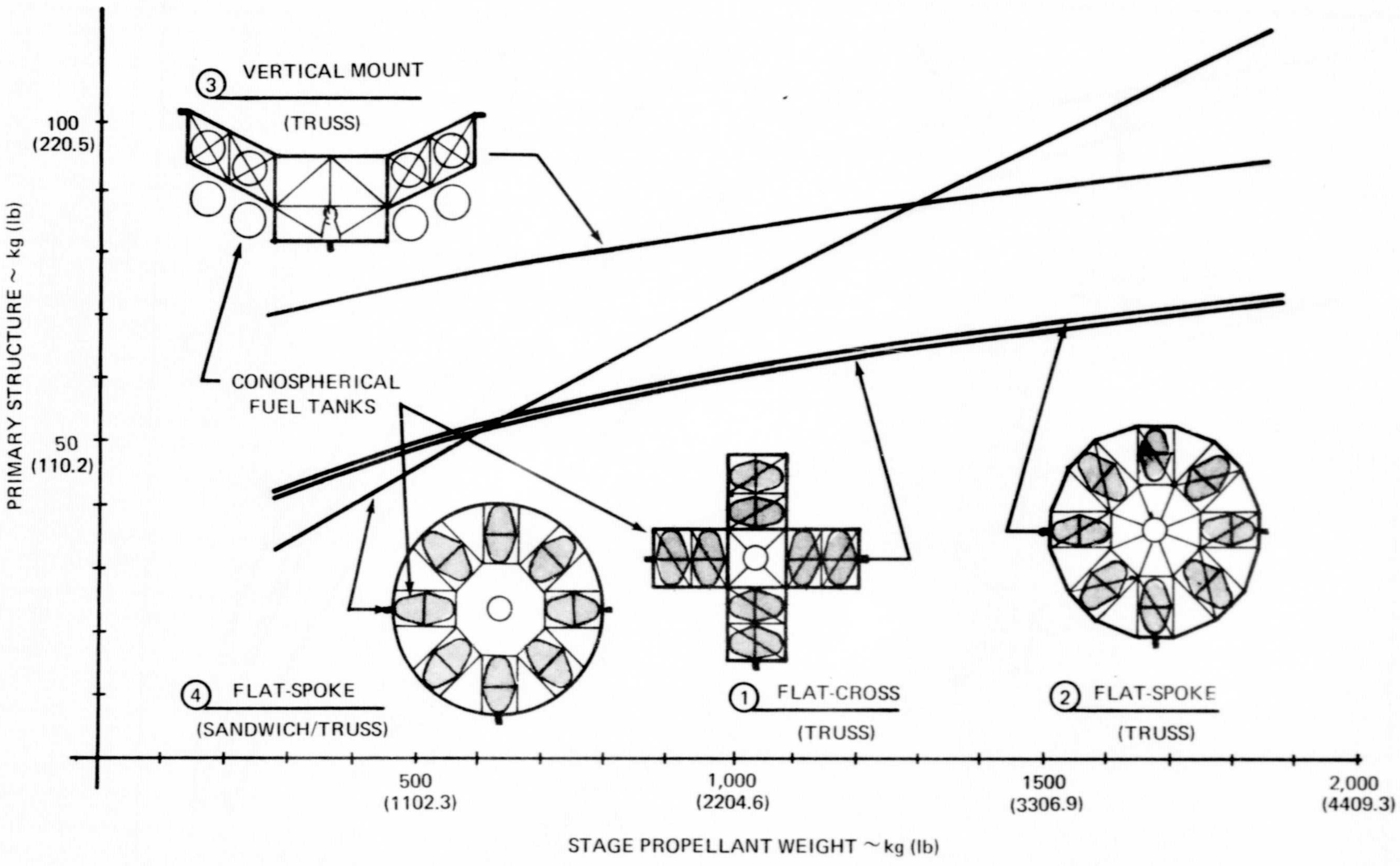


FIGURE 4.4 STAGE STRUCTURAL WEIGHT COMPARISON

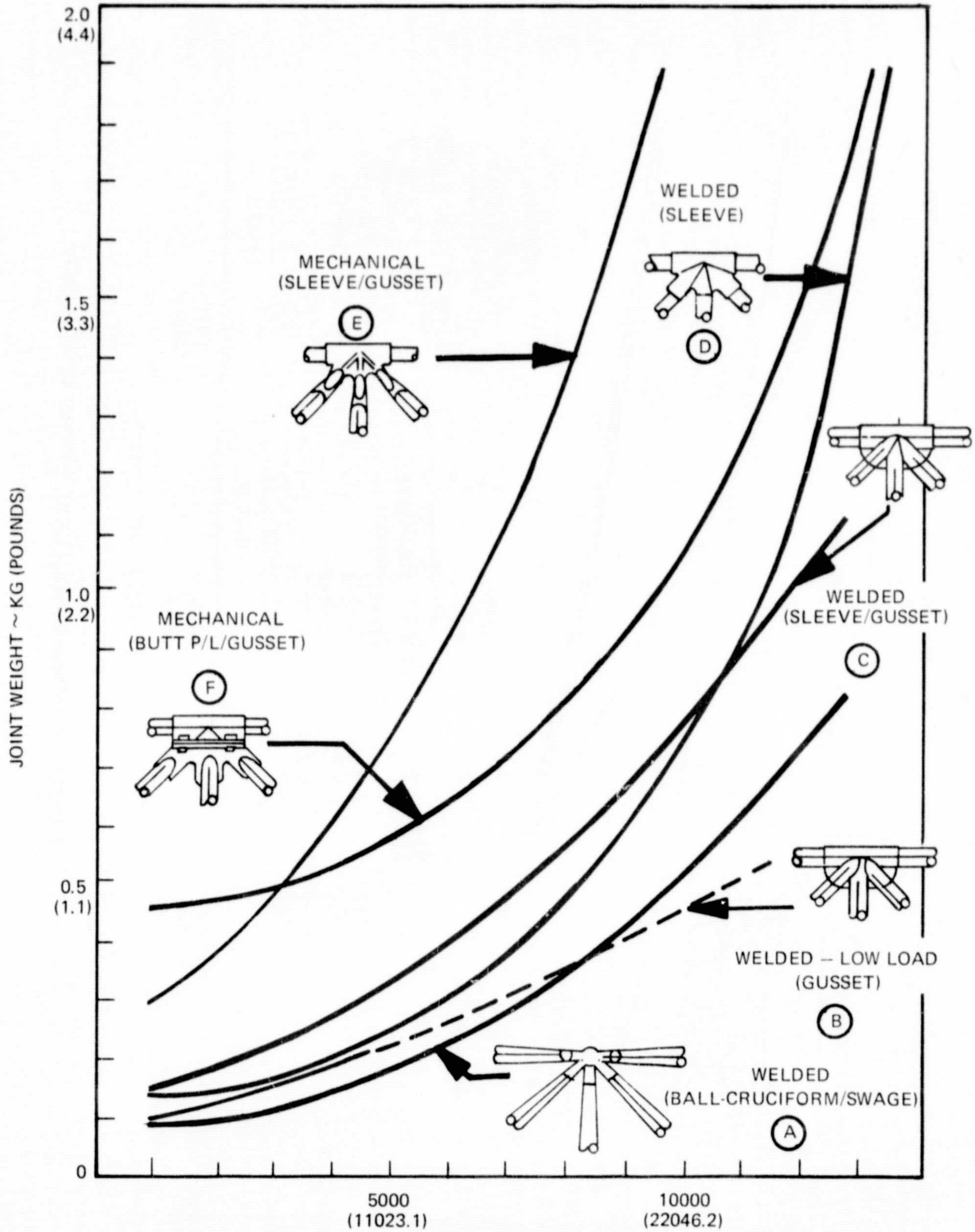


FIGURE 4.5 JOINT WEIGHT EFFICIENCY

this data the welded-ball cruciform joint (A) was selected for welded joints and the Butt Plate/Gusset Joint (F) was selected for mechanical joints.

Alternate materials to be considered in future study should include titanium and graphite composites to determine the cost/benefits of lighter but more costly construction. Titanium (3Al-2½V) has similar specific stiffness but higher cost than 2219 aluminum. However, joint weld strength and specific tensile strength of titanium could result in a lower structural weight at a higher unit cost. Graphite composites have a potential of 20% to 25% weight saving but also have a much higher unit cost and additional higher development risks.

LES stages structural weight is affected by the stage, payload and ASE interface. The weight impact is minimized by using ASE which has support points for both the stage and for the larger payloads. The LES modularized ASE cradle assemblies #1 through #4, shown later in Figure 5.10, provide variable geometry for supporting horizontal and vertical mounted stage/payloads. Large horizontal payloads are supported near the cg so that loads in the stage do not exceed the design condition where the stage supports a 200 kg (441 lbs.) payload without the additional ASE support at the payload cg. The ASE is configured with a walking beam including a sliding pivot joint to prevent Orbiter torsional and thermal deflections from loading the stage structure.

Adaptation configurations of SSUS-A, SSUS-D and the LES stage use existing SSUS ASE cradles which do not have additional support points for payloads. However, these configurations are used to launch small payloads weighing 200 kg (441 lbs.) or less which do not penalize the LES stage structure.

4.2.3 Stage Accuracy Requirements and Error Budgets

A pointing error budget of 2.5° (most stringent accuracy requirement) at apogee kick ignition coupled with altitude, inclination and velocity error accumulated throughout initialization deployment and flight will meet typical three sigma Scout payload delivery accuracies as depicted in Volume II Table 2-VI which was based on orbital mechanics analysis. The error budget for the guidance and control portion of the accumulated error are shown in Table 4-VIII.

TABLE 4-VIII LES ERROR BUDGET

ERROR SOURCE	3σ ERROR BUDGET (DEGREES)
Shuttle Cargo Bay Alignment Uncertainty	2.00
Shuttle IMU Error	0.5
LES Deployment Tip-Off	0.92
LES Guidance Error	0.5
LES ASE Alignment Uncertainty (Alignment between cargo bay and the stage)	0.5
RSS Value	2.37

4.2.4 Guidance Subsystem Selection

A trade study comparing the advantages and the disadvantages of spin stabilization versus 3-axis stabilization concluded the 3-axis stabilization approach was the optium guidance system for the LES configurations. Early in the study it had been assumed that a spin stabilized stage with an active nutation control, similar to that used in SSUS, would probably be adequate. However, additional analysis shows that the LES must have a more sophisticated guidance system than the SSUS. Table 4-IX is a comparison of the SSUS mission and the LES mission and shows that the requirements are essentially the same through perigee kick (PK) burn. The missions are different after this point in the flight. The SSUS mission is finished after perigee burnout but the LES mission continues through apogee kick (AK) burnout, de-spin, and separation. In the case of the LES, it must be maneuvered to the correct attitude for orbit injection just prior to ignition of apogee burn. Because of this attitude reorientation requirement, the LES must contain accurate information from the time it is ejected from the Shuttle until apogee burnout. The attitude reference must not accumulate more than an additional one (1) degree error at apogee burnout to remain within the error budgets established in paragraph 4.2.3. Table 4-X presents the advantages and disadvantages of spin stabilized versus 3-axis stabilized

TABLE 4-IX SSUS/LES MISSION COMPARISON

SSUS CONCEPT	LES CONCEPT
• Erect SSUS in Shuttle Cargo Bay • Maneuver Shuttle and Align SSUS to Required Perigee Velocity Vector • Spinup SSUS with ASE Spin Table • Eject SSUS from Shuttle at 1ft/sec Utilize Active Nutation Control for Cone Damping • Coast SSUS for 25 to 45 Minutes or More • Ignite SSUS PKM at Appropriate Time or Position • Separate Payload After PKM Thrust Tailoff • Payload Provides Required Reorientation for Orbit Injection and the AKM Burn	• Erect LES in Shuttle Cargo Bay • Maneuver Shuttle and Align LES to Required Perigee Velocity Vector • Spinup LES with ASE Spin Table • Eject LES from Shuttle at 1-2 ft/sec Utilize Active Nutation Control Provided by Guidance System for Cone Damping • Coast LES for 25 to 45 Minutes or More • Ignite LES Perigee Burn at Appropriate Time or Position • Rotate LES to less Than 1° of Proper Attitude for Final Orbit Injection • Ignite LES Apogee Burn - Maintain Attitude Error $\leq 1^\circ$ • Separate Payload After AKM Thrust Tailoff

TABLE 4-X LES SPIN STABILIZED VS 3-AXIS STABILIZED STAGES

LES G & C TYPE	ADVANTAGES	DISADVANTAGES
Spin-Stabilized with $\leq 1^\circ$ Attitude Error Accumulation and AKM Reorientation Capability	• Fewer RCS Thrusters (1 or 2) • Less RCS Control Impulse • Lower RCS Cost • Lower RCS Weight	• Requires spin balancing of both the stage and the payload • Requires spin table, spin motor(s) and slip-rings in the Shuttle cargo bay • Stage and payload must be designed to withstand centrifugal loads • Must be rotated (erected) in the Shuttle cargo bay and thereby requires additional cargo bay length • Requires air-bearing simulator for spin stabilization verification for each LES/payload combination prior to flight • Requires incorporation of payload de-spin system • Requires roll stabilized platform and more complex software • Inherently less accurate than a 3-axis system
3-Axis Stabilized with $\leq 1^\circ$ Attitude Error Accumulation and AKM Reorientation Capability	• No spin balancing required for stage or payload • No spin table, spin motor(s) or slip-rings required in Shuttle cargo bay • Stage and payload need not be designed for centrifugal loads • Need not be rotated (erected) in Shuttle cargo bay resulting in decreased cargo bay length	• More RCS thrusters (4 or more) • Higher RCS control impulse • Higher Cost RCS • Higher Weight RCS

TABLE 4-X LES SPIN STABILIZED VS 3-AXIS STABILIZED STAGES (CONT'D)

LES G & C TYPE	ADVANTAGES	DISADVANTAGES
3-Axis Stabilized with $\leq 1^\circ$ Attitude Error Accumulation and AKM Reorientation Capability (Continued)	• No air-bearing simulator required for preflight control verification • No payload despin system required • No roll stabilized platform required and simpler software • Inherently more accurate than spin stabilized system • Lower guidance system cost	

stages. The principal advantage of the spin stabilized stage is that it requires a simpler and lower cost reaction control system (RCS). The 3-axis system requires a more complex RCS but it requires a less complex guidance system, structure, and Shuttle ASE.

As shown in Table 4-XI, the development cost for the 3-axis stabilized system is $3,959,590 less than for the spin stabilized system and the recurring unit cost is $581,490 less. This cost benefit is valid for the 3-axis system provided that only four reaction control thrusters are required for control. It was assumed that four thrusters could be oriented to provide the three axis control. This four thruster RCS arrangement is shown in Figure 4.6. This is a valid concept provided that appropriate moment arms resulting in acceptable control accelerations are available in each of the LES/Payload configurations.

The 8-tank and the 4-tank bipropellant configurations with selected payloads were examined using the four (4) thruster RCS installed at 45 degrees with respect to the appropriate control axis and the attitude control Reaction Control System (RCS) fuel quantity was calculated. The Seasat B payload on the 8-tank bipropellant stage required 122 kg (270 lbs.) of RCS propellant assuming all worse case tolerances. This was the highest realistic fuel quantity required for the 8-tank configuration and was used for sizing the propellant tanks for the three-axis stabilized configuration. The RCS propellant requirement was subsequently reduced to 14 kg (30 lbs.) by orienting the attitude control thrusters to provide velocity change as well as attitude control.

Table 4-XII shows the stage unit cost and user cost for the spin stabilized versus the three-axis stabilized stage with the propellant tanks sized to accomodate the 122 kg (270 lbs.) of RCS fuel. The larger propellant tanks required for the three-axis configuration result in an increased stage length and an increased user cost.

The total comparative launch costs of the spin stabilized versus three-axis stabilized stages are shown in Table 4-XIII. These costs are based upon the 92 payloads included in reference missions B, C and D of the initial mission model. The total three-axis configuration launch costs are slightly less than the total launch costs for the spin stabilized configuration.

TABLE 4-XI COMPARATIVE COSTS OF SPIN STABILIZED & 3-AXIS STABILIZED STAGES

(THOUSANDS OF DOLLARS)

ITEM	SPIN STABILIZED		SPIN STABILIZED WITH NUTATION CONTROL		SPIN STABILIZED WITH NUTATION AND ORIENTATION CONTROL		3-AXIS STABILIZATION WITH ORIENTATION CONTROL	
	NON RECURRING	RECURRING	NON RECURRING	RECURRING	NON RECURRING	RECURRING	NON RECURRING	RECURRING
ACS				134.0	9566.21	323.48	6377.47	252.59
ACCELEROMETERS						(26.3)		(26.3)
ATTITUDE GYROS						(65.0)		(65.0)
COMPUTER						(122.3)		(85.5)
SOFTWARE					(1500)		(1000)	
ROLL STABILIZED PLATFORM					(1300)			
RCS								
100# THRUST UNIT(S)					283	65.5	825	165.0
GSE								
AIR BEARING SIMULATOR			100.0	1.0	100.0	1.0		
TEST PROCEDURE			4.25		4.25			
ASE								
SPIN BEARING, SLIP-RINGS SPIN-MOTOR	851.0 (3 Sets)		851.0 (3 Sets)		851.0 (3 Sets)			
OTHER								
SPIN BALANCE FACILITY & LABOR		9.0		9.0		9.0		
SPIN BALANCE ADAPTER	18.0		18.0		18.0			
SPIN BALANCE PROCEDURE	14.0		14.0		14.0			
INERT TANKS (8) & FILL	325.6		325.6		325.6			
SPIN TABLE CARGO BAY LT CHARGE(15"x$40417/in.)		600		600		600		
TOTAL	1208.6	609.0	1312.85	744.0	11162.06	998.98	7202.47	417.59

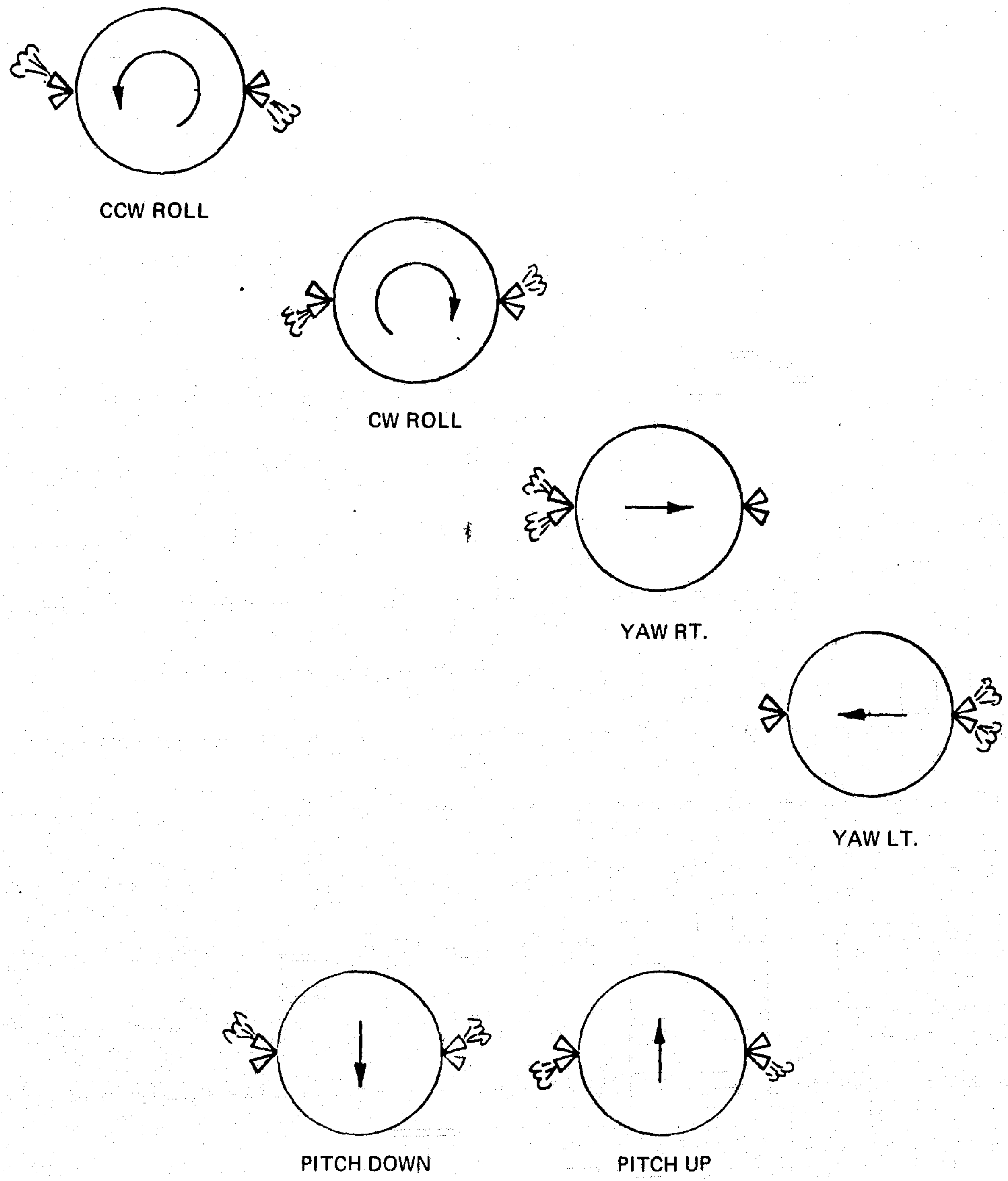


FIGURE 4.6 THRUSTER ARRANGEMENT FOR 4 THRUSTER REACTION CONTROL

TABLE 4-XII STAGE UNIT COST AND USER COST COMPARISON FOR SPIN VS 3-AXIS STABILIZATION

	SPIN STABILIZED	3-AXIS STABILIZED
G & C PROPELLANT WEIGHT (LB)	13.6 kg[1] (30 lb)	124.4 kg[2] (270 lb)
TANK DIAMETER (IN)	0.61 m (1.99 ft)	0.63 m (2.07 ft)
STAGE LENGTH (IN)	0.70 m (2.31 ft)	0.73 m (2.39 ft)
STAGE UNIT COST ($\$ \times 10^6$)		
8-TANK BIPROPELLANT VERSION ($\$ \times 10^6$)	1.41	1.42
4-TANK BIPROPELLANT VERSION ($\$ \times 10^6$)	1.21	1.22
STAGE UNIT USER COST ($\$ \times 10^6$) (BASED ON STAGE LENGTH)	1.12	1.16

NOTES: (1) 13.6 kg (30 lbs.) ESTIMATE IS PROBABLY LOW

(2) 124.4 kg (270 lbs.) BASED ON CONTROL REQUIREMENTS FOR SEASAT B PAYLOAD (OLD MODEL)

The total launch cost comparison of Table 4-XIII does not include costs associated with the stage and payload structural and subsystem integrity necessary to accomodate centripetal loads of a spin stabilized configuration. The stage and each of the payloads would have to be designed with heavier structure to accomodate the increase loads. Also not addressed in this trade study are the user costs associated with additional length required to accomodate erection of the stage and payload to the spin-up and launch position in the cargo bay. The typical additional cargo bay length required is 30 inches and is equivalent to a user charge of approximately $1,212,500, for each payload ($112M for 92 payloads). The 3-axis system can be deployed from the cargo bay without erecting, therefore, no additional bay length for launch is required. This, in fact is the primary cost benefit associated with the 3-axis system and overshadows previously discussed cost advantages. The additional costs associated with the structural integrity and spin balancing will further increase the cost of the spin stabilized stage compared to the three-axis stabilized stage, thus making the latter configuration even more attractive.

An additional advantage of the three-axis stabilized system is its ease of adaptability to various payload size and accuracy requirements. The system checkout, both in-plant and at the launch site, is also simpler for the three-axis system. The spin stabilized system requires careful gain tailoring to accomodate various payload inertia ratios and stability verification using an airbearing supported simulator for each payload.

Based upon the indicated cost benefits together with the additional intangable benefits resulting from the simpler structural design requirements and the simpler stage/payload processing, the three-axis stabilized configuration was selected for incorporation in all LES designs.

A survey of available flight-qualified guidance systems that could satisfy the LES requirements resulted in the selection of a single system: the Teledyne Systems Company system presently being developed to meet the NASA Scout Phase VIII guidance and control requirements. This system, utilizing the Teledyne SDG-5 tuned rotor gyros, has a three sigma

TABLE 4-XIII COMPARATIVE TOTAL LAUNCH COSTS OF SPIN STABILIZED VS 3-AXIS STABILIZED STAGES

	SPIN STABILIZED ($ x 10^6)	3-AXIS STABILIZED ($ x 10^6)
G & C DEVELOPMENT COSTS	11.2	7.2
STAGE UNIT COSTS (92 PAYLOADS)	118.0	119.5
SHUTTLE USER CHARGE (92 PAYLOADS)	158.2	106.7
TOTAL	287.4	233.4

drift rate of only 0.045 degrees/hour. The accumulated drift would be 0.09 degrees/hour maximum after a LES flight time of 1-1/2 to 2 hours which is considerably less than 0.5 degree guidance error allowable, shown in Table 4-VIII.

The LES/Scout Phase VIII guidance unit consists of two, two-degree-of-freedom gyros, three single axis accelerometers, supporting electronics, and a stored program general purpose digital computer. This unit measures vehicle attitudes, accelerations, and performs the necessary computations, transformations, and signal processing for vehicle control and event sequencing. Also contained within the guidance unit is all the signal conditioning and formatting necessary for the telemetry system. The computer contains 10K, 16 bit words of memory and uses fixed point arithmetic. The functional schematic of this system is shown in Figure 4.7.

The development testing required for the selected guidance system will be minimal since it is an existing, qualified system. The only development testing that will be necessary is that associated with verifying the performance and integrity of the input/output modifications necessary to adapt the system to the LES application. Most of the development effort will be involved with the modification of existing software and the generation of new software compatible with the LES system requirements. All input/output hardware modifications and the development testing necessary to prove these modifications would be conducted by the guidance system manufacturer. Included in this vendor development effort will be the software required to perform the development testing and the system acceptance testing at the vendor's facility. The LES flight software and any LES contractor in-house test software would be developed by the LES contractor and the development testing assumed to be accomplished in the LES contractor's Software Development Laboratory.

4.2.5 Reaction Control System Requirements

The Reaction Control System (RCS) thrust level and motor location was determined to ensure that the proper control authority would be available for all combinations of payloads and LES modular configurations. Worse case tolerance build-ups were assumed in the analysis to insure that the RCS authority would be conservative. In each case it was assumed that

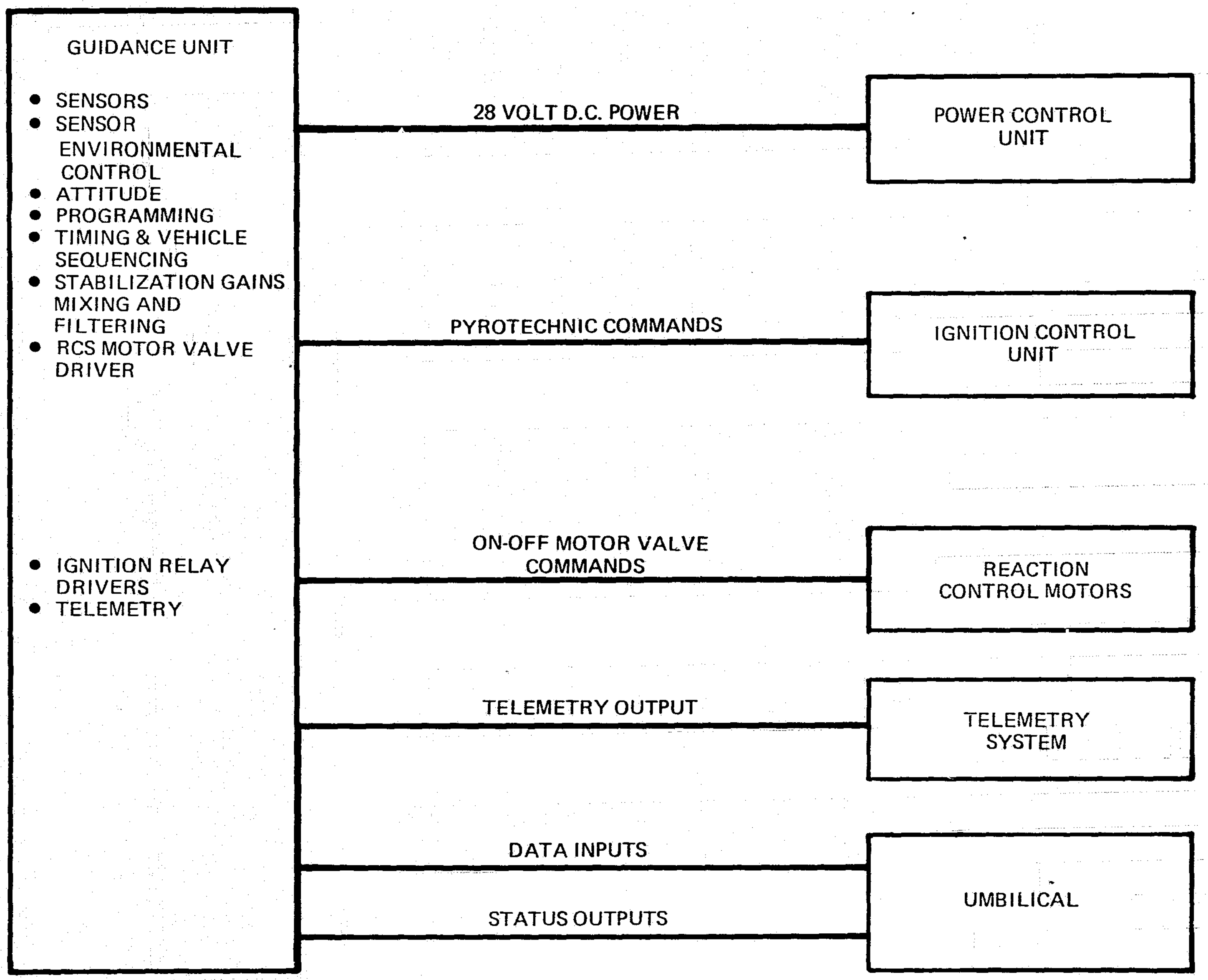


FIGURE 4.7 GUIDANCE SYSTEM SCHEMATIC

tolerances such as main propulsion thrust misalignment or LES center-of-gravity offset, were additive and that the maximum value of each occurred at the same time. Both the control forces required and the system dynamic response were considered in the analysis along with the extremes of propellant weights at the beginning and end of each flight.

Table 4-XIV shows typical control force requirements resulting from this analysis. Standard 444.8N (100 lbs.) force thrusters were selected for the bipropellant configuration. These are mounted at a 10° cant angle to the LES centerline for roll control as well as pitch and yaw control in all bipropellant stages except for the "adaptation" concepts. For the SSUS-D and SSUS-A "adaptations" the thrusters were oriented perpendicular to the thrust axis to prevent the RCS plume from impinging on the booster. Analysis of the SSUS-A "adaptation" showed that the RCS control force required to overcome the maximum thrust misalignment forces of the SSUS-A motor was 1103.1N (248 lbs.). In all other bipropellant stage cases, less than 444.8N (100 lbs.) was required. Typical stage characteristics used in the bipropellant analysis are as follows:

- Configuration Characteristics: 4 RCS Thrusters located 965 mm (38 in.) from C_L. Each thruster canted at 10° for roll control.
- Main propulsion thrust misalignment = 0.5°
- Main propulsion installation tolerance = 0.25°
- Main propulsion thrust = 3870N (870 lbs.) ±3%
- RCS Thruster thrust misalignment = 1.0°
- RCS Thruster installation tolerance = 0.25°
- RCS Thruster sizes available are 111.2N (25 lbs.) and 444.8N (100 lbs.)
- CG offset = 0.635 mm (0.25 in.) at burnout

4.3 PROPULSION SUBSYSTEM REQUIREMENTS AND COMPONENT SELECTION

This section summarizes the propulsion studies conducted, requirements imposed and derived, and descriptions of resulting component hardware selections. Included are both the modular bipropellant and modular monopropellant propulsion approaches selected in Task 2. The majority of

TABLE 4-XIV REACTION CONTROL SIZING

CONFIGURATION	PAYLOAD		RCS CONFIGURATION		RCS FORCE REQUIRED N (LB)	RCS FORCE SELECTED N (LB)
	NO.	WEIGHT KG (LB)	PARALLEL TO C_L	PERPENDICULAR TO C_L		
8-Tank Bipropellant	21	3400 (7497)	X		193.5(43.5)	444.8(100)
8-Tank Bipropellant	9	300 (661)	X		153.5(34.5)	444.8(100)
4-Tank Bipropellant	43	3110 (6857)	X		221 (49.7)	444.8(100)
4-Tank Bipropellant	1	310 (683)	X		157 (35.3)	444.8(100)
4-Tank Bipropellant Vertical Configuration	5	900 (1984)	X		152 (34.2)	444.8(100)
4-Tank Bipropellant Vertical Configuration	52	200 (441)	X		219 (49.2)	444.8(100)
4-Tank Bipropellant/SSUS-D	48	200 (441)		X	439 (98.7)	444.8(100)
4-Tank Bipropellant/SSUS-A	52	200 (441)		X	1099 (247)	1344.4(300)
2-Tank Monopropellant	1	200 (441)	X		N/A	622.7(140)
2-Tank Monopropellant	43	3110 (6857)	X		N/A	622.7(140)

the Task 3 propulsion-related conceptual design effort involved system resizing based on refinement of stage weights, packaging studies, thermal analyses, examination of alternate components, and definition of system characteristics.

4.3.1 Requirements

The low-cost objective of this study for a conceptual propulsion stage was to develop a configuration that would have a short package length, low development cost, low recurring cost and a low risk factor. Configurations selected and components baselined in Task 2 met these requirements, therefore they were retained in Task 3 and were refined to establish the conceptual designs. A summary of requirements and design objectives are in Table 4-XV. Sizing studies conducted for the low energy regime requirements of Figure 4.1 resulted in the propulsion system characteristics in Table 4-XVI and Table 4-XVII for the bipropellant and monopropellant systems, respectively. A typical schematic of the bipropellant system is illustrated in Figure 3.7 (Volume II). Table 4-XVIII provides the bipropellant system parts lists, quantities and weights consistent with Table 4-XVI.

4.3.2 Main Thrusters

The desired thruster features are: short length, compact size, moderate thrust level, high impulse thruput, low cost, lightweight, flexible duty cycle, low feed pressure, suitability for buried packaging, off-the-shelf proven, Shuttle compatible and low attendant risk. Considering the above features, the Marquardt R-40A Space Shuttle Reaction Control Thruster was baselined as the LES bipropellant PK/AK thruster and the Rocket Research Corp. MR-104 Mariner Jupiter/Saturn '77 RCS Thruster was baselined as the LES monopropellant PK/AK thruster.

Thirty-eight R-40A thrusters are used in the reaction control subsystem and the aft propulsion subsystem of the Shuttle Orbiter. The unit provides the control force for vehicle attitude control and 3-axis translation. The thruster is designed to provide high reliability, minimum weight, high performance and minimum maintenance and servicing. Coated columbium C-103 alloy construction is used for the combustion chamber and nozzles. A dynaflex insulation system is retained by a steel mesh enclosure. The individual propellant fluid pressure actuated valves provide

TABLE 4-XV

PROPULSION SYSTEM REQUIREMENTS AND DESIGN OBJECTIVES

- SHORT PACKAGE LENGTH
- LOW DEVELOPMENT AND RECURRING COST
- LOW RISK
- 3-AXIS STABILIZED
- MAIN PROPULSION AND REACTION CONTROL SYSTEMS SHARE SAME TANKAGE
- PREPACKAGED PROPELLANT TANKAGE WITH LONG TERM STORAGE POTENTIAL AND NO SERVICING AT LAUNCH SITE
- AVOID TWO-PHASE FLOW TO THRUSTERS
- PROVEN HARDWARE OR TECHNOLOGIES WITH LITTLE OR NO COMPONENT DEVELOPMENT OR QUALIFICATION TESTING
- PROPELLANT TEMPERATURE -4° TO 49°C (25° TO 120°F) --BIPROPELLANT
1.4° TO 49°C (34.5° TO 120°F) --MONOPROPELLANT
- MODERATE ACCELERATION/THRUST LEVELS
- EQUAL BIPROPELLANT TANK VOLUMES FOR FUEL AND OXIDIZER
- DESIGN VERIFICATION TESTING AT SYSTEM LEVEL TO DEMONSTRATE NO DEGRADATION FOR INTEGRATED SYSTEM AND SUITABILITY FOR QUALIFICATION
- QUALIFICATION TESTING AT SYSTEM LEVEL TO PROVE COMPLETE SHUTTLE SUITABILITY, PERFORMANCE ADEQUACY AND READINESS FOR PRODUCTION
- COMPATIBLE WITH A LARGE RANGE OF VELOCITY CHANGE REQUIREMENTS (MODULARITY)

TABLE 4-XVI MODULAR BIPROPELLANT 8-TANK SYSTEM CHARACTERISTICS

WEIGHTS kg (lbm)	Main Propulsion System (excludes reaction control thrusters, lines and instrumentation) 2147 (4733) • Useable Perigee/Apogee ΔV Propellant 1597 (3520) • Useable RCS ΔV Propellant 72.5 (160) • System Inerts and Unuseable Propellants and Pressurant 477.5 (1053)
CONFIGURATION cm (in.)	• 1 - R-40A Bipropellant Thruster Isp - 2746 N-sec/kg (280 lbf-sec/lbm) Steady State • 8- Conospherical Propellant Tanks - O.D. 63.5 (25) - Length 101.6 (40) • 8- Spherical Pressure Tanks - O.D. 39.4 (15.5) • 4- R-4D Bipropellant Reaction Control Thrusters - Share Main Propellant

TABLE 4-XVII MODULAR MONOPROPELLANT 8-TANK SYSTEM CHARACTERISTICS

WEIGHTS kg(lbm)	Main Propulsion System (includes reaction control system since integral with PK/AK system) 3289 (7252) ● Useable ΔV Propellant 2412 (5318) ● System Inerts and Unuseable Propellants and Pressurant 877 (1934)
CONFIGURATION cm(in.)	● 4 - MR-104 Monopropellant AK/PK and Reaction Control Thrusters Isp = 2046 N-sec/kg (208.6 lbf-sec/lbm) Installed Steady State ● 8 - Conospherical Propellant Tanks - O.D. 74.9 (29.5) - Length 120.1 (47) ● 8 - Spherical Pressurant Tanks - O.D. 59.7 (23.5)

TABLE 4–XVIII TYPICAL PROPULSION AND REACTION CONTROL EQUIPMENT LIST
• BIPROPELLANT

SYSTEM	ITEM NO.	COMPONENT	MFGR.	DESIGNATION PART/NO.	CONFIGURATION COMPONENT DETAILS WEIGHT KG	WEIGHT LBM	PRIOR APPLICATION	8 TANK QUANT PER SYSTEM	8 TANK WT/SYSTEM KG	8 TANK WT/SYSTEM LBM	4 TANK QUANT PER SYSTEM	4 TANK WT/SYSTEM KG	4 TANK WT/SYSTEM LBM
Main Prop. WBS 0224	1	Pressurant Tank	PSI	80074	10.57	23.3	Lunar Landing Training Veh.	8	84.55	186.4	4	42.28	93.2
	2	Pressurant Squib Valve	Quantic	1512-01	.73	1.6	SCOOP	1	.73	1.6	1	.73	1.6
	3	Overboard Vent Squib Valve	Quantic	1512-01	.73	1.6	SCOOP	1	.73	1.6	1	.73	1.6
	4	Pressurant Fill & Vent Fitting	Purolator	7542808	.23	.5	SCOOP	1	.23	.5	1	.23	.5
	5	Thrust Neutralizer	New	New	.82	1.8	New	1	.82	1.8	1	.82	1.8
	6	Press. Regulator	New	New	1.59	3.5	New	1	1.59	3.5	1	1.59	3.5
	7	Check Valve	James Pond & Clark	8528 Mod	1.36	3.0	New	2	2.72	6.0	2	2.72	6.0
	8	Relief Valve	Parker Hannifin	5762052	.77	1.7	Shuttle RCS	2	1.54	3.4	2	1.54	3.4
	9	Tank Squib Vlv.	Quantic	1512-01	.73	1.6	SCOOP	16	11.61	25.6	8	5.81	12.8
	10	Propellant Tank	ARDE	New	30.03	66.2	New	8	240.23	529.6	4	120.11	264.8
	11	Propellant Fill & Drain	Future Craft	900287	.14	.3	SCOOP	16	2.18	4.8	8	1.09	2 4
	12	Filter	WINTER	15241-690-1	.59	1.3	SCOOP	16	9.43	20.8	8	4.72	10.4
	13	Manual Drain Vlv.	Future Craft	30485-3	.68	1.5	SCOOP	2	1.36	3.0	2	1.36	3.0
	14	Thruster	Marquardt	R-40A	9.75	21.5	Shuttle RCS	1	9.75	21.5	1	9.75	21.5
	15	Line Set	New	New	Varies		New	1	10.34	22.8	1	5.49	12.1
	16	He. Pres. Xducer	Teledyne	2403-4000-1	.09	.2	SCOOP	1	.09	.2	1	.09	.2
	17	Pc Pres. Xducer	Teledyne	2403-200-1	.09	.2	SCOOP	1	.09	.2	1	.09	.2
	18	Thermister	Stock Item	–	.05	.1	–	1	.05	.1	1	.05	.1
	19	Thermocouple	Stock Item	Iron/Constant.	.05	.1	–	3	.14	.3	3	.14	.3
	20	Thermocouple	Stock Item	Chromel/Alum.	.05	.1	–	1	.05	.1	1	.05	.1
	21	Cartridges	SOS	NSI-1	Incl. in Sq. Vlv.		SCOOP	36	–	–	20	–	–
	22	Pressurant	–	Helium	–	–	–		8.62	19.0		4.31	9.5
	23	Oxidizer	–	N_2O_4	–	–	–		1079.3	2379.4		539.6	1189.7
	24	Fuel	–	MMH	–	–	–		674.6	1487.1		337.3	743.5
Reaction Control WBS 0225	1	Reaction Contr. Thruster	Marquardt	R-4D	2.40	5.3	SPP/Apollo	4	9.62	21.2	4	9.62	21.2
	2	RCS Line Set	New	New	Included in Item 15		New	1	–	–	1	–	–
	3	RCS PC Xducer	Teledyne Tabor	2403-200-1	.09	.2	SCOOP	4	.36	.8	4	.36	.8
	4	RCS Thermocouple	Stock Item	Iron/Constant.	.05	.1	–	4	.18	.4	4	.18	.4

fast and repeatable response, while the unlike doublet injectors provide rapid and efficient combustion. The unit has demonstrated specific impulse in excess of 2795 N-sec/kg (285 lb_f-sec/lb_m). Design requirements of the thruster are shown in Table 4-XIX. A cross-section of the thruster is illustrated in Figure 4.8. The R-40A qualification testing is projected for completion in 1979. No supplemental qualification testing is expected.

The MR-104 thruster provided pitch and yaw control for the Mariner Jupiter/Saturn 1977 (MJS77) mission. The thruster consists of the thrust chamber, normally-closed solenoid propellant valve, pressure transducer, heaters on valve and engine, engine thermal shield and support structure. The thrust chamber is a Tungsten Inert Gas (TIG) welded assembly of HASTELLOY B and 347 stainless steel. The catalyst bed is of the radial outflow type consisting of two concentric annular sections with the inner containing 25 to 30 mesh and the outer 14 to 8 mesh Shell 405 catalyst. The valve is all welded and incorporates a soft poppet and AFE-411 elastomer seat. This thruster has demonstrated a propellant total impulse thruput of 578,266 N-sec (130,000 lb_f-sec) and is capable of meeting a thruput of 1.779 x 10^6 N-sec (400,000 lb_f-sec). Performance and environmental requirements are given in Table 4-XX. The thruster is illustrated in Figure 4.9. Supplier design verification tests to the required thruput impulse would be necessary.

4.3.3 Propellant Tankage

Propellant tankage requirements and desired features are: conospherical or cylindrical shape, lightweight, low cost, low-cost packaging capability, prepackaging capability with long-term storage, single-phase propellant transfer capability under varying loads and maneuvers, center-of-gravity control potential, operating capability under 3-axis environment, propellant compatibility, and tankage of either off-the-shelf or of a size range for which proven technology exists. These requirements resulted in baselining of the ARDE, Incorporated ring-stabilized diaphragm/cryoformed propellant positive expulsion tankage approach. ARDE experience with the ring stabilized diaphragm of conospherical shape has ranged from

TABLE 4-XIX

R-40A DESIGN REQUIREMENTS AND CHARACTERISTICS

(BIPROPELLANT THRUSTER)

OPERATING LIFE	- 50,000 cycles 20,000 secs.
USEFUL LIFE	- 10 years
OXIDIZER	- MON-3 (SE-S-0073) (N_2O_4 & NO)
FUEL	- MMH (SE-S-0073)
MIXTURE RATIO	- 1.6 ± .032
THRUST VACUUM	- 3879 N @ 1.641×10^6 N/m^2) (872 lb$_f$ @ 238 psia)
PROPELLANT DELIVERY PRESSURE	- $1.207 \times 10^6 - 1.802 \times 10^6$ N/m^2 (175 - 264 psia)
PROPELLANT DELIVERY TEMPERATURE	- 4° - 38°C (40° - 100°F)
VALVE VOLTAGE	- 21 - 32 VDC
CHAMBER PRESSURE	- 1.048×10^6 N/m^2 (152 psia)
EXIT NOZZLE AREA RATIO (UNSCARFED)	- 22:1
MAXIMUM RUN DURATION	- 500 seconds
NUMBER OF MISSIONS	- 100
STABILITY	
HIGH FREQUENCY	- Per CPLA 247
LOW FREQUENCY	- ± 5%
MAXIMUM OUTER WALL TEMPERATURE	- 177°C (350°F)
SPECIFIC IMPULSE	- 2746N-sec/kg (280 lb$_f$-sec/lb$_m$) steady state - 1863N-sec/kg (190 lb$_f$-sec/lb$_m$) pulsing

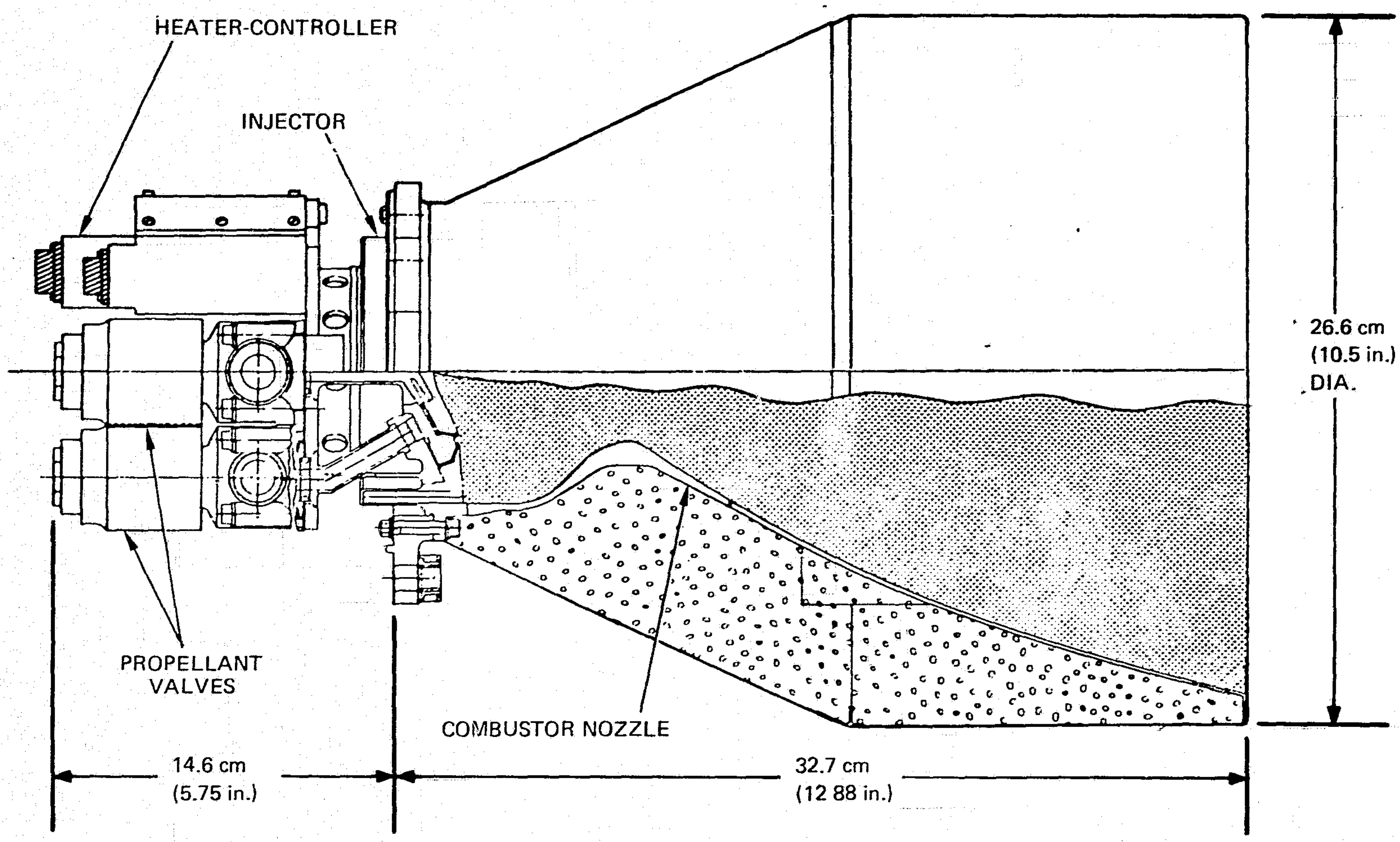


FIGURE 4.8 R-40A THRUSTER

TABLE 4-XX

MR-104 THRUSTER PERFORMANCE AND ENVIRONMENTAL REQUIREMENTS AND CHARACTERISTICS

(Monopropellant Thruster)

FEED PRESSURE	- 3.068×10^6 to 1.517×10^6 N/m^2 (445 to 220 psia)
THRUST LEVEL	- 623 to 343 N (140 to 77 lb_f)
MINIMUM PULSE WIDTH	- 40 ms
DUTY CYCLES	- Unlimited
RESPONSE TIME	- Signal to 90% peak pulse thrust $\leq$ 40 ms
DECAY TIME	- Signal to 15% peak pulse thrust $\leq$ 35 ms
THERMAL ENVIRONMENT	- 5° - 75°C (41° to +167°F)
RANDOM VIBRATION	- 11.1 g's rms (0.10 g^2/h_z)
MINIMUM VACUUM SPECIFIC IMPULSE	- 2157 N-sec/kg (220 lb_f-sec/lbm)
TOTAL IMPULSE	- 444,820 N-sec (100,000 lb_f-sec)
FUEL	- MIL-P-26536-Ammend. 1 Mod (N_2H_4)
VALVE VOLTAGE	- 26 to 34 VDC
CHAMBER PRESSURE	- 1.806×10^6 to 1.007×10^6 N/m^2 (262 to 146 Psia)
EXPANSION RATIO	- 50:1

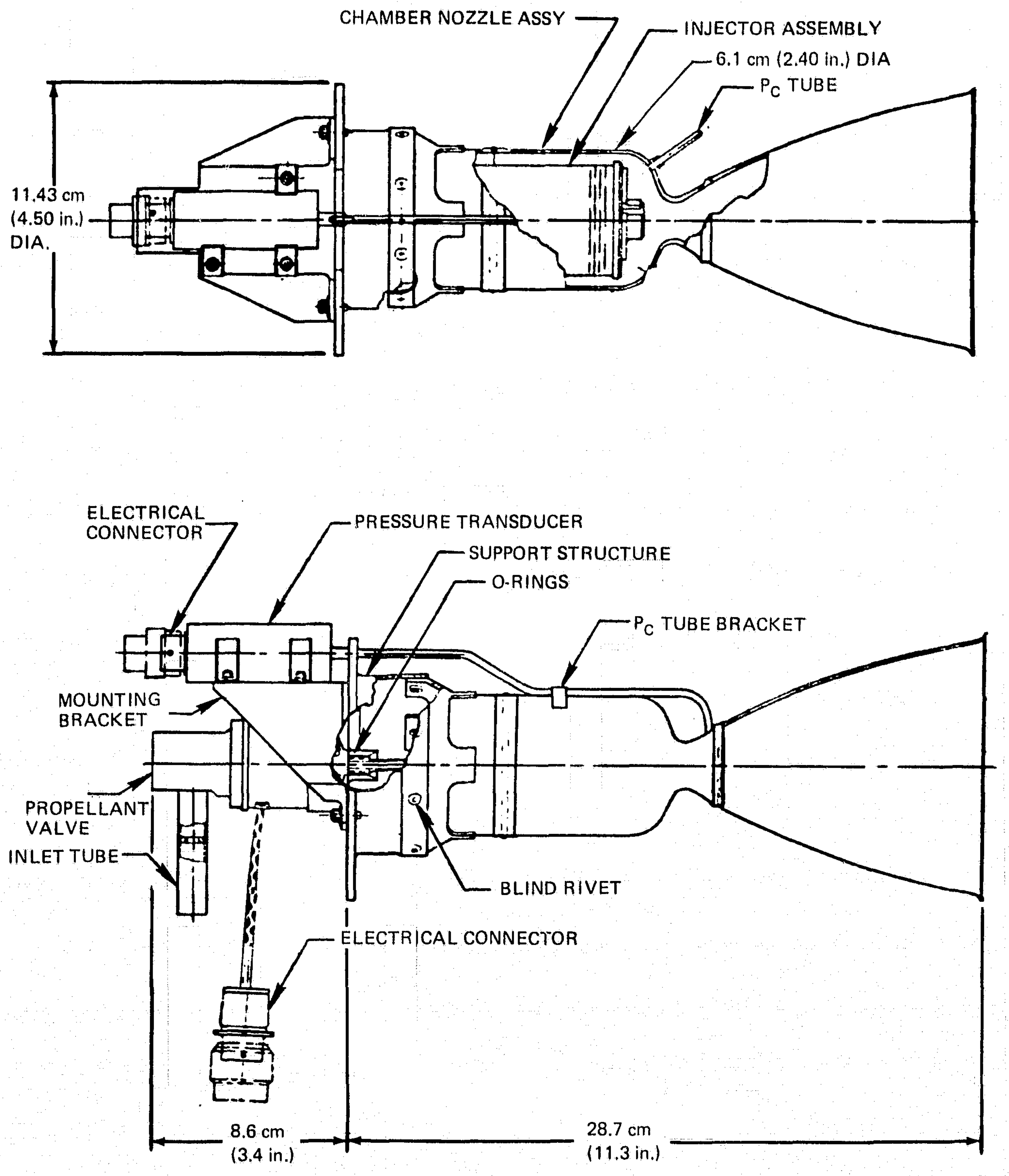


FIGURE 4.9 MR-104 THRUSTER

from sizes of 84 to 21 cm (33 to 8.4 inches) in diameter on programs such as WS-120A/PBPS, AFRPL Storability/Compatibility Test, Atmospheric Explorer and other missile programs using hydrazine.

The metallic diaphragm is constructed with support control rings nickel/gold brazed at various positions along its surface. The diaphragm is hydroformed from 304L stainless steel sheet stock with the rings of 308L stainless. The propellant tank shell is constructed from 301 and 304L stainless steel sheet stock. The tank shape is achieved by hydroforming into female dyes at ambient and cryogenic temperatures. A 304L girth ring is attached to each shell half and the final weld joins both halves and diaphragm. For mounting/attachment purposes, end supports are provided at the tank poles. Cleaning and propellant loading is accomplished through bosses on the shell dome surface. Figure 4.10 illustrates the propellant tankage configuration. Tankage design verification and qualification tests would be required.

4.3.4 Pressurant Tankage

Pressurant tankage requirements and desired features are: spherical shape, lightweight and low unit cost. For the monopropellant system these requirements were satisfied by an off-the-shelf Pressure Systems, Inc. (PSI) 6 AL-4V spherical tank and a new PSI 6AL-4V spherical tank. The existing bipropellant tank is identified as PSI part number 80074-1. This tank is spherical, contains 28,530 cm^3 (1741 in^3) internal volume, is 39.4 cm (15.5 inches) outside diameter, weighs 10.57 kg (23.3 lbm) and has operating, proof and burst pressures of 27.58×10^6, 41.37×10^6 and 55.16×10^6 N/m^2 (4000, 6000, and 8000 psig), respectively. The tank was originally developed for the Lunar Landing Training Vehicle. The new monopropellant tank would be fabricated by PSI using an existing forging die. The tank is spherical, contains 102,583 cm^3 (6260 in^3) volume, is of 59.7 cm (23.5 inches) outside diameter, weighs 40.0 kg (88.2 lbm) and has operating, proof and burst pressures of 24.82×10^6, 37.23×10^6, and 49.64×10^6 N/m^2 (3600, 5400 and 7200 lb_f/in^2), respectively.

4.3.5 Reaction Control System

Paragraph 4.2.5 identifies the reaction control system control force requirements. The Marquardt R-4D Apollo/LEM thruster was baselined as the LES bipropellant RCS thruster. For the monopropellant system,

TANK PARAMETER	BIPROPELLANT	MONOPROPELLANT
VOLUME		
●MAXIMUM ~ m^3 (ft^3)	0.2016 (7.12)	0.3336 (11.78)
●USABLE ~ m^3 (ft^3)	0.1821 (6.43)	0.3013 (10.64)
●VOLUMETRIC EFFICIENCY ~ %	98	98
●EXPULSION EFFICIENCY ~ %	97	97
●ULLAGE ALLOWANCE ~ %	5	5
PRESSURE ~ N/m^2 (psia)		
●MAXIMUM	2.07×10^6 (300)	3.45×10^6 (500)
●OPERATING	1.79×10^6 (~260)	3.17×10^6 (~460)
●PROOF	3.10×10^6 (450)	5.17×10^6 (750)
●BURST	4.14×10^6 (600)	6.89×10^6 (1,000)
WEIGHT ~ Kg (lbm)	30.03 (66.2)	48.63 (107.2)

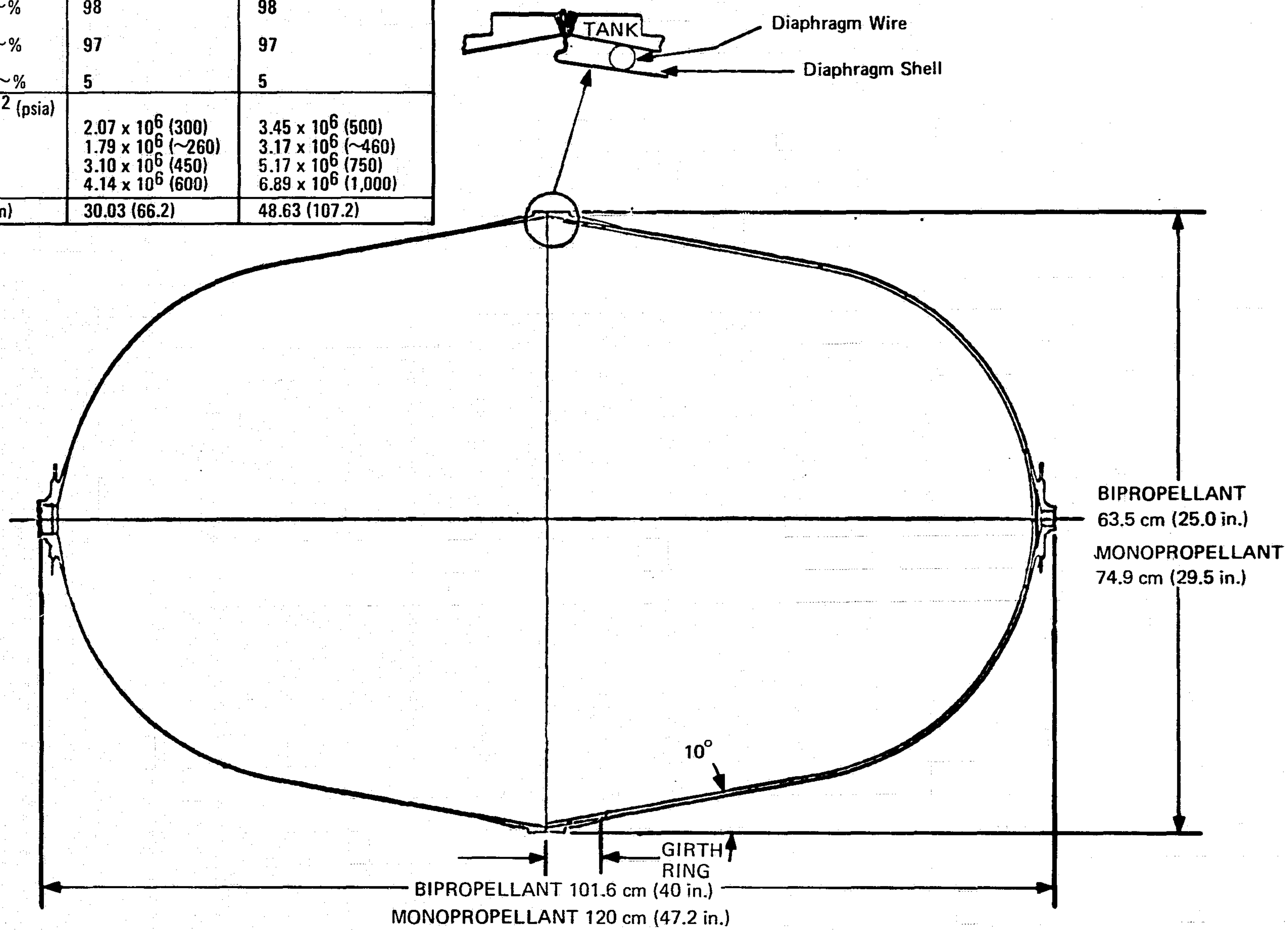


FIGURE 4.10 PROPELLANT TANKAGE

the PK/AK thrusters discussed in paragraph 4.3.2 provide RCS control requirements.

The R-4D engine was qualified for the Apollo Service Module, Lunar Excursion Module, Lunar Orbiter and the Special Defense Program (Reference 56) and System Technology Office Confirmation of Optical Phenomenology (SCOOP) (Reference 57). The engine design incorporates the following features. The radiation and film-cooled combustion chamber is fabricated from forged molybdenum and coated with molybdenum disilicide which provides oxidation protection. The radiation-cooled, ribbed L-605 nozzle extension minimizes weight. Propellant flow control is provided by two normally closed, dual coil, solenoid actuated, coaxial flow, poppet type valves with integral propellant flow regulation orifices. The fixed orifices are incorporated into each valve to provide the correct propellant pressure drops at the design inlet pressure to obtain the required mixture ratio and thrust level. Attachment of these valves to the injector head provides a metal-to-metal spherical seat plus double static seals to eliminate all external leakage paths. Table 4-XXI and Figure 4.11 show the R-4D characteristics and cross-section, respectively. It is anticipated that additional R-4D testing would not be required for the LES program.

4.4 OTHER SUBSYSTEM REQUIREMENTS AND COMPONENT SELECTION

The structural, guidance, and propulsion selections are discussed in paragraphs 4.2 and 4.3 Other subsystems required for stage design include telemetry, electrical power, ignition and thermal protection.

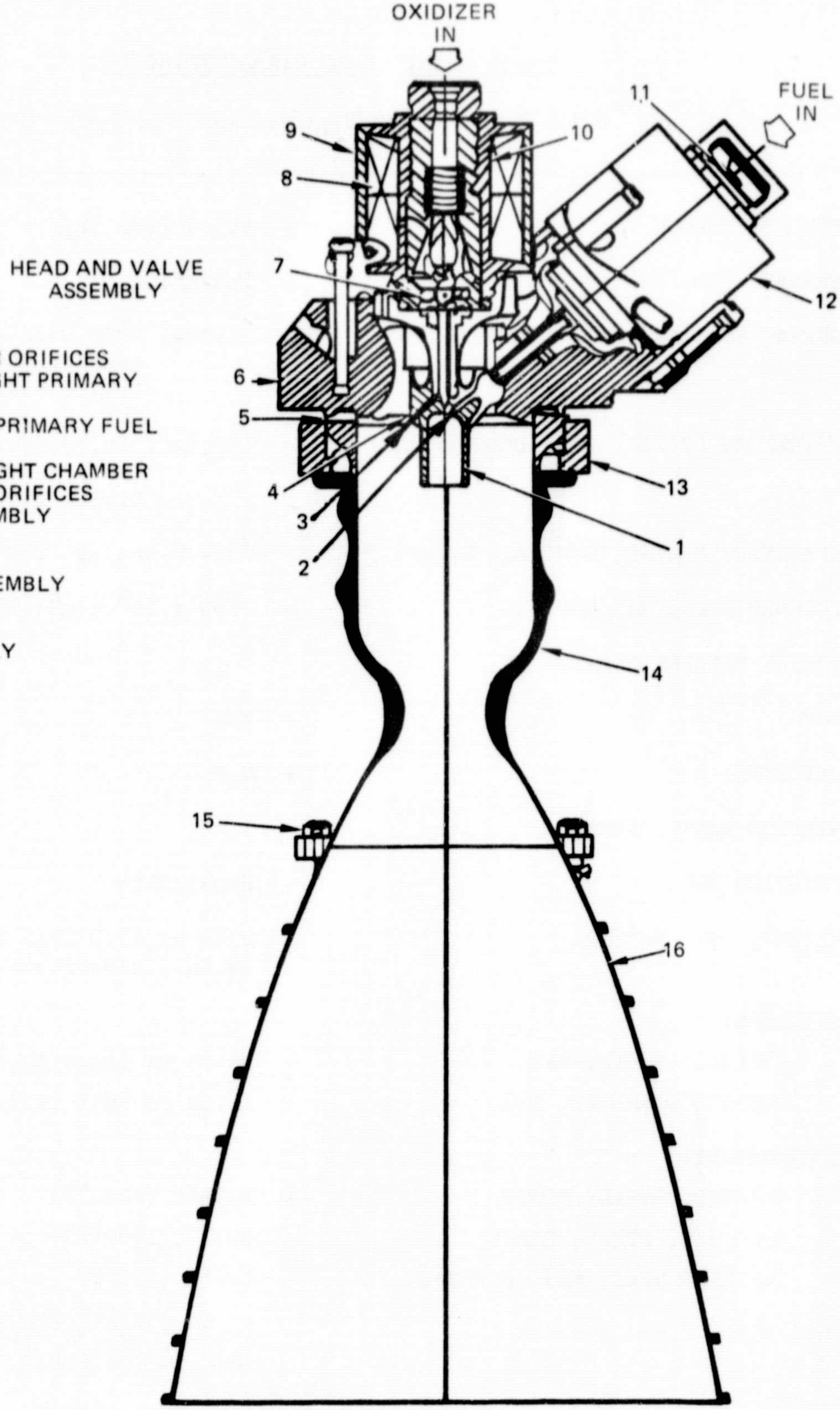
4.4.1 Telemetry

Performance monitoring of the subsystems within the LES is provided by the LES Telemetry (TM) System. The TM system selected consists of a Conic Model CTM-UHF-310E transmitter which has a power output of 8 Watts at S-band (2200-2300 MHz), four omni directional TECOM Industries, Inc., bi-convex blade antennas, or equivalent, and a coax switch. The appropriate antenna can be switched to the transmitter output with the coax switch. Since the selected guidance system is capable of computing the orientation of the LES relative to the Orbiter, the guidance system was used to command the coax switch and select the antenna that is pointed toward the Orbiter. All signal conditioning and data formatting for the

TABLE 4-XXI R-4D CHARACTERISTICS

(RCS Thruster for Bipropellant Concept)

THRUST, VACUUM	- 445 N (100 lb_f)
CHAMBER COOLING	- Radiation
OPERATING TIME, MAX. DEMONSTRATED	- 31,800; 4790 Seconds During Qualification and Off Limits Tests
OPERATING CYCLES, MAX. DEMONSTRATED	- 103,548; 26,530 Starts During Qualification and Off Limits Tests
SPECIFIC IMPULSE, NOMINAL, STEADY STATE	- 2834 N-sec/kg (289 lb_f-sec/lbm)
CHAMBER PRESSURE, NOMINAL	- .669 x 10^6 N/m^2 (97 psia)
NOZZLE EXPANSION RATIO	- 40:1
FUEL	- MMH
OXIDIZER	- N_2O_4
MIXTURE RATIO, NOMINAL	- 1.60
IGNITION METHOD	- Hypergolic
WEIGHT, DRY, NOMINAL	- 2.40 kg (5.3 lb_m) with 10 cm (4 in.) electrical lead cable
ENVELOPE:	
OVERALL LENGTH, MAX.	- 34.04 cm (13.4 in.)
OVERALL DIAMETER, MAX.	- 16.51 cm (6.5 in.)
TEMPERATURE:	
CHAMBER, STEADY STATE	- 1094°C (2000°F)
BELL NUT, STEADY STATE	- 927°C (1700°F)
INJECTOR HEAD, MAX. SOAKBACK	- 149°C (300°F)
PROPELLANT FEED PRESSURE	- 1.172 x 10^6 to 1.724 N/m^2 (170 to 250 Psig)



1. PREIGNITER
2. PREIGNITER CHAMBER ORIFICES
3. INNERMOST RING - EIGHT PRIMARY OXIDIZER ORIFICES
4. MIDDLE RING - EIGHT PRIMARY FUEL ORIFICES
5. OUTERMOST RING - EIGHT CHAMBER WALL FUEL COOLING ORIFICES
6. INJECTOR HEAD ASSEMBLY
7. SEAT ASSEMBLY
8. DIRECT COIL
9. OXIDIZER VALVE ASSEMBLY
10. AUTOMATIC COIL
11. FUEL ORIFICE
12. FUEL VALVE ASSEMBLY
13. ATTACH RING
14. COMBUSTOR
15. ATTACH NUT
16. BELL

FIGURE 4.11 R-4D BIPROPELLANT REACTION CONTROL THRUSTER

TM system was designed to be accomplished within the guidance system. This system could provide data transmission directly to any STDN ground station that is within line-of-sight range of the LES. The TM system could also provide data transmission directly to the Orbiter from the time of the LES deployment from the Orbiter cargo bay until sometime after the PK ignition. Subsequent to PK ignition, at a range of approximately 1.852 KM (1 nm) from the Orbiter, communication with the Orbiter could be maintained until the useful transmission range of approximately 92.6 km (50 nm) is exceeded.

The 96.2 km (50 nm) communication range could be extended if the mission requires it by the addition of a 0.9144 m (3 ft) diameter parabolic receiving antenna, a low noise (3.1 dbNF) preamplifier, and an 8 dbNF receiver in the LES ASE in the Orbiter cargo bay. This additional equipment could extend the TM communication range from 92.6 km (50 nm) to 1896.5 km (1024 nm) for the maximum altitude circular orbit case as specified by the LES payload model. This range estimate is based upon the Orbiter omni antenna and receiver requirement of an effective isotropic radiated power (EIRP) of -105 dbm to achieve a BER (Bit Error Rate) of 10^{-2}. The Inter-Range Instrumentation Group 106 document indicates that an improvement of approximately 6.5 db in S/N ratio is required to get from a 10^{-2} to 10^{-5} BER. This then equates to a required EIRP at the Orbiter of -98.5 dbm to achieve a BER of 10^{-5}. An additional gain of 4.9 db could be realized by using a 3.1 db noise figure preamplifier, like that used by TRS in the Orbiter, resulting in a required EIRP at the Orbiter of -103.4 dbm. The required antenna gain at the Orbiter is then as follows: The 1896.5 km (1024 nm) path loss (-165.2 db) plus the LES 8 watt transmitter and antenna gain (39 db), plus Orbiter EIRP (-103.4 dbm) equals antenna gain required (22.8 db). A typical 0.9144 m (3 ft) diameter parabolic antenna with Right Hand Circular Polarization (RHCP) has a gain of 23.2 db and a halfpower beam width of 11.5 degrees at S-band.

Another alternative would be to incorporate an antenna system like that shown in Reference 27. This system consists of a 13 db Helix antenna installed in the Orbiter and five (5) "Turnstile-over-Cone" antennas installed on the stage. Each Turnstile-over-Cone antenna has a gain of 5 db and a beam width of 140 degrees between the half power points.

Again in this application the appropriate antenna would be switched into the RF system by the guidance system. Based upon the "Telemetry Range Capability" graph shown in Figure 4.12, a positive margin of 4.3 db would be provided at a range of 1896.5 km (1024 nm) for a BER = 10^{-5}.

The baseline TM system will accomodate most of the missions as defined by the LES payload model. Missions such as AMPTE A and AMPTE B which are highly elliptical having apogees of 20 and 8 earth radii respectively are considered to be special cases. For these missions the payload transmitter and antenna system should be used to transmit the LES TM data.

The development testing considered necessary for the Telemetry system was primarily that associated with antenna pattern testing. This effort would be accomplished by the LES contractor and would involve the use of the LES development test (DVT) unit as the test bed. The antenna pattern measurements would be performed on the contractor's antenna range using the DVT with the omni directional antennas installed in the design locations. In the event that the antenna coverage is not adequate, i.e. less than 4π steradian, the antenna can be relocated as appropriate and the patterns reverified.

4.4.2 Electrical Power

The electrical power system provides electrical energy to all utilizing equipment on the LES. It also provides the capability for switching between the external and LES internal power sources during prelaunch checkout. Table 4-XXII shows the electrical energy required during a normal two hour flight, and Figure 4.13 shows a typical electrical load profile for a bipropellant configuration. The energy source selected to provide the required energy is an automatically activated silver-zinc battery. The silver-zinc battery is light in weight and has a long history of successful use in space applications. A battery employing lithium would be lighter in weight, but lithium systems are relatively new and are still in the development stage. An automatically activated silver-zinc battery which supplies the energy shown in Table 4-XXII for each configuration with a fifteen percent margin weighs 15.9 kg (35 lbs.) and has a volume of 7374 cc (450 cubic inches).

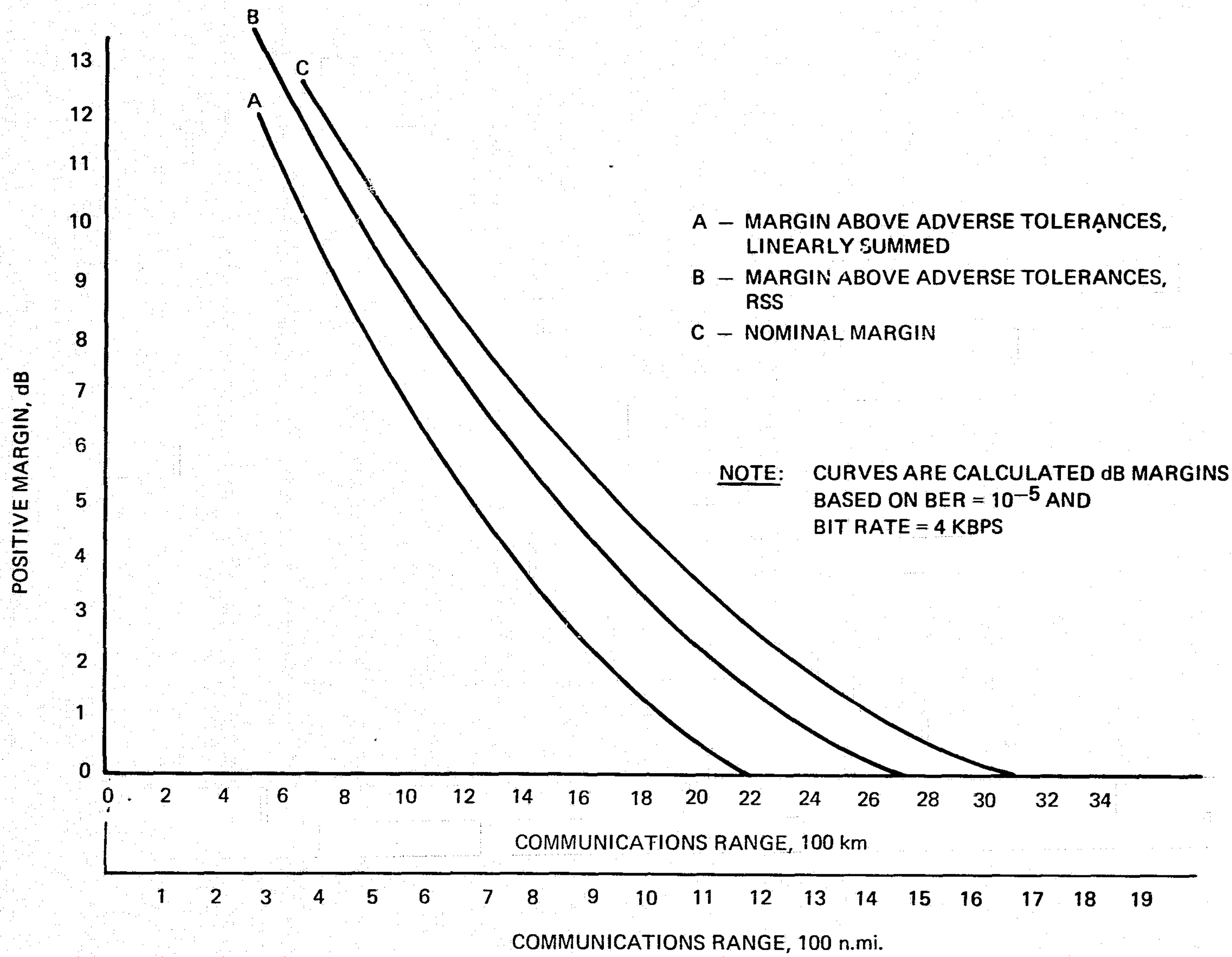


FIGURE 4.12 TELEMETRY RANGE CAPABILITY

TABLE 4-XXII ELECTRICAL ENERGY REQUIREMENTS OF LES EQUIPMENT

Equipment	STAGE CONFIGURATION											
	Bipropellant				Monopropellant				Adaptations			
	Volts	Amps	Hrs.	Watt-hrs.	Volts	Amps	Hrs.	Watt-hrs.	Volts	Amps	Hrs.	Watt-hrs.
Guidance System	28	4.3	3	240.8	28	4.3	2	240.8	28	4.3	2	240.8
Telemetry System	28	3.6	2	201.6	28	3.6	2	201.6	28	3.6	2	201.6
Propulsion Motor Valve(s)	28	2.6	.53	38.6	28	7.3	.64	130.8	28	7.3	.17	34.7
Reaction Control System Motor Valves	28	3.8	.19	20.2	-	-	-	-	28	7.3	.03	6.1
Prop. Motor Heaters	28	.7	1	19.6	28	.5	2	28.0	28	.5	2	28
RCS Motor Heaters	28	2.9	1	81.2	-	-	-	-	28	.5	2	28
Total Electrical Energy	-	-	-	602	-	-	-	601.2	-	-	-	539.2

Notes:

1. Energy requirement of ignition subsystem is negligible.

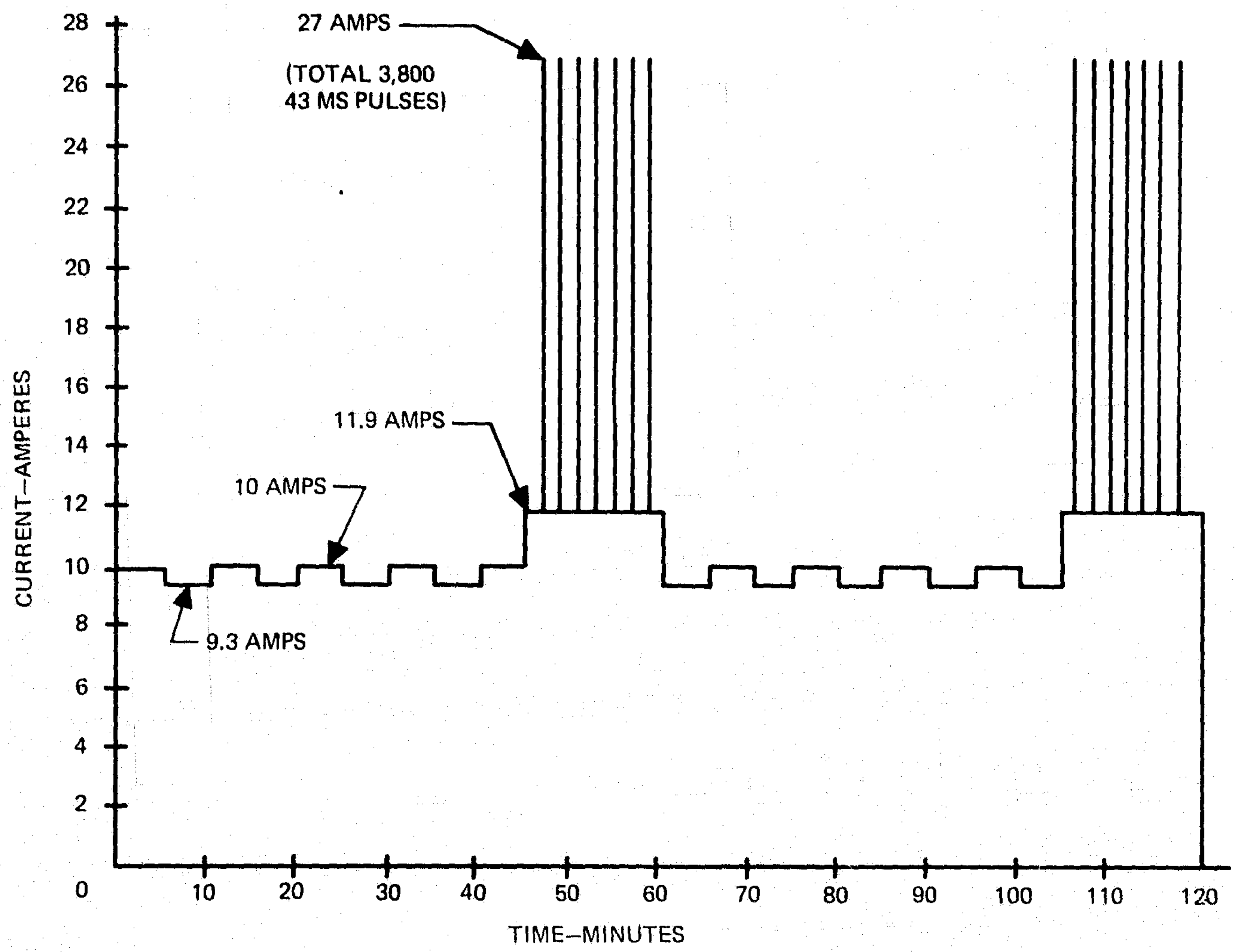


FIGURE 4.13 TYPICAL ELECTRICAL LOAD PROFILE – BIPROPELLANT CONFIGURATION

A schematic of the electrical power system is shown in Figure 4.14. Relay "K1" is the internal/external power relay. External power is supplied to the guidance and telemetry subsystems through diodes "CR1" and "CR2". These diodes isolate the open faced umbilical connector from internal power during flight. Separate lines for guidance and telemetry external power are provided so that the guidance system may be operated in the Shuttle bay without radiation from the telemetry transmitter if desired. The nqrmal sequence of operation begins with checking out the guidance and telemetry systems on external power. Approximately 10 to 15 minutes prior to deployment the vehicle battery is activated by firing the activation squib. After the necessary soak time the battery may be load checked by energizing relay "K2" to supply battery voltage through the umbilical. Relay "K2" is necessary to isolate the open faced umbilical connector from the battery voltage during flight. Shortly prior to deployment relay "K1" is switched to the internal position, and battery power is applied to the equipment in parallel with external power. The external power is turned off and the equipment operates on internal battery power only. This method of power application is necessary in order to prevent interruption of power to the guidance system during this switchover from external to internal power. If the mission should be aborted after activation of the battery, the battery should be discharged to prevent possible overheating and explosion after its safe wet stand time expires. The safe wet stand time for this application was specified to be at least eight hours. Discharge of the battery in this situation may be accomplished by energizing relay "K2" and applying an external load. Relays "K1" and "K2" and diodes "CR1" and "CR2" can be packaged in a power control unit which weighs approximately .68kg (1.5 lb.) and is approximately 443 cc (27 cubic inches) in size.

A two hour flight time is consistent with the two burn circular orbit flight requirements of the revised mission model. Highly elliptic orbits requiring a single perigee burn also fall within the two hour flight time. The model shows two payloads, less than two percent of the mission model, that have highly elliptic orbits that require a velocity

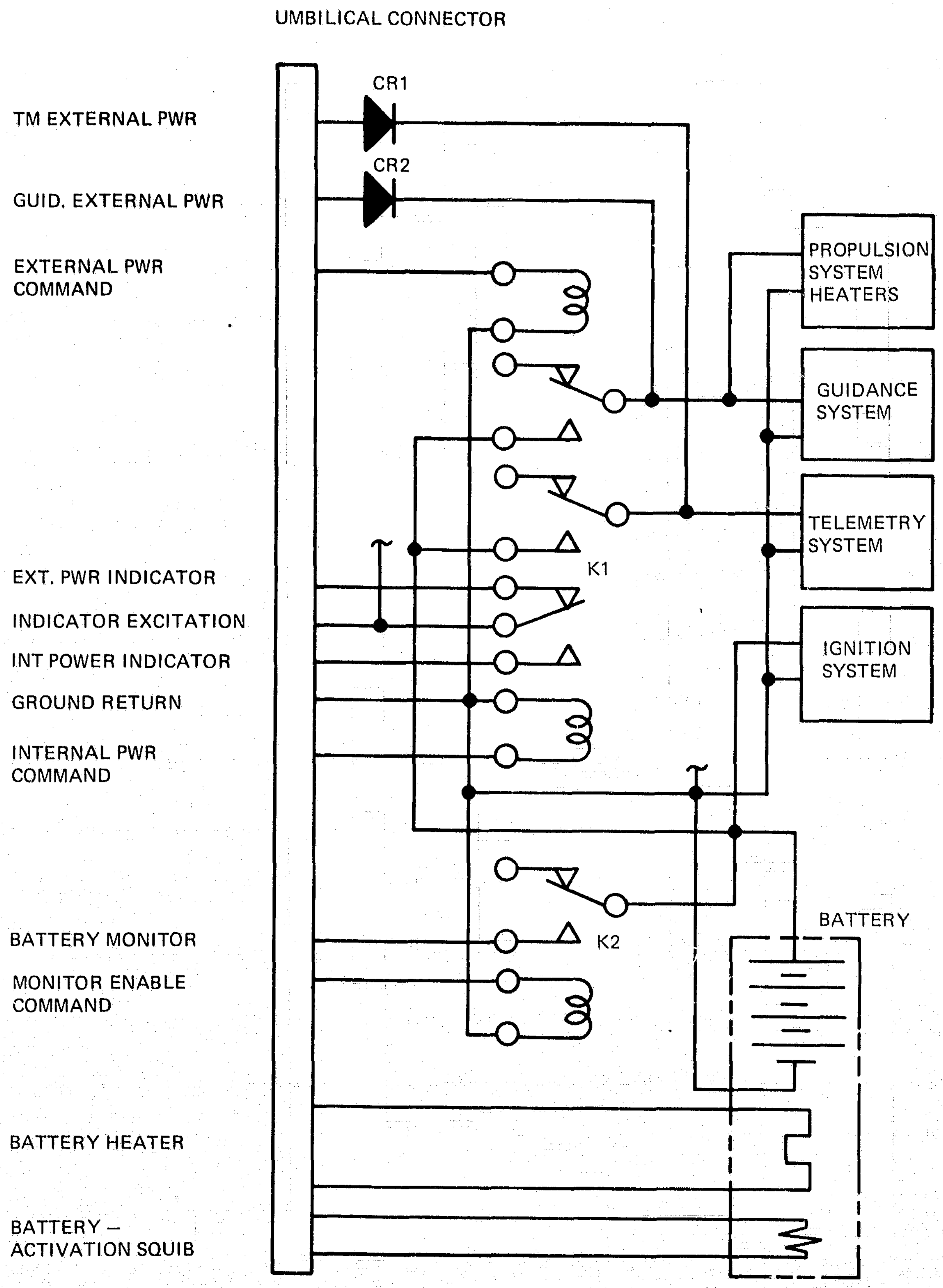


FIGURE 4.14 ELECTRICAL POWER SYSTEM SCHEMATIC

change burn at apogee. These missions would require more electrical energy for the longer flight times. This can be provided by the addition of batteries of the same size, by a larger battery, or by integration with the payload electrical energy provisions.

For cost evaluation, the Power Control Unit (PCU) was considered to be subjected to the full range of development testing by the LES contractor. The development testing includes breadboard, prototype, and qualification testing consistent with the environmental requirements of the LES.

4.4.3 Ignition System

The ignition system is configured to provide the electrical power, firing commands, and safe/arm for all pyrotechnic initiators on the LES. All pyrotechnic initiators are NASA Standard Initiators (NSI), and the functions to be initiated by these devices are shown in Table 4-XXIII. This table reflects the requirements for the twelve-tank bipropellant configuration which has the greatest number of initiators of all the configurations being considered. The guidance system provides the commands to switch

TABLE 4-XXIII - PYROTECHNIC INITIATED
FUNCTIONS 12 TANK BIPROPELLANT CONFIGURATION

Function	Number of Pyro Initiators (Total for both both redundant systems)	Times of Initiation (Minutes after Deployment)
Pressure Manifold Valve	2	30.5
Fuel Tank Outlet Valves	12	31.0
Oxidizer Tank Outlet Valves	12	31.5
Fuel Tank Pressure Inlet Valves	12	32.0
Oxidizer Tank Pressure Inlet Valves	12	32.5
Payload Separation	8	120

power to the pyrotechnic initiators for all events. The general sequence of operation for the ignition system begins after the LES is a safe distance from the Orbiter which occurs approximately thirty minutes after deployment. At this time the system is armed. Then the tank valves are fired to activate the propulsion and attitude control fuel systems. The remaining pyrotechnic initiated function is payload separation which occurs at the end of the LES mission. Redundant ignition systems are provided in order to increase reliability.

Power for firing the initiators can be provided either directly by the main battery or by capacitors which are charged at a low current by the main battery. If the initiators are fired directly by the battery a minimum of 60 amperes must be supplied to fire twelve initiators simultaneously. The normal battery load is 10 to 12 amperes and the additional 60 ampere load for firing pyrotechnic initiators could momentarily reduce the line voltage below acceptable limits unless the battery size is substantially increased to handle this large load. On the other hand,

28 volts, and the weight of the capacitors required to fire twelve initiators simultaneously is 0.96 kg (2.12 lbs.). These capacitors can be recharged in ten seconds for firing the next event while drawing a maximum current of 0.1 ampere from the main battery. The capacitors discharge method of firing pyrotechnic initiators was selected for this application, because it isolates the high current surges associated with firing the initiators from the main bus.

Transistor switches turned on by signals from the guidance system are used for switching energy from the capacitor banks to the initiators. Relays are not used in this application, because contact bounce could adversely affect the proper flow of energy to the initiators. The firing signal from the guidance system is to be approximately one second in duration so that the circuit is turned off to permit recharging the capacitors in the event of a post firing short circuit in an initiator.

Safety is a prime consideration in the design of the ignition system. Safe/arm relays are provided which open the firing circuit and apply a short circuit to the initiator bridgewires prior to arming. It is a requirement that three failures must occur in a pyrotechnic system in order to initiate inadvertently any event which is hazardous to the Orbiter. Even though the basic three failure mode system is identified in this study for LES, final approval for use of this system and other hazards affecting the safety of the Orbiter crew and Orbiter must come from JSC Safety department prior to implementation and use. For this study, the three failure modes for the pyrotechnic circuitry is provided by three independent commands from the guidance system before an initiator can be fired: the first command energizes magnetic latching relays which arm the ignition capacitor busses and permits the firing capacitors to charge; the second command arms the magnetic latching safe/arm relays which are connected directly to the initiators; the third command is the firing command which turns on the firing transistors. Figure 4.15 is a simplified schematic of the ignition system showing these safety provisions. Monitor circuits not shown on the schematic are provided to indicate that all safe/arm relays are in the safe position prior to deployment.

The firing capacitors, charging resistors, switching transistors, and safe/arm relays are packaged in an Ignition Control Unit (ICU). Estimated size and weight of the ICU for each LES configuration is shown

TABLE 4-XXIV - IGNITION CONTROL UNIT WEIGHT AND SIZE

LES Configuration	ICU Weight kg (lb)	ICU Volume cc (cu. in.)
2 Tank Monopropellant	2.12 (4.7)	1639 (100)
4 Tank Monopropellant or Bipropellant	2.54 (5.6)	1999 (122)
8 Tank Monopropellant or Bipropellant	3.18 (7.0)	2786 (170)
12 Tank Bipropellant	4.3 (9.5)	3835 (234)
Adaptation - Solid First Stage plus 4 Tank Bipropellant Second Stage	2.86 (6.3)	2425 (148)

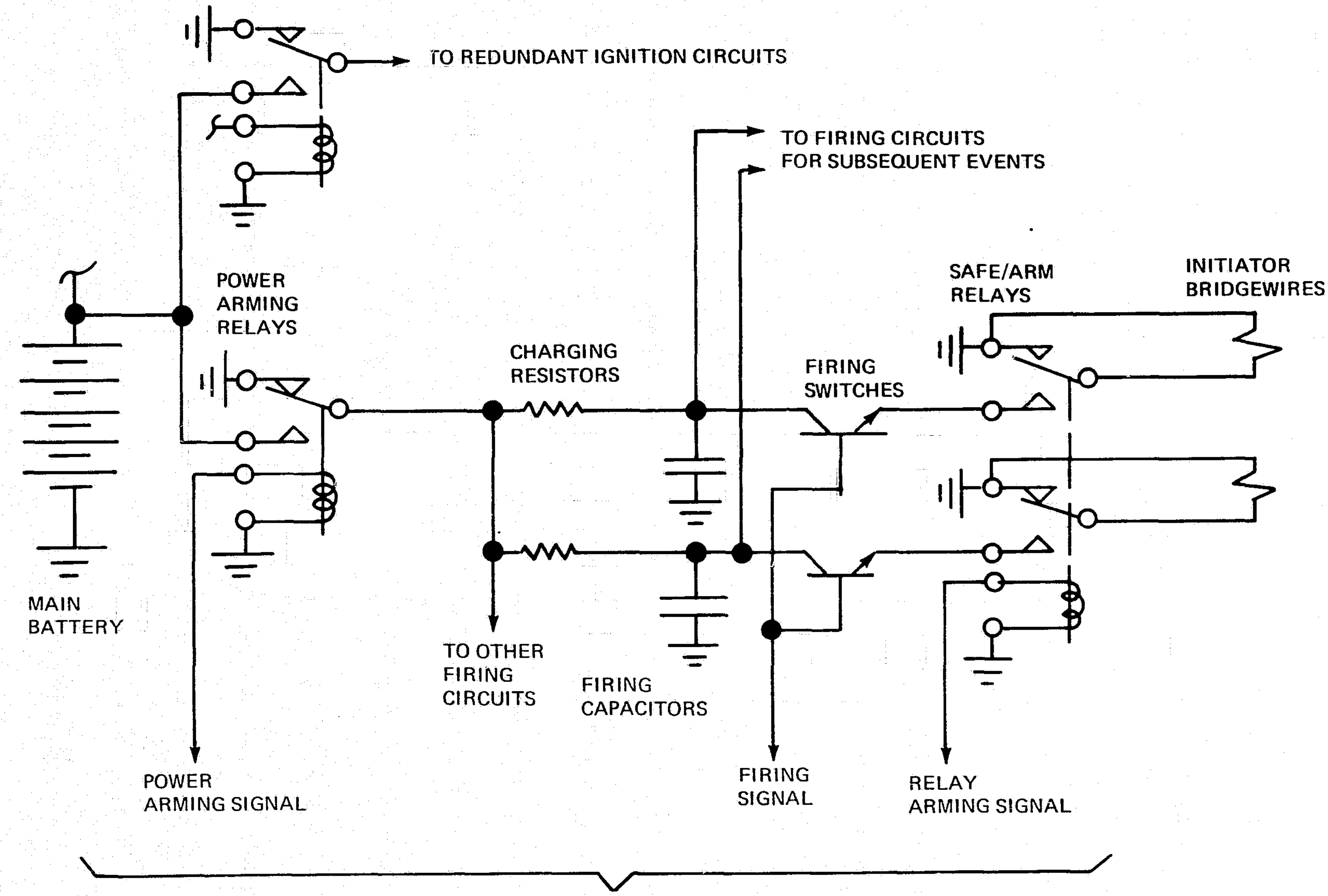


FIGURE 4.15 SCHEMATIC — TYPICAL PYROTECHNIC IGNITION CIRCUIT

in Table 4-XXIV. If several modules are selected to meet mission requirements, one ICU configuration will be designed to meet the worst case requirements and used without modification for other modules.

For cost evaluation the Ignition Control Unit (ICU) was considered to be subjected to full development testing by the LES contractor. The development testing includes breadboard, prototype, and qualification testing consistent with the environmental requirements of LES.

4.4.4 Thermal

An analysis of the bipropellant configuration is presented as typical of the thermal considerations for an expendable low energy stage. Worse case thermal environments were examined to determine the sensitivity of the various components to hot and cold soak conditions. Thermal protection was selected based on the worst case approach. Much of the selected protection could be eliminated under operational conditions that permit the Orbiter to rotate slowly while in orbit and the stage to rotate slowly while in the coast condition. Future studies should examine this approach in greater detail.

The Orbiter is presently being designed to stay on station in orbit for seven days during which time the LES, stowed in the payload bay, will be exposed to potentially severe thermal environments. No barbecue continuous hot and cold conditions were analyzed. Assuming the LES/payload combination is the last payload discharged from the Shuttle, the LES would be in the cargo bay approximately 160 hours. This time was used to determine final soak temperatures in hot and cold environments. LES electronics will be activated for checkout for approximately 30 minutes before discharge. The R-40 main thruster was assumed to have two burns, the first being 565 seconds and the second being 620 seconds. The R-4D thrusters used for vehicle attitude control was assumed to fire intermittently on a 10% duty cycle.

4.4.4.1 Propellant Tanks Thermal Analysis - Temperature excursions of propellant/tank system during the 160 hours stowage require thermal protection and a multilayer insulation (MLI) blanket was selected to cover the entire vehicle with the exception of the R-4D thrusters. Figure 4.16 shows the temperature change of a fully loaded propellant system from an initial temperature of 21.1°C (70°F) if all the heat passing through the MLI blanket affects only the propellant system temperature. The analysis is conservative since the additional thermal capacitance of vehicle structure will tend to reduce the magnitude of the temperature excursion from the initial temperature. All the electronics are dormant during the stowage time with the possible exception of the battery which has an internal heater to maintain its temperature at acceptable levels.

In the event the mission is aborted and the LES/payload remains in the payload bay during re-entry, Figure 4.17 shows the temperature rise of the full propellant tanks for an initial temperature of 21.1°C (70°F).

4.4.4.2 Electronics Thermal Analysis - Only two electronic packages generate sufficient internal heat to create concern; these are the guidance and control unit (GCU) and the telemetry (TM) transmitter. The GCU temperature (operating constantly) will rise at 36°F/hr and therefore the worst case thermally can be handled by good thermal connections to the LES structure. The TM has very low thermal mass and a high heat generation which requires the TM be mounted outside the MLI blanket on a 22 inch square radiator to reject the excess heat. The 22 by 22 inch radiator thermal characteristics shown in Figure 4.18 will cool the TM below acceptable temperature limits even in full sun; therefore a TM heater is required when the TM is off and the radiator can see deep space.

4.4.4.3 Thrusters Thermal Analysis - The LES vehicle concept analyzed proposes a Marquardt R-40 for the main thruster and four Marquardt R-4D's for attitude control. All five thrusters point essentially aft although the R-4D's are canted 10°. Plume backwash in the form of adiabatic heating rates for both types of thrusters was calculated and is presented in Figure 4.19. The combined heating rates of the R-40 and the R-4D worst duty cycle require vehicle protection from the heating and possible plume

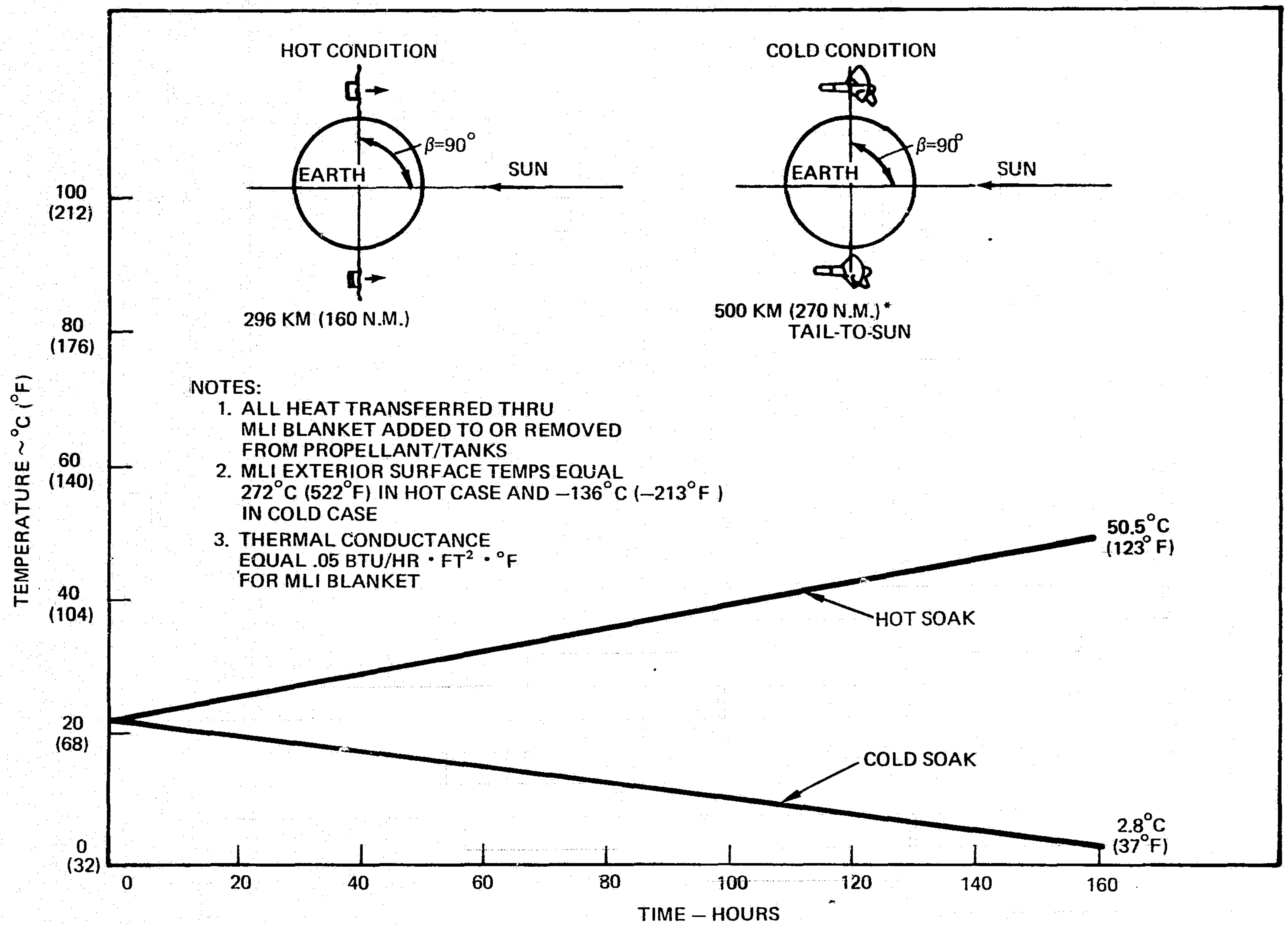


FIGURE 4.16 PROPELLANT/TANK TEMPERATURE (LES COVERED WITH MLI BLANKET)

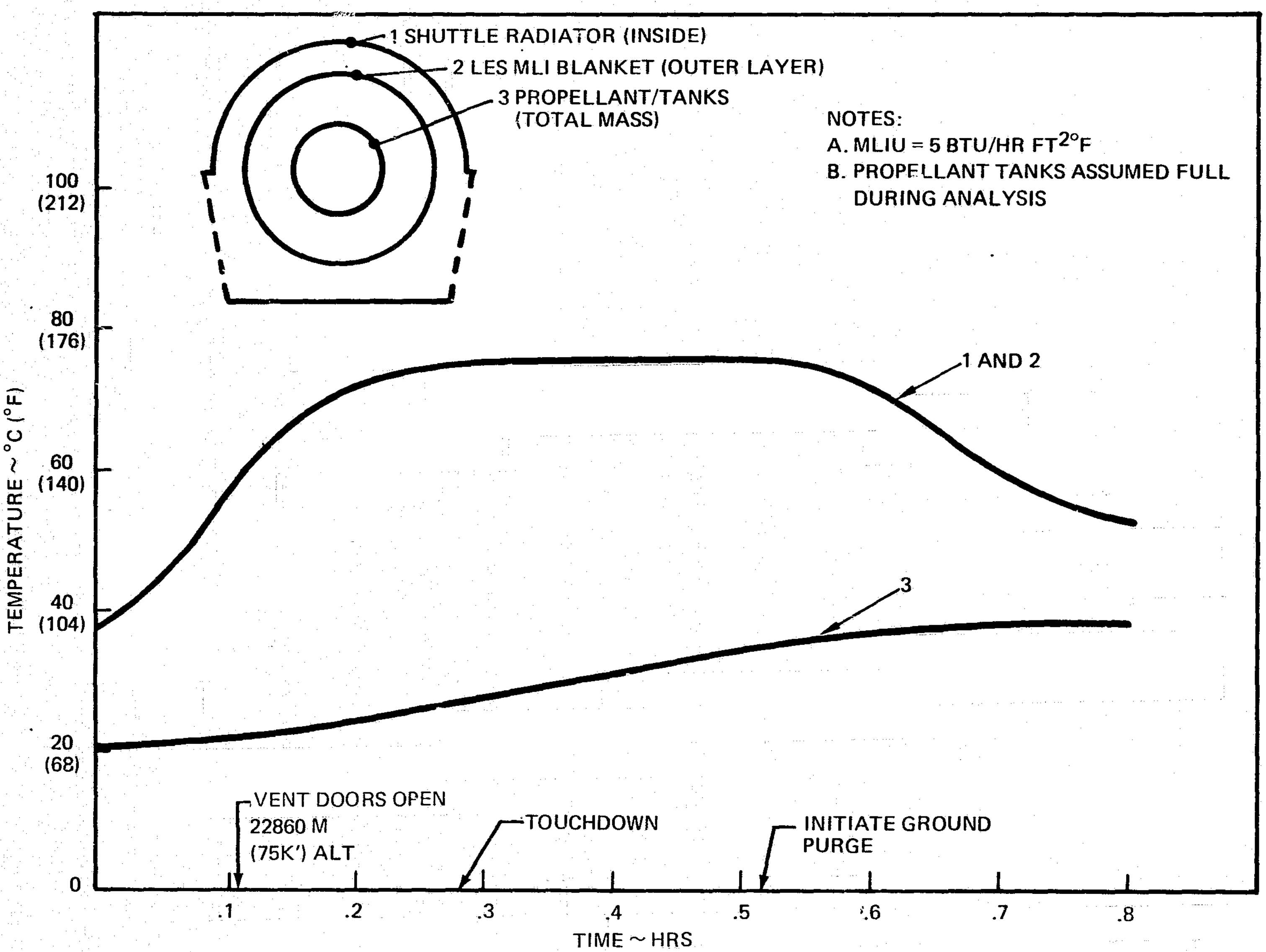


FIGURE 4.17 PROPELLANT SYSTEM TEMPERATURE PROFILE (LES IN CARGO BAY DURING REENTRY)

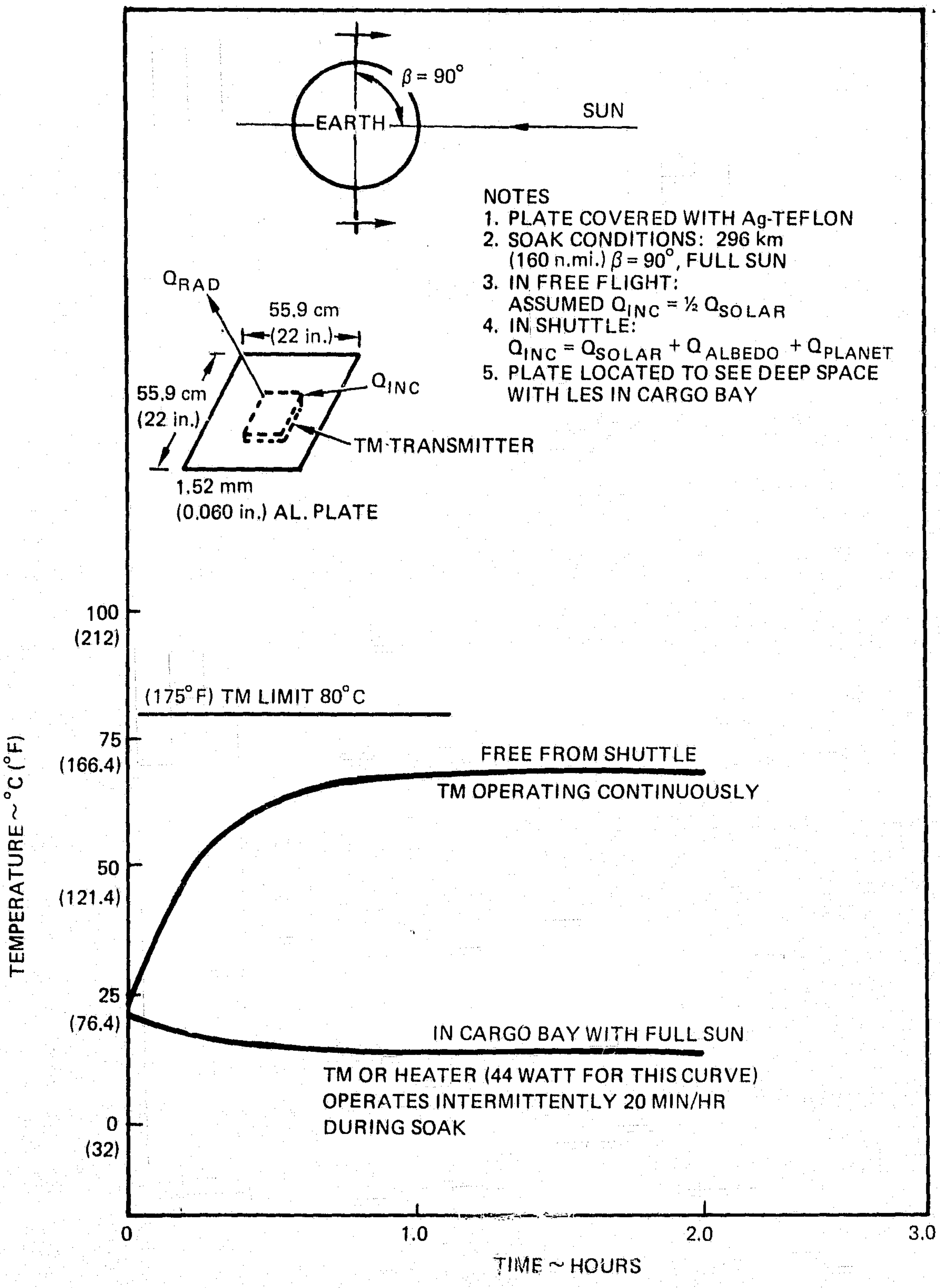


FIGURE 4.18 TELEMETRY TRANSMITTER TEMPERATURE PROFILE

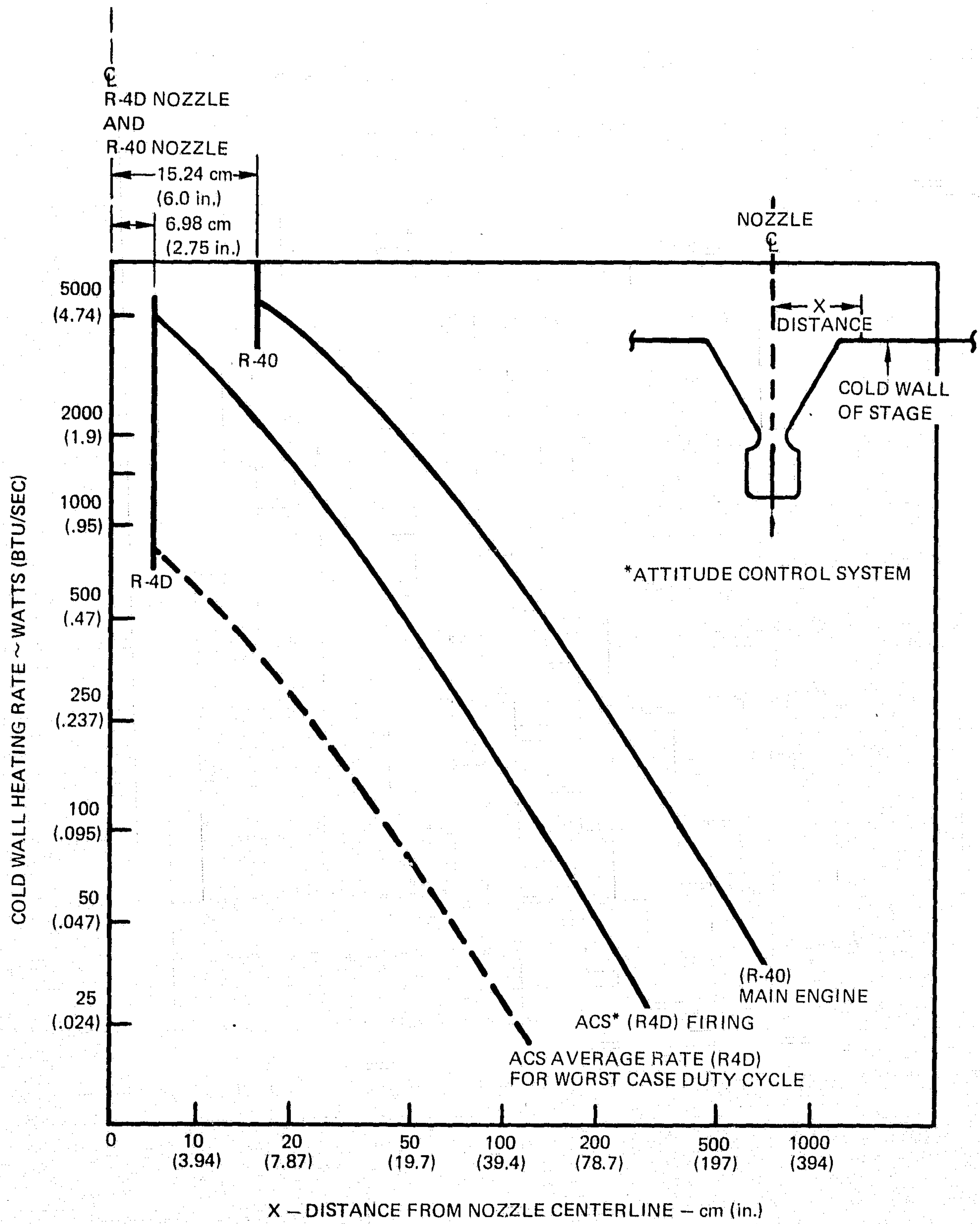


FIGURE 4.19 PLUME HEATING

erosion. A 762 mm (30 in.) radius, 1.02 mm (.04 in.) thick titanium shield was selected to cover the vehicle around the R-40 nozzle exit. The interior side of the shield is insulated to reduce thermal soakback as is the exterior of the R-40 bell nozzle which is located within the vehicle. Figure 4.20 shows the exterior surface shield temperatures along a line connecting the thrusters after steady state temperatures are reached. These high temperature surfaces are insulated with a type of glass foam insulation of the proper thickness to prevent overheating of the internal components. Thermal soakback through the thruster structural mountings was calculated and causes an insignificant vehicle temperature rise. The R-4D thrusters are designed to be radiation cooled and due to their location on the vehicle could provide considerable heat input to the vehicle. The MLI blanket will insulate the vehicle components and structure but must be overlayed by a high temperature MLI like Kapton in the area of the R-4D's. If a non-firing R-4D thruster is exposed to deep space (payload bay stowage) for short periods of time, the thruster temperature will drop below desirable values. Figure 4.21 shows the results of installing a heater to keep the thruster above propellant freezing temperatures.

4.4.4.4 Plume Contamination/Separation Distance - No fixed contamination levels have been established for the Orbiter and the requirement for each mission must be determined on the basis of the specific experiments carried and their operational status at the time of LES engine burn. Other studies have attempted to define separation distance based on deposition of solid motor particle flux on optical surfaces. Contamination flux densities between 10^{-6} and 10^{-5} gram/cm^2 were used in these studies. The bipropellant LES engines emit no solid particle flux; therefore the contamination from the gaseous portion of the plume is of concern only and then only for those experiments which employ cryogenic surfaces and are operating during the engine burn. Engine orientation at the time of firing has a significant impact on initial separation distances. Figure 4.22 presents the initial separation distances for an engine burn perpendicular to the Orbiter and Figure 4.23 shows the shorter distances permissible for an engine burn parallel to the Orbiter.

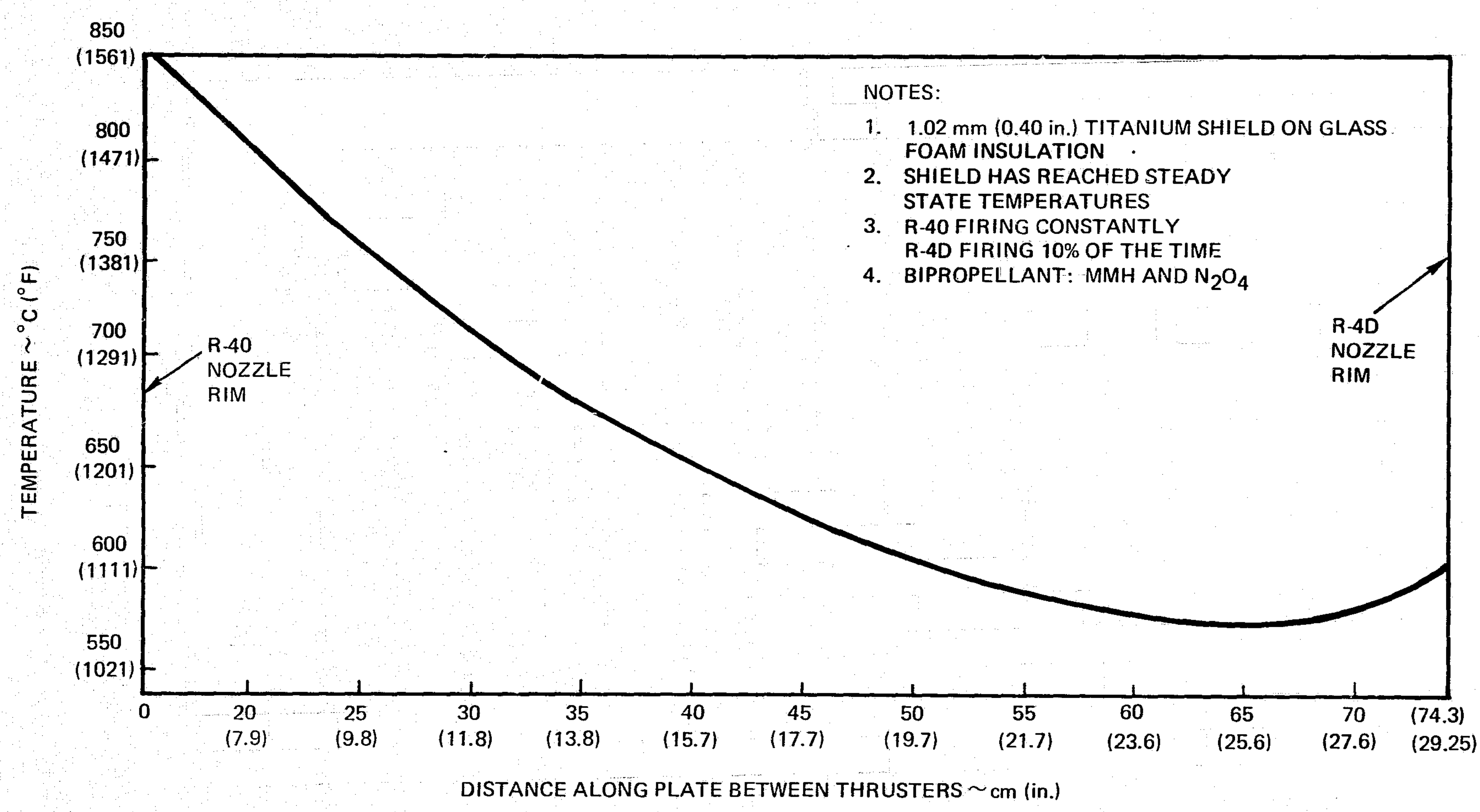


FIGURE 4.20 TEMPERATURE PROFILE OF SHIELD BETWEEN R-40 AND R-4D AFTER 0.10 HOUR

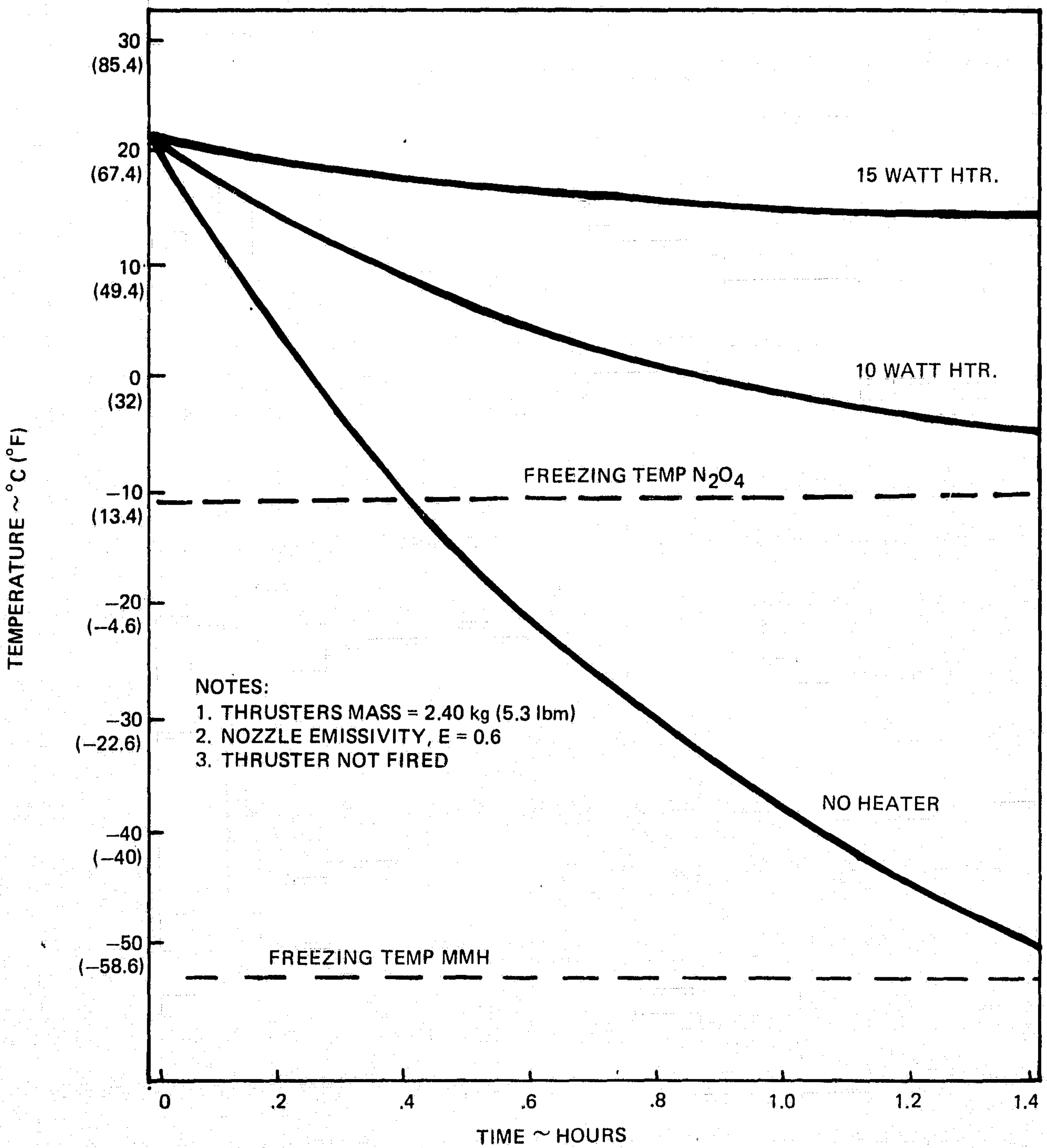


FIGURE 4.21 R-4D ENVIRONMENT COLD SOAK

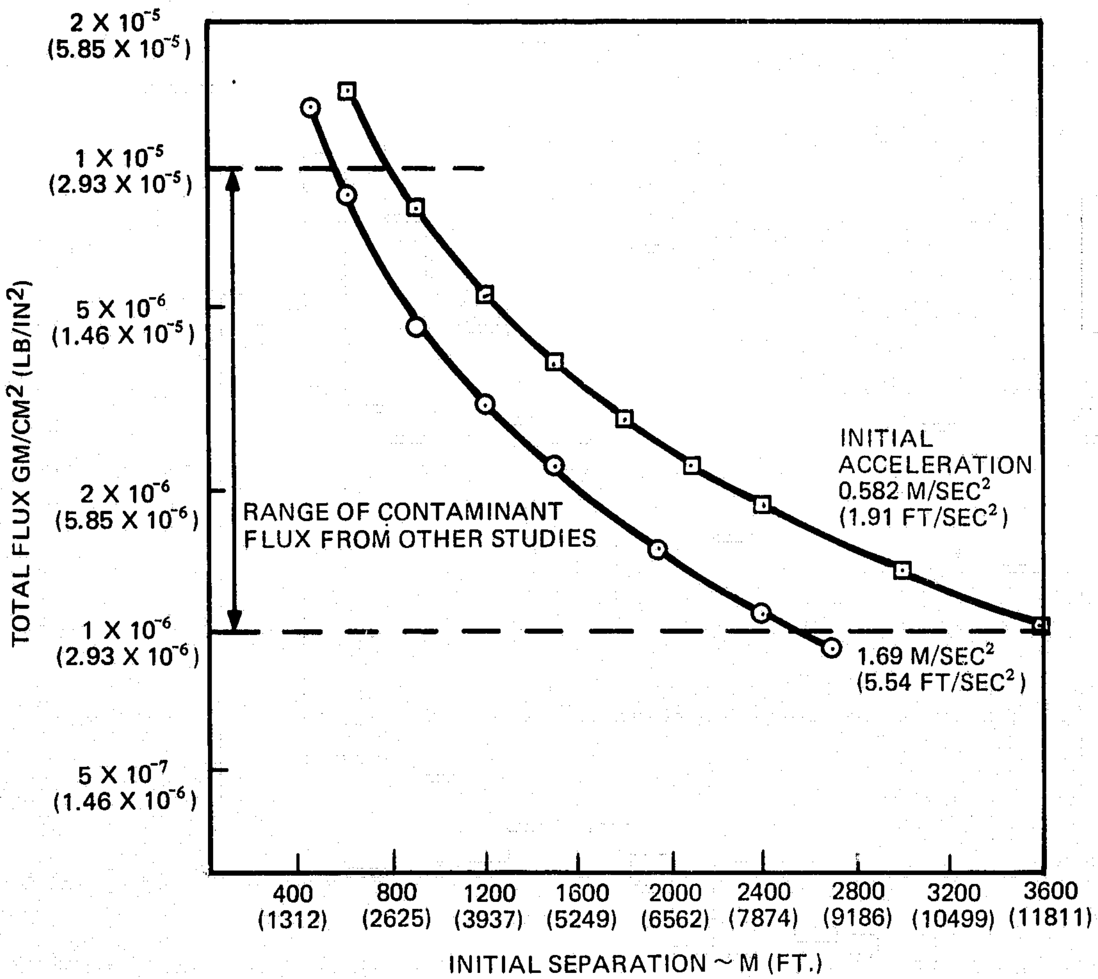


FIGURE 4.22. PLUME CONTAMINATION (PERPENDICULAR LAUNCH)

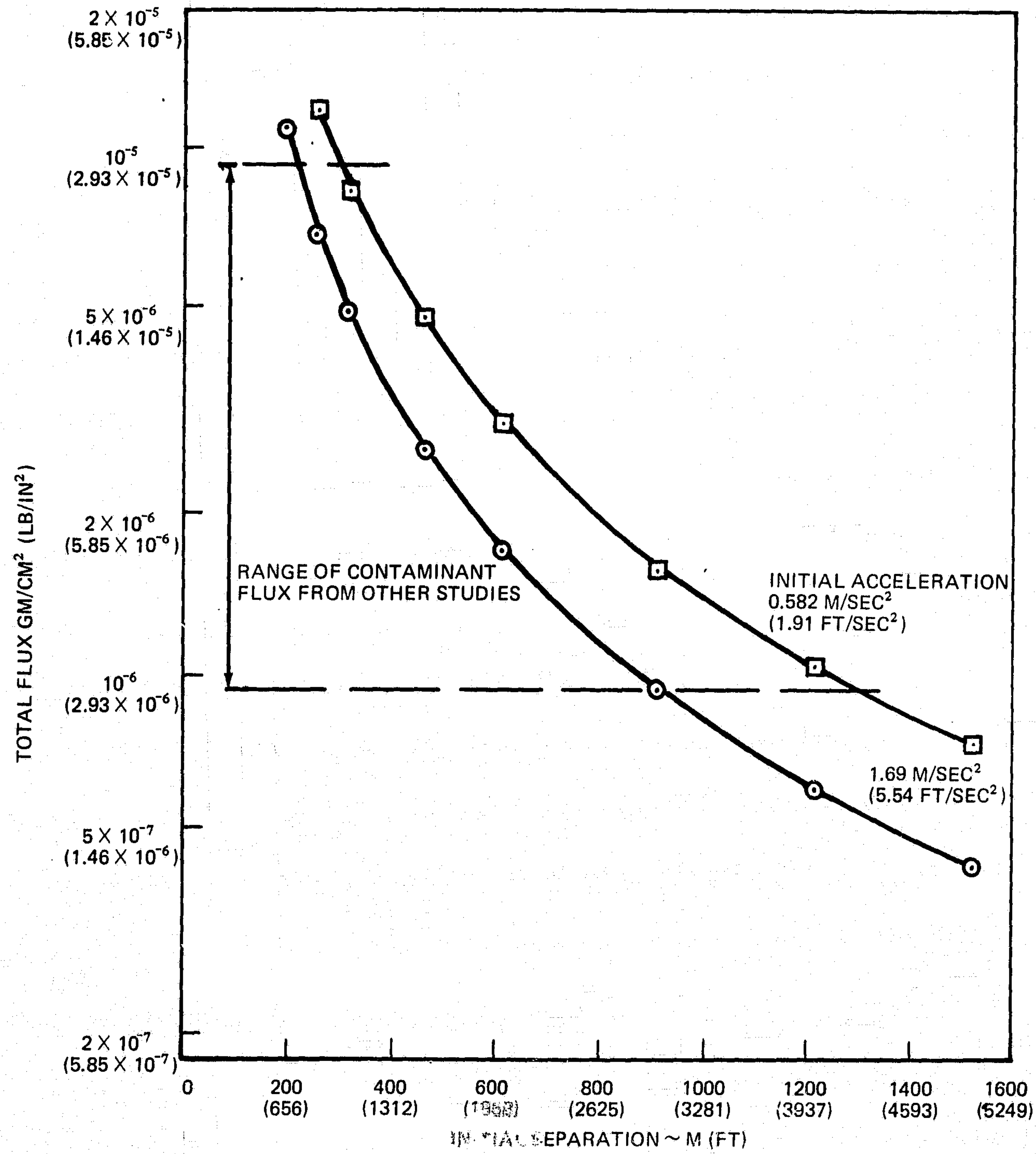


FIGURE 4.23 PLUME CONTAMINATION (PARALLEL LAUNCH)

4.5 CONCEPTUAL DESIGNS

The conceptual design effort was based on the results from the Task 2 screening and on Task 3 systems trade studies. A summary of these results are as follows:

- The most cost effective new LES concepts are the modular liquid bipropellant and monopropellant approaches.
- The LES structural configuration design should be a pure truss as determined by the trade studies described in paragraphs 4.2.1, 4.2.2 and 4.2.3.
- The guidance and control system selected should be a 3-axis type, as described in paragraph 4.2.4.
- Propulsion subsystems design should incorporate the features described in paragraph 4.3.
- The other LES subsystems design should incorporate the features in paragraph 4.4.

An assessment of these data, the payload sizes and characteristics of paragraph 4.1.4 and an initial investigation of the existing airborne support equipment defined additional design requirements. These design requirements are applicable to both the bipropellant and monopropellant approaches and are summarized as follows:

- Standardize the payload-to-stage mechanical interface to utilize the existing, proven and accepted separation plane V-band ring type of mounting. Further, three sizes were selected to provide flexibility in matching payload sizes .914 m (3.0 ft.), 1.219 m (4.0 ft.) and 1.524 m (5.0 ft.) basic diameters.
- Standardize LES mounting trunnions to use the standard MMS size trunnions and thus the standard MMS/FSS latching mechanisms being developed by an existing NASA/GSFC program.

- A maximum design LES space envelope of 4.0 m (13.12 ft.) diameter by 1.0 m (3.28 ft.) long.
- Establish the payload/stage separation by means of mechanical springs. The design concept employs four springs spaced 90° apart and located just inside the separation plane V-band ring with the size of the springs to be selected so that a .610 mps (2.0 fps) nominal design delta velocity could be achieved. This provides a slant range separation distance of 5370 m (17,618 ft.) in 45 minutes. (Reference 54).
- Establish the general location for the payload/LES electrical umbilical connector, if one is required, at the 0° aximuth angle (top) of the stage with the stage mounted horizontally in the Orbiter cargo bay and located just inside the V-band separation ring.
- Establish the starboard side of the installed LES for the Y-axis location of the LES/ASE cradle electrical umbilical connectors. Additionally, these connectors would be located between Orbiter station Z_O 10160 (400) and Z_O 10515.6 (414) in order to be accessible following final installation in the Orbiter cargo bay.

4.5.1 Bipropellant Configurations

The baseline LES used during the stage conceptual design effort was the horizontally mounted eight tank bipropellant flat X configuration using a modular construction approach to permit removal of the four outer tanks, their plumbing and electrical connections, and their support structure to produce a four tank stage suitable for either horizontal or vertical mounting in the Orbiter cargo bay. Figure 4.24 shows the eight tank baseline LES. Every effort was made in conceptual design to keep the stage length (that dimension measured in a direction parallel to the Orbiter cargo bay X-axis) to a minimum. The structural frame width

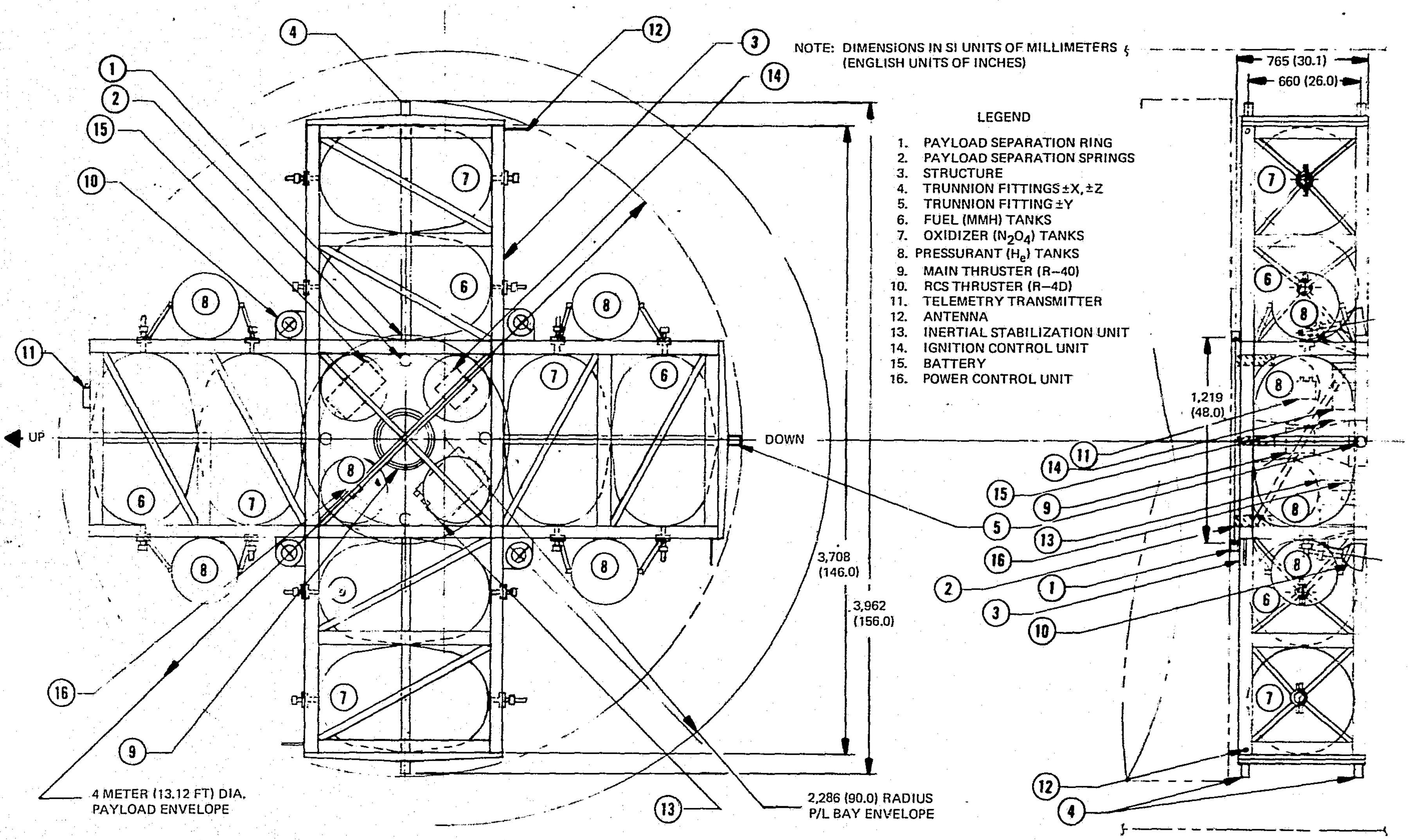


FIGURE 4.24 EIGHT TANK MODULAR BIPROPELLANT LES

of 737 mm (29.00 in.) is only slightly larger than the tank girth diameter of 635 mm (25.00 in.) in order to allow the diagonal braces to pass over the tank at its maximum girth. The modular features of the structural frame assembly are depicted in Figure 4.3, which show the capability to remove the four outer tank mounting structures and to reposition the end plates with their mounting trunnions to the vacated structural interface to provide a four-tank stage. Access to the avionics equipment mounted near the aft end of the stage is obtained by unfastening and laying back the thermal blanket and then reaching into the open structure around the main thruster.

The four-tank modular bipropellant LES is shown by Figure 4.25. The overall length from V-band separation plane to aft end is the same as for the eight-tank stage. Equipment items relocated for the four-tank stage are the telemetry transmitter and its antennae.

Figure 4.26 depicts the arrangement of the modular components to produce a four-tank vertically mounted LES. This configuration is obtained by removing the end plates from the four-tank version and repositioning the upper and lower oxidizer tanks into the in-line arrangement as shown. An auxiliary structural frame is used to reposition the ± X and ± Z trunnions in order to open the load reaction points and give better support to a fully cantilevered payload. Another vertical arrangement studied during the Task 3 conceptual design is shown in Figure 4.27. This arrangement shows a twelve tank bipropellant stage configured for vertical mounting in the Orbiter bay. This stage employs a structural assembly similar to the previously discussed stages; however, it is unique to this vertical arrangement. The driving factor for developing a structural concept was to construct a low slung structural bridge from Orbiter LH longeron to RH longeron and of minimum cargo bay length. Each of the two wings would support six tanks and the capability of off loading the tanks in symmetrical pairs to produce an eight-tank or a four-tank version of the stage. In this arrangement the design objective of minimum length was achieved, the length of 1575 mm (62.0 in.) being 7.5% shorter than the 1702 mm (67.0 in.) length of the four tank vertical modular configuration shown in Figure 4.26. The cost effectiveness of this arrangement was

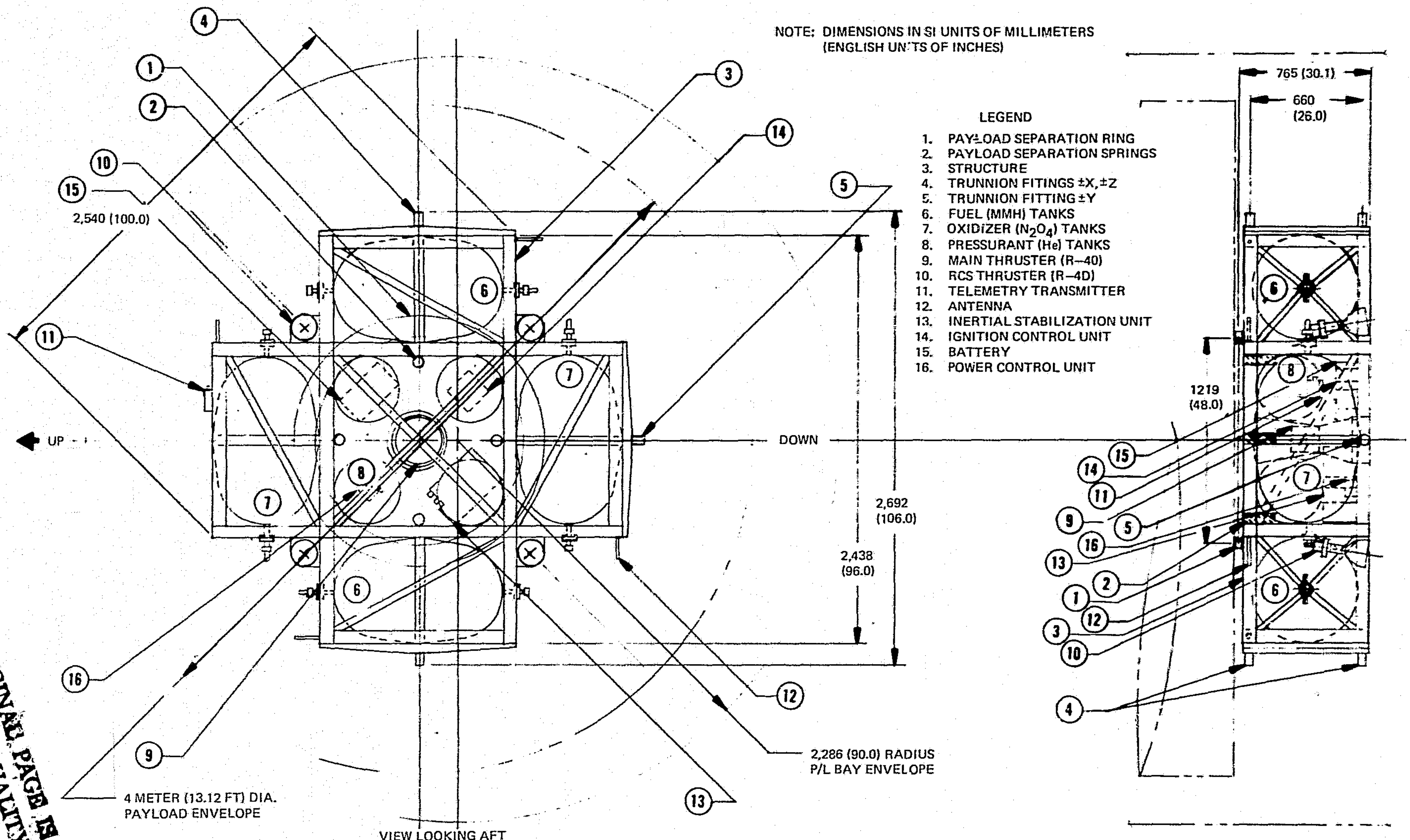


FIGURE 4.25 FOUR TANK MODULAR BIPROPELLANT LES

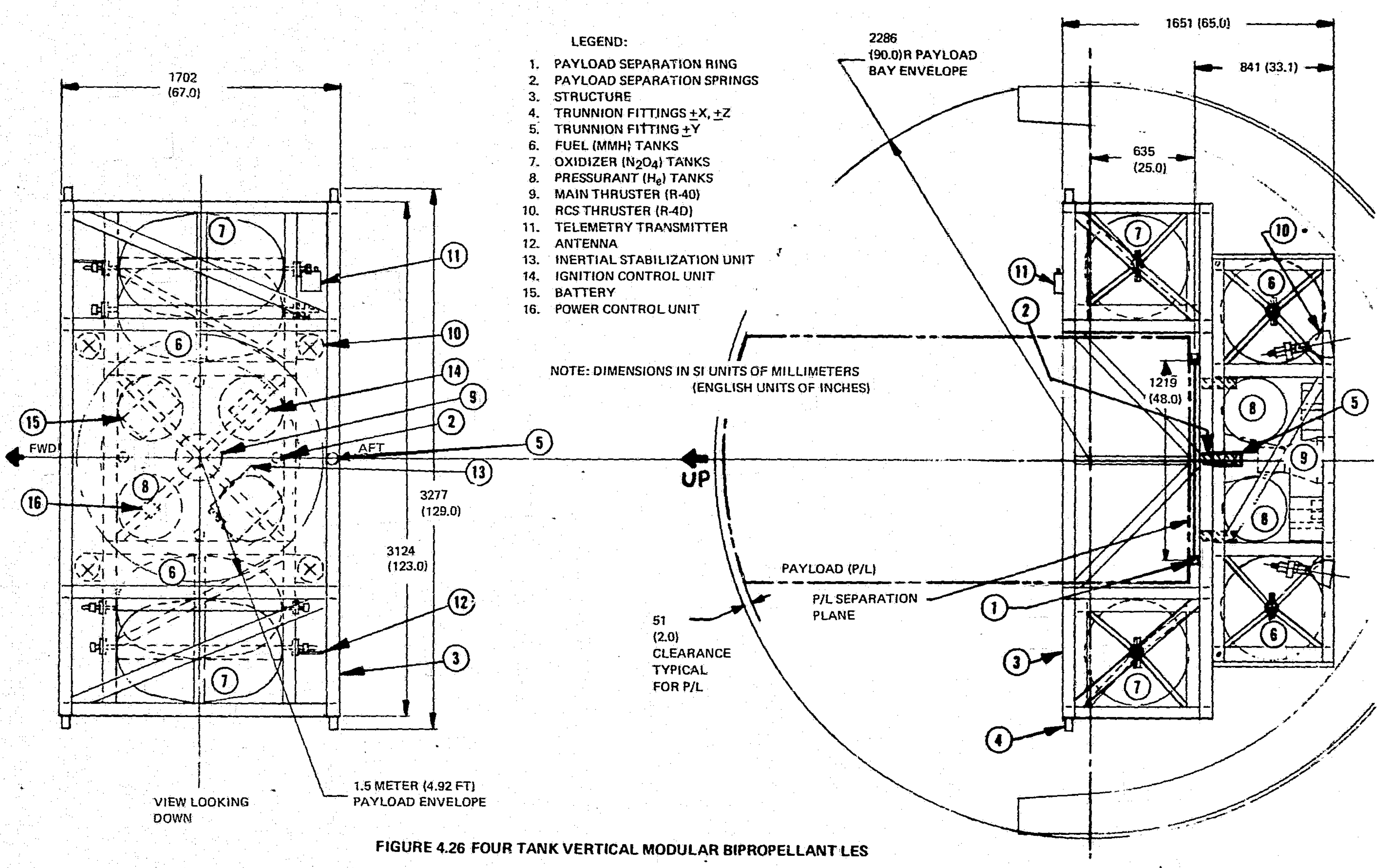


FIGURE 4.26 FOUR TANK VERTICAL MODULAR BIPROPELLANT LES

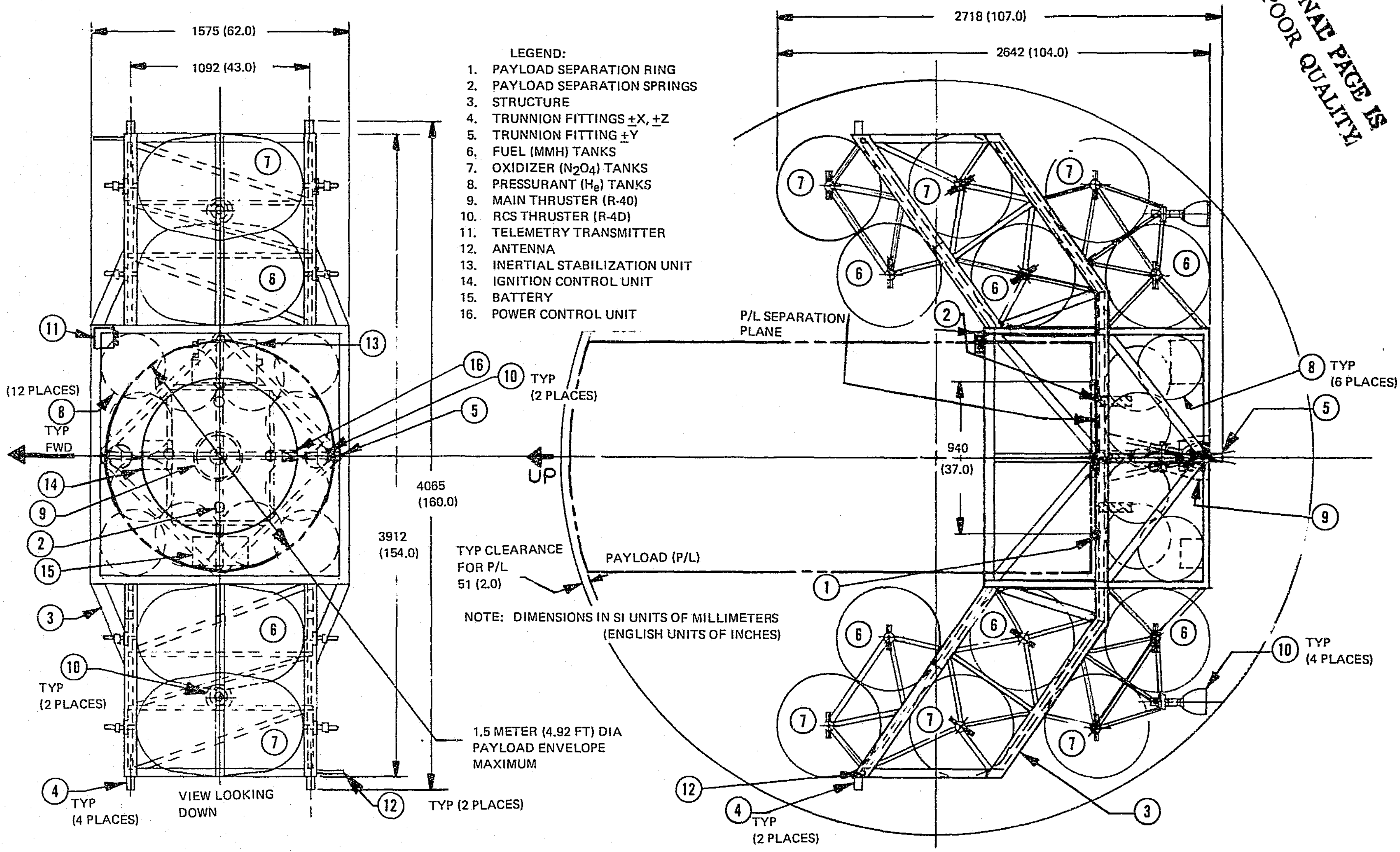


FIGURE 4.27 TWELVE TANK VERTICAL MODULAR BIPROPELLANT LES

C-2

tested during the evaluation in Task 6. Although the stage is of a shorter length, thus effecting a reduced user's charge based on cargo bay length, the savings was not great enough to offset the DDT&E costs of developing a second basic structural assembly. This arrangement cannot be used for horizontal mounting in a cost effective manner and the flat-X four and eight tank versions are still needed to handle the larger payloads.

Several of the payloads in the LES mission model are small in size and mass but require a high velocity increment. These payloads are prime candidates for launch by means of an "adaptation" to an existing stage. In this concept an existing upper stage is used to produce the perigee kick of the transfer orbit and then a smaller stage provides the apogee kick. Figure 4.28 shows an "adaptation" of the four tank vertical modular LES onto the existing SSUS-D and mounted in the SSUS-D vertical cradle. This same "adaptation" may be mounted horizontally in the Orbiter cargo bay on the SSUS-A horizontal cradle assembly. An adapter section is added between the SSUS-A spin table and the SSUS-D spin table separation ring.

A larger and more powerful "adaptation" stage is achieved by substitution of the SSUS-A for the SSUS-D and again using the SSUS-A horizontal cradle assembly. This configuration is shown by Figure 4.29. Although not shown, it is possible to use the four-tank modular horizontal stage (Figure 4.25) on top of SSUS-A or SSUS-D as alternative arrangements to those shown by Figures 4.28 and 4.29. These alternatives were also evaluated in Task 6 because they offered a small weight savings and provided improved access to the subsystems.

4.5.2 Monopropellant Configurations

Three monopropellant single stages and two "adaptation" two stage configurations were developed during Task 3. Again, as in the bipropellant systems, an eight-tank version arranged in a flat-X constituted the baseline configuration, see Figure 4.30. The longer stage length (as compared to the bipropellant eight-tank configuration of Figure 4.24) and the larger diameter are due to use of larger propellant tanks. The increased tank size required the relocation of the pressurization tanks

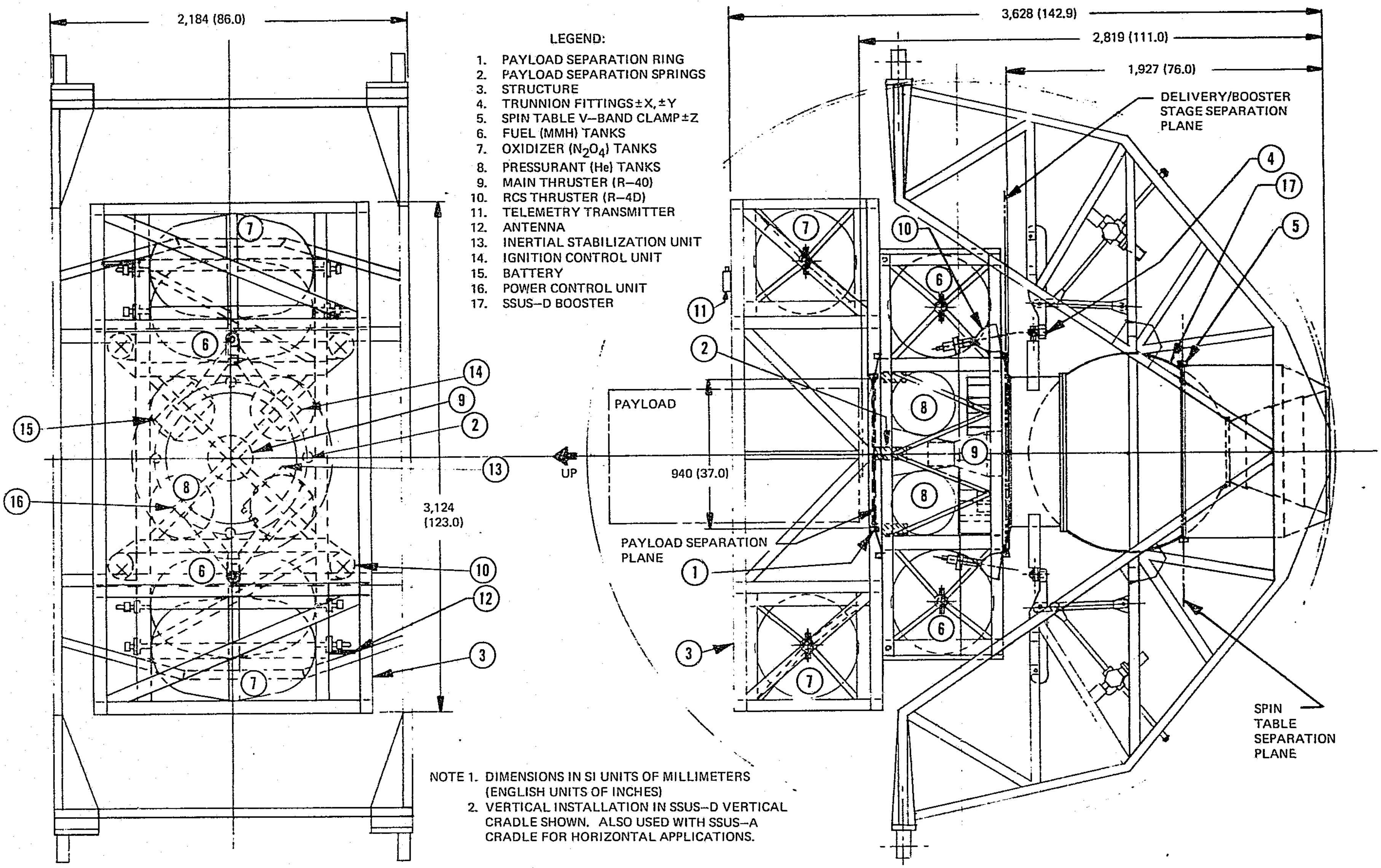
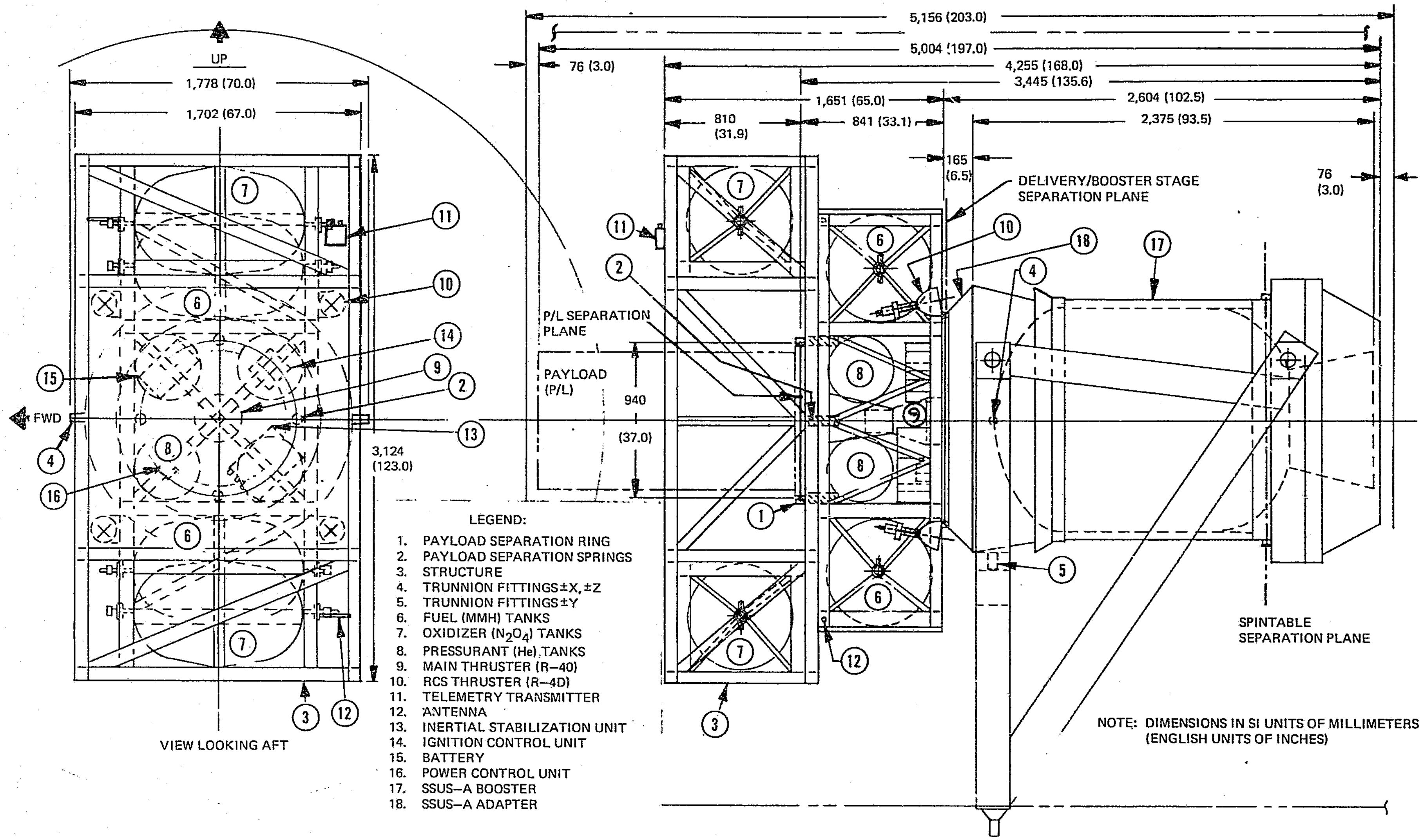


FIGURE 4.28 FOUR TANK MODULAR BIPROPELLANT/SSUS–D (VERTICAL OR HORIZONTAL)



LEGEND:

1. PAYLOAD SEPARATION RING
2. PAYLOAD SEPARATION SPRINGS
3. STRUCTURE
4. TRUNNION FITTINGS±X, ±Z
5. TRUNNION FITTINGS±Y
6. FUEL (MMH) TANKS
7. OXIDIZER (N_2O_4) TANKS
8. PRESSURANT (He) TANKS
9. MAIN THRUSTER (R–40)
10. RCS THRUSTER (R–4D)
11. TELEMETRY TRANSMITTER
12. ANTENNA
13. INERTIAL STABILIZATION UNIT
14. IGNITION CONTROL UNIT
15. BATTERY
16. POWER CONTROL UNIT
17. SSUS–A BOOSTER
18. SSUS–A ADAPTER

FIGURE 4.29 FOUR TANK MODULAR BIPROPELLANT/SSUS–A (HORIZONTAL)

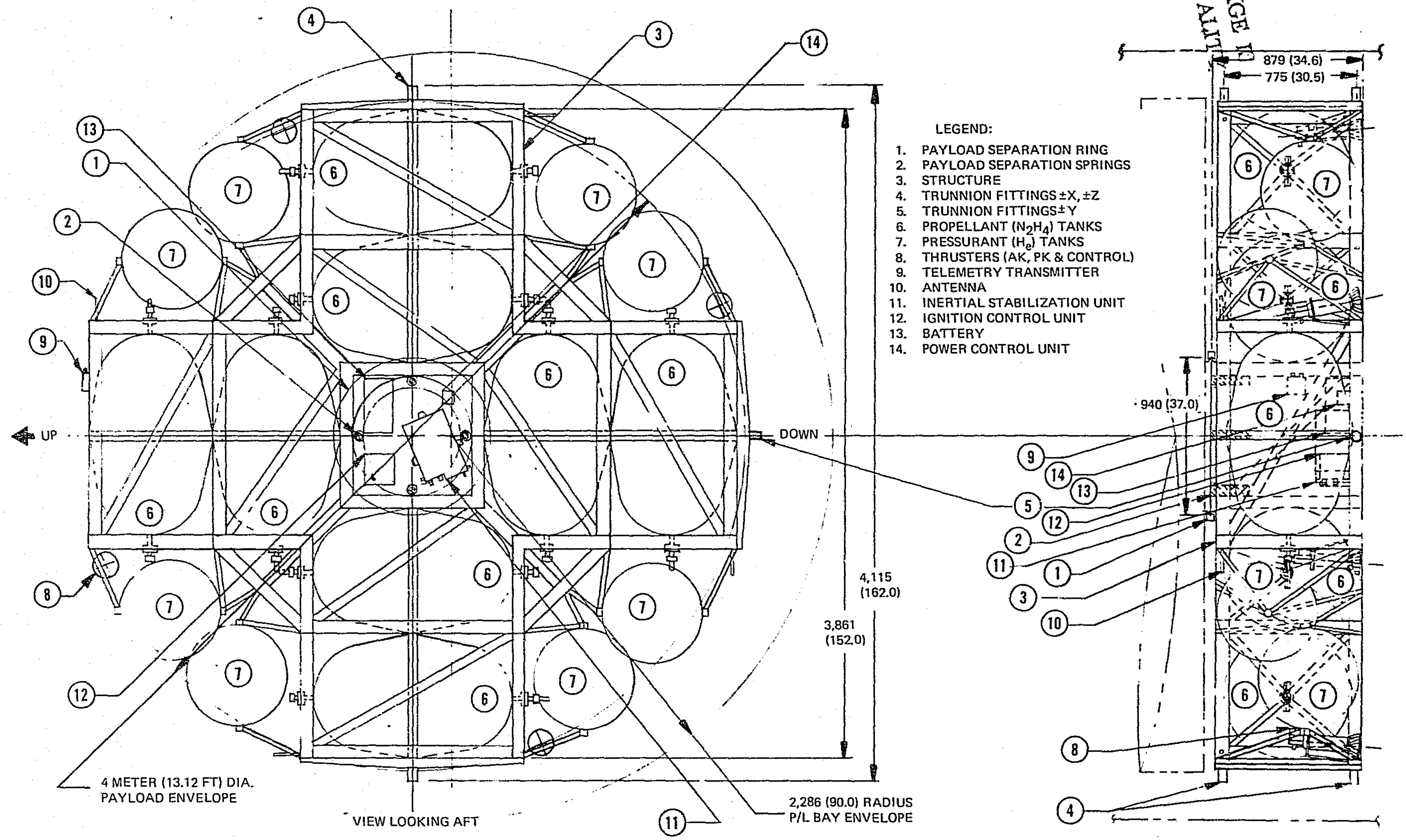


FIGURE 4.30 EIGHT TANK MODULAR MONOPROPELLANT LES

(from the center of the stage for four of the tanks and from the corners further outward for the other four tanks, see Figures 4.24 and 4.30).

The modular structural arrangement concept is harder to apply in the case of the monopropellant systems due to less advantageous load paths. The lower performance required for the smaller velocity increment payloads coupled with the large individual propellant tank size permits handling this class of payloads with a two-tank monopropellant LES mounted horizontally (Figure 4.31). The arrangements developed for vertical mounting is shown by Figure 4.32. The vertical mounting differs in that the width has been reduced by relocation of the two pressurization tanks and the telemetry transmitter to achieve a 635 mm (25.0 in.) cargo bay length reduction over the case of horizontal mounting of the LES shown by Figure 4.31.

The configurations for the monopropellant-type "adaptation" stages are shown by Figure 4.33 for the SSUS-D booster stage and Figure 4.34 for the SSUS-A booster stage. These two-stage units are utilized in the same manner as the bipropellant adaptations discussed in paragraph 4.5.1. A small potential for user's charge length reduction exists in the case of horizontal mounting in the SSUS-A cradle for either the bipropellant or monopropellant "adaptation" stages through the use of pretilting of the SSUS-A cradle spin table to a 10° to 15° tilt-up angle. Pretilting to a greater angle by rotating the LES delivery stage by 90° could offer another reduction in payload bay length. However, lock down during STS transit with the stages pretilted woudd require modification of the SSUS-A cradle assembly and might not prove to be cost effective.

4.6 MASS PROPERTIES

The design philosophy for the modular liquid propellant LES vehicle dictated as much commonality as practical between differing configurations. Achieving varied performance levels under this philosophy was accomplished by using differing quantities of identical or near identical components. This design approach resulted in only minor weight penalties for the modular concept.

It was desirable from a Shuttle user charge standpoint to have as small a system as possible, therefore the structure was constrained

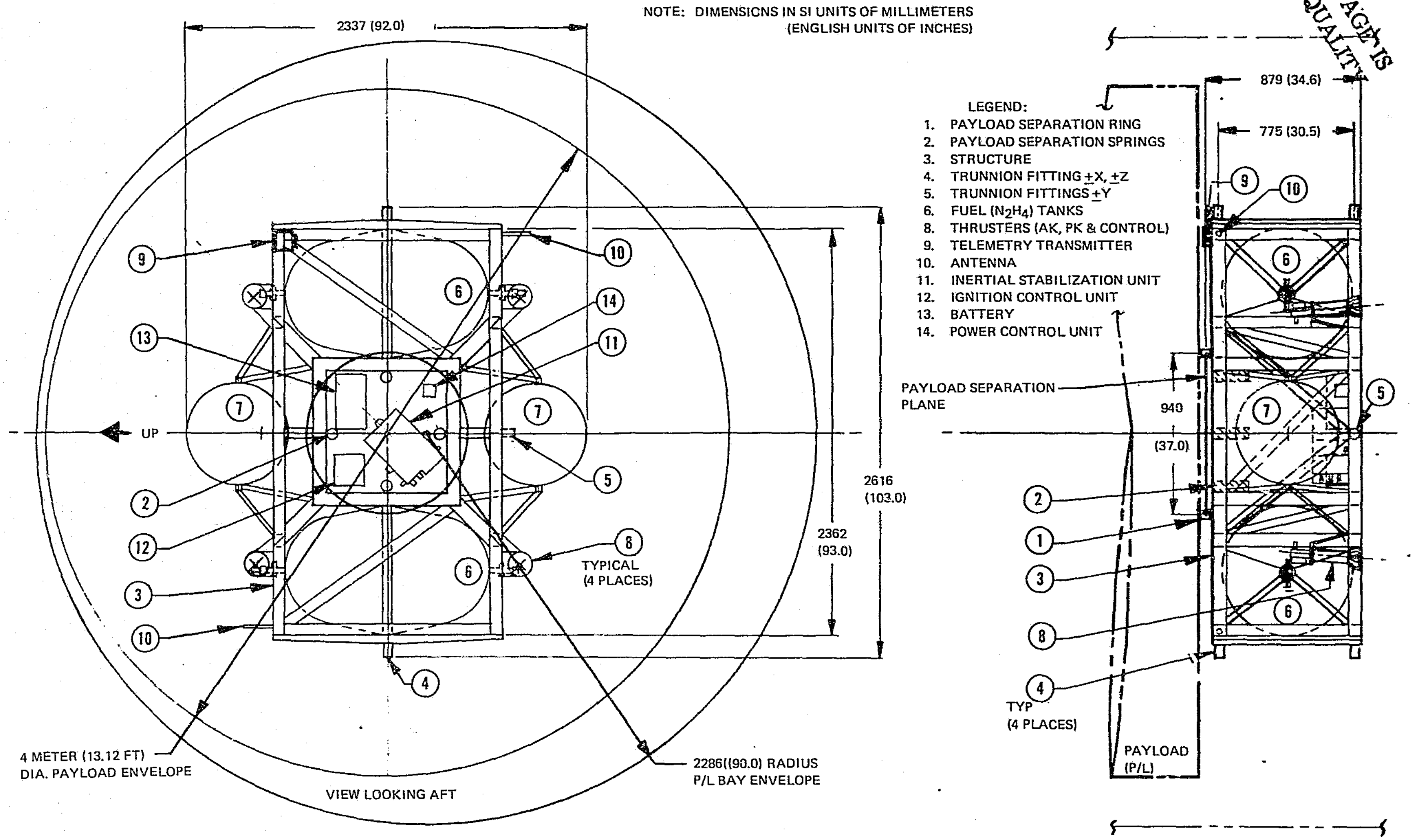


FIGURE 4.31 TWO TANK MODULAR MONOPROPELLANT LES

NOTE: DIMENSIONS IN SI UNITS OF MILLIMETERS
(ENGLISH UNITS OF INCHES)

LEGEND:
1. PAYLOAD SEPARATION RING
2. PAYLOAD SEPARATION SPRINGS
3. STRUCTURE
4. TRUNNION FITTINGS ±X, ±Z
5. TRUNNION FITTINGS ±Y
6. FUEL (N_2H_4) TANKS
7. PRESSURANT (H_e) TANKS
8. THRUSTERS (AK, PK & CONTROL)
9. TELEMETRY TRANSMITTER
10. ANTENNA
11. INERTIAL STABILIZATION UNIT
12. IGNITION CONTROL UNIT
13. BATTERY
14. POWER CONTROL UNIT

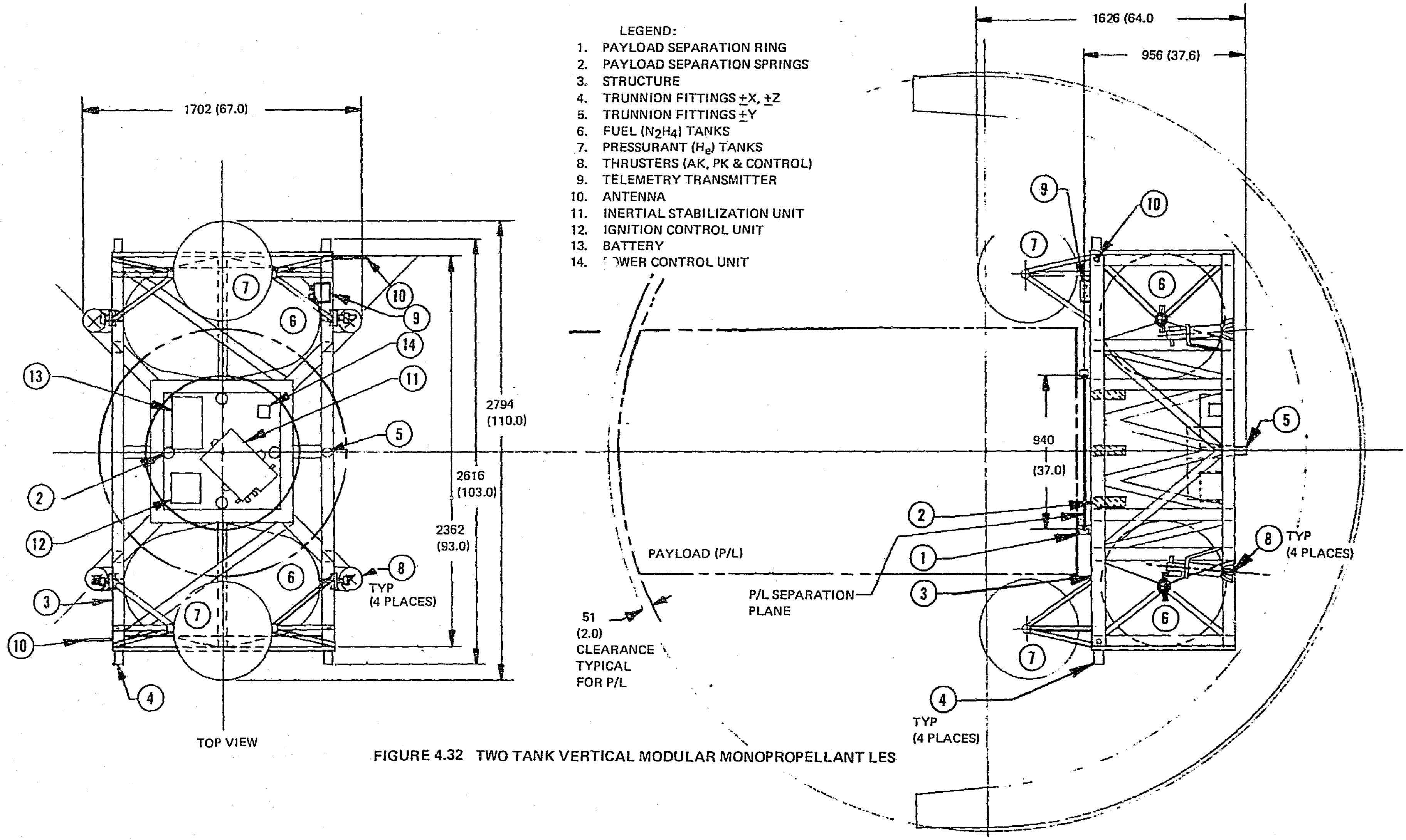


FIGURE 4.32 TWO TANK VERTICAL MODULAR MONOPROPELLANT LES

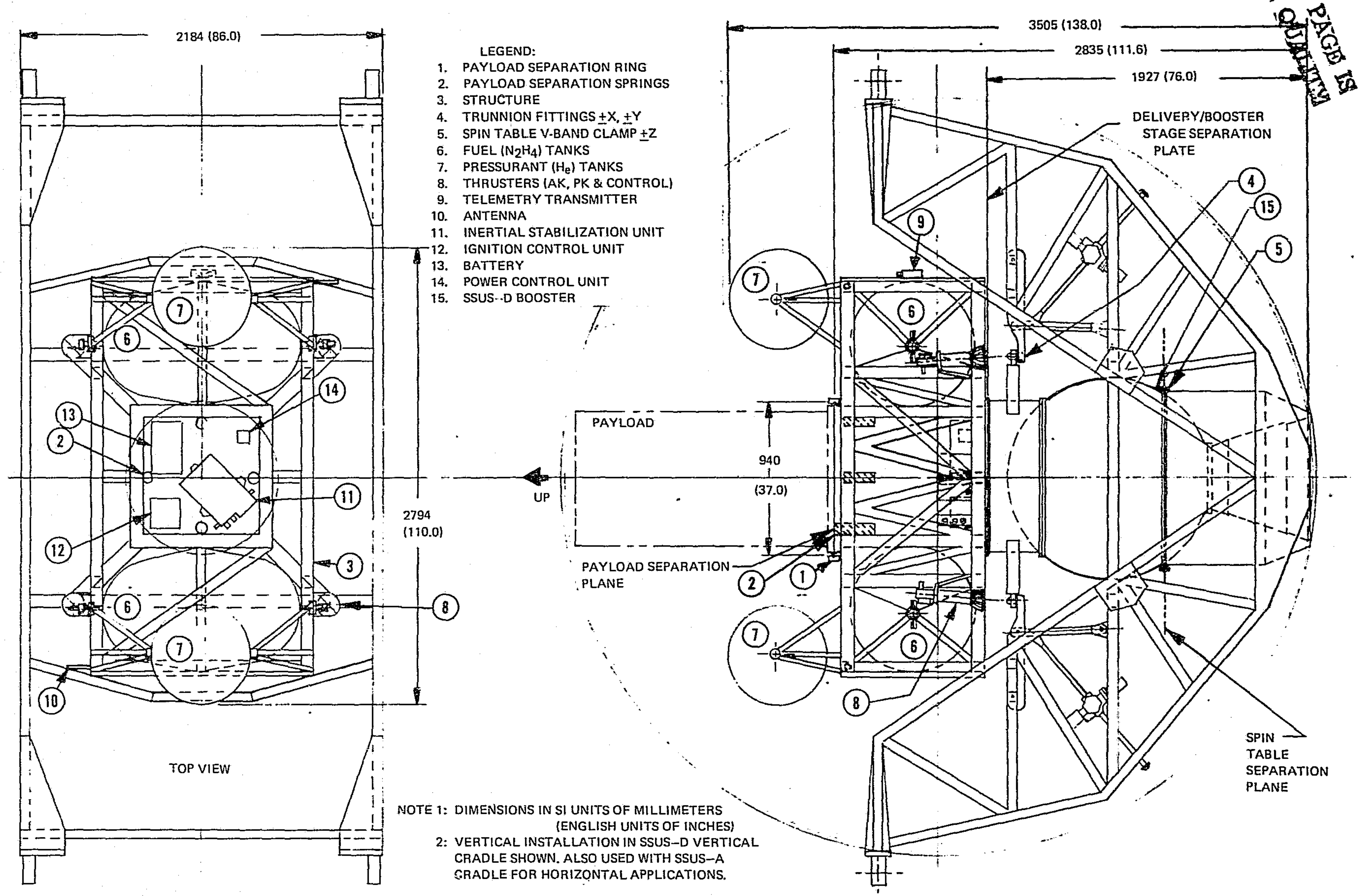


FIGURE 4.33 TWO TANK MODULAR MONOPROPELLANT/SSUS–D (VERTICAL OR HORIZONTAL)

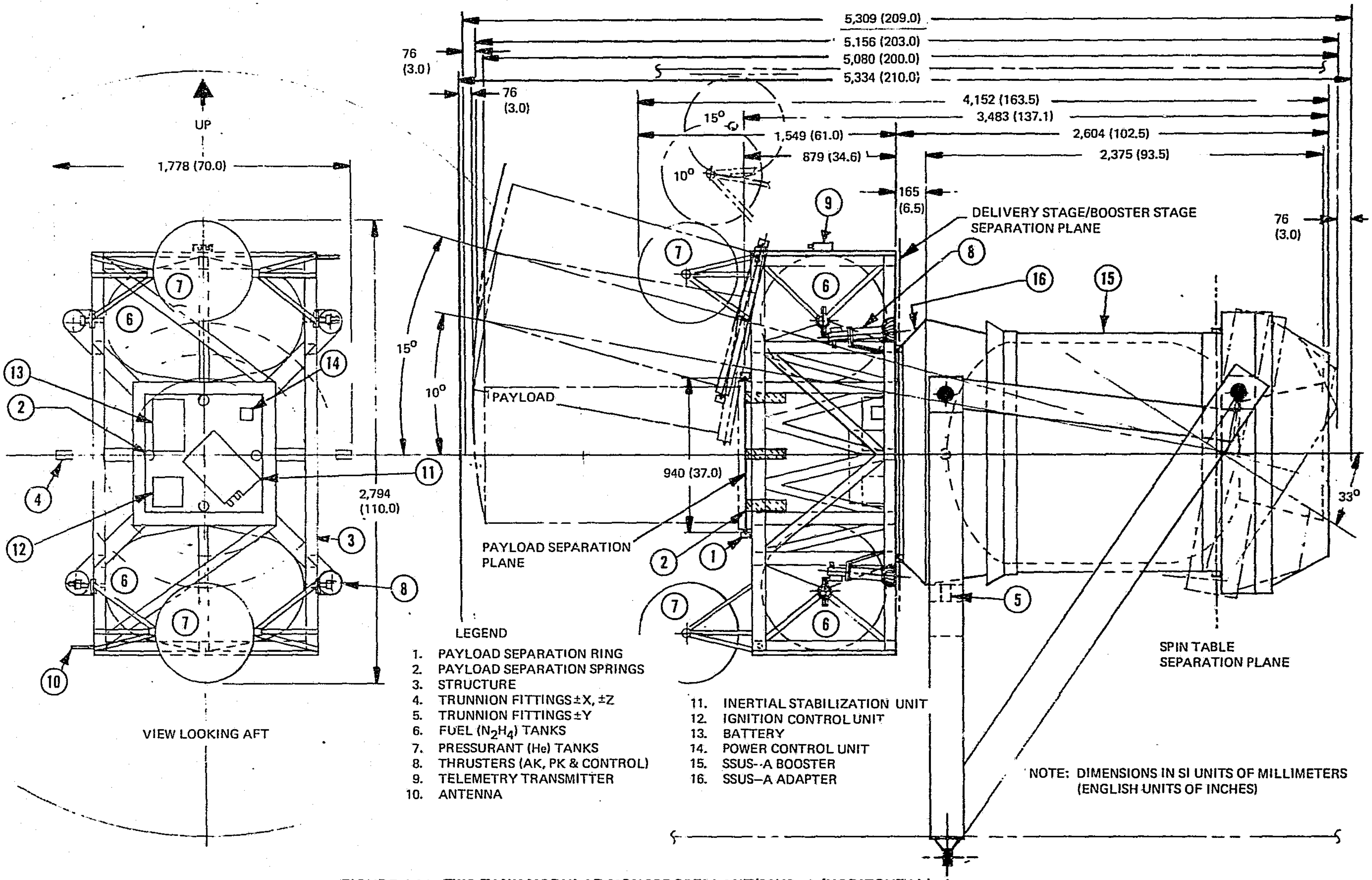


FIGURE 4.34 TWO TANK MODULAR MONOPROPELLANT/SSUS–A (HORIZONTAL)

to add no more size than practical to the system. Structural weight is directly related to the propellant tank volume required as the propellant tanks are the largest components on the vehicle.

The structural weight of the LES vehicle was derived from parametric data generated from detail structural sizing studies discussed in 4.2.2. Preliminary structural sizing was very close to the final structural size in terms of loads and physical characteristics. This enhanced the validity of the structural weight calculations. Structural weight values given in the following weight summaries include support weight for all vehicle systems. Systems supports represent approximately 20% of the structural weight specified in the summaries.

Vehicle unit cost dictated using existing equipment where ever possible. The mass properties derivation was generated from a dual level of analysis with approximately 80% of the vehicle empty weight obtained directly from existing sources and the remaining empty weight being estimated from parametric or detail sizing analysis. A 10% contingency was was applied to all non-propulsion system items.

Table 4-XXV details those components selected from existing hardware. Information is also included on physical size and manufacturer.

4.6.1 Modular Bipropellant Configurations

The most critical payload requirements occurred in the 8-tank payload group, thus establishing propellant tank volume and hence physical size of the vehicle. This same tank size was used for the 4-tank vehicle. Bipropellant vehicles with only 2 propellant tanks were not considered due to the center of mass variation produced as fuel and oxidizer of different specific weight are consumed. For this reason 100 percent and 50 percent propellant load conditions were used for the bipropellant performance and cost comparisons. The prepackaged propellant tanks were loaded at the propellant loading facility and delivered direct to the launch site in either 100 or 50 percent loaded condition as required for the scheduled launch. See paragraph 4.7.1 for performance comparisons.

Table 4-XXVI presents a weight summary of the modular bipropellant configuration. All weights are given for full propellant tanks.

TABLE 4-XXV COMPONENT SUMMARY

ITEM	SOURCE	WEIGHT KG	WEIGHT (LB)	STATUS	SIZE CENTIMETERS	SIZE INCHES
R-40A Bipropellant Engine	Marquardt	9.75	21.5	E	26.7 Ø x 47.2 L	10.5 Ø x 18.6 L
R-4D Bipropellant Engine	Marquardt	2.40	5.3	E	16.5 Ø x 34.0 L	6.5 Ø x 13.4 L
MR-104 Monopropellant Engine	Rocket Research	2.27	5.0	E	11.9 Ø x 39.4 L	4.7 Ø x 15.5 L
Attitude Control System Computer	Scout	20.46	45.1	E	39.4 x 28.7 x 19.6	15.5 x 11.3 x 7.7
Telemetry Transmitter	Scout	0.91	2.0	E	11.7 x 11.7 x 3.6	4.6 x 4.6 x 1.4
Battery	Eagle Picher	14.52	32.0	E	33.0 x 12.7 x 17.8	13.0 x 5.0 x 7.0
Squib Valve	Quantic	0.73	1.6	E	8.9 x 8.9 x 8.9	3.5 x 3.5 x 3.5
Check Valve	Rocketdyne	0.27	0.6	E	3.8 Ø x 8.9 L	1.5 Ø x 3.5 L
Pressure Regulator	New	1.59	3.5	M	10.2 x 20.3 x 7.6	4.0 x 8.0 x 3.0
Thrust Neutralizer	Vought	0.82	1.8	E	10.2 Ø x 12.7 L	4.0 Ø x 5.0 L
Relief Valve	Parkerh.	0.77	1.7	E	8.9 Ø x 17.8 L	3.5 Ø x 7.0 L
Pressure Transducer	Teledyne	0.09	0.2	E	----	----
Thermocouple	Stock Item	0.05	0.1	E	----	----
Thermistor	Stock Item	0.05	0.1	E	----	----
Fill & Drain Valve - Propellant	Futurecraft	0.14	0.3	E	2.0 Ø x 2.5 L	0.8 Ø x 1.0 L
- Pressurant	Purolator	0.23	0.5	E	5.1 Ø x 7.6 L	2.0 Ø x 3.0 L
Manual Propellant Drain Valve	Futurcraft	0.68	1.5	E	6.4 Ø x 17.8 L	2.5 Ø x 7.0 L
Propellant Filter	Winter	0.59	1.3	E	2.5 Ø x 15.2 L	1.0 Ø x 6.0 L
Pressurant Tank Bipropellant	P.S.I.	10.57	23.3	E	39.4 Ø	15.5 Ø
Propellant Tank Bipropellant	ARDE	30.03	66.2	M	63.5 Ø x 101.6 L	25.0 Ø x 40.0 L
Monopropellant	ARDE	48.63	107.2	M	74.1 Ø x 118.6 L	29.2 Ø x 46.7
Pressurant Tank Monopropellant	---	40.01	88.2	New	59.4 Ø	23.4 Ø
Ignition Control Unit Biprop	---	4.31	9.5	New	15.7 x 15.7 x 15.7	6.2 x 6.2 x 6.2
Monoprop	---	3.18	7.0	New	14.0 x 14.0 x 14.0	5.5 x 5.5 x 5.5
Telemetry Antenna	Tecom	0.05	0.1	E	7.6 x 7.6 x 1.5	3.0 x 3.0 x 0.6
Coaxial Switch	Transco	0.36	0.8	New	9.7 x 9.1 x 8.1	3.8 x 3.6 x 3.2
Power Control Unit	---	0.68	1.5	New	7.6 x 7.6 x 7.6	3.0 x 3.0 x 3.0

E = Existing, M = Modified Existing, Ø = Diameter, L = Long

TABLE 4-XXVI LOW ENERGY STAGE BIPROPELLANT CONFIGURATION WEIGHT SUMMARY

CONFIGURATION

WBS	ELEMENT DELIVERY	MODULAR 8 TANK HORIZONTAL		MODULAR 4 TANK HORIZONTAL		MODULAR 4 TANK VERTICAL		MODULAR 4 TANK VERTICAL + SSUS-D		MODULAR 4 TANK VERTICAL + SSUS-A		MODULAR 12 TANK VERTICAL		MODULAR 12 TANK VERTICAL FRAME WITH 4 TANKS	
		KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)
0221	Stage/Payload Sep.	7.7	17	7.7	17	7.7	17	7.7	17	7.7	17	7.7	17	7.7	17
0222	Structure & Supports	85.7	189	68.0	150	96.6	213	108.0	238	108.0	238	122.5	270	95.7	211
0223	Thermal	26.8	59	24.5	54	24.5	54	24.5	54	24.5	54	27.2	60	27.2	60
0224	Propulsion Inerts	384.2	847	201.4	444	201.4	444	201.4	444	201.4	444	566.5	1249	201.4	444
0224	Trapped Propellant	57.2	126	30.8	68	30.8	68	30.8	68	30.8	68	83.5	184	30.8	68
0224	RCS Propellant	7.7	17	7.7	17	7.7	17	89.4	197	103.0	227	7.7	17	7.7	17
0224	Mixture Ratio Tol.	20.0	44	9.5	21	9.5	21	9.5	21	9.5	21	30.8	68	9.5	21
0224	Pressurant	8.6	19	4.5	10	4.5	10	4.5	10	4.5	10	13.1	29	4.5	10
0225	Reaction Control	11.3	25	11.3	25	11.3	25	11.3	25	34.0	75	11.3	25	11.3	25
0226	Data Mngmnt/Communication	2.7	6	2.7	6	2.7	6	2.7	6	2.7	6	2.7	6	2.7	6
0227	Guidance/Navigation	20.4	45	20.4	45	20.4	45	20.4	45	20.4	45	20.4	45	20.4	45
0227	Ignition Control	4.5	10	4.5	10	4.5	10	4.5	10	4.5	10	4.5	10	4.5	10
0228	Electrical	26.3	58	23.6	52	23.6	52	23.6	52	23.6	52	29.5	65	23.6	52
	Contingency	18.6	41	16.3	36	19.1	42	20.4	45	21.3	47	22.2	49	19.5	43
	Stage Inerts	681.7	1503	432.9	955	464.3	1024	558.7	1232	595.9	1314	949.6	2094	466.5	1029
	Δ V Consumables	1669.2	3680	829.2	1828	829.2	1828	747.5	1648	733.9	1618	2509.3	5532	829.2	1828
	Stage Ignition	2350.9	5183	1262.1	2783	1293.5	2852	1306.2	2880	1329.9	2932	3458.9	7626	1295.7	2857
	Booster Adaptation	---	---	---	---	---	---	1754.5	3868	3770.3	8312	---	---	---	---
	Total Stage Weight	2350.9	5183	1262.1	2783	1293.5	2852	3060.7	6748	5100.2	11244	3458.9	7626	1295.7	2857
		M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)
	Length-Installed Horiz./Vert.	.77/-	2.53/-	.77/-	2.53/-	-/1.70	-/5.58	3.52/ 2.18	11.55/ 7.15	3.57/-	11.71/-	-/1.58	-/5.18	-/1.58	-/5.18
	Stage Diameter	3.96	12.99	2.69	8.83	3.28	10.76	3.12	10.24	1.78	5.84	4.07	13.35	4.07	13.35
	Number of Propellant Tanks	8		4		4		4		4		12		4	
	Booster Adaptation	-		-		-		SSUS-D		SSUS-A		-		-	

The weight of the 50 percent loaded configuration can be obtained by subtracting 50 percent of the ΔV consumables listed in the table. The structural weight for adaptations of the 4-tank bipropellant vehicles to SSUS-D and SSUS-A have been increased to account for LES/SSUS interface.

4.6.2 Modular Monopropellant Configurations

Modular monopropellant vehicle sizing was performed similar to the bipropellant vehicle. Modular monopropellant vehicles selected were an 8-tank, 2-tank and 2-tank/SSUS adaptations. Again the 8-propellant tank vehicle determined the propellant tank size. Table 4-XXVII presents a weight summary of the modular monopropellant configuration. All weights are for full propellant tanks.

4.6.3 Inertia Data

Figure 4-35 and Figure 4-36 present inertia and center-of-mass variation for propellant consumption typical of the 8-tank bipropellant vehicle with a small and large payload respectively. This data is typical of data used in determining reaction control system requirements as described in paragraph 4.2.5.

4.7 CONCEPT PERFORMANCE

The performance of the LES concepts refined in Task 3 was determined in order to establish the specific LES mission model capture of each configuration. The performance of each configuration was computed based on the ideal velocity equation which assumes that the velocity capability of the stage is imparted instantaneously and that the specific impulse is constant. The configuration mass properties and propulsion characteristics presented and discussed in paragraphs 4.6 and 4.3, respectively, were used as the basis for performance capabilities.

The performance capability (velocity increment) is presented parametrically as a function of spacecraft mass. A performance curve for each configuration is overlayed on the LES Mission Model mass/energy requirements of paragraph 4.1.3 so that the relationship of each configuration's performance capability to the Mission Model performance requirements could be easily assessed.

TABLE 4-XXVII LOW ENERGY STAGE MONOPROPELLANT CONFIGURATION WEIGHT SUMMARY

CONFIGURATION

WBS	ELEMENT DELIVERY	MODULAR 8 TANK HORIZONTAL		MODULAR 2 TANK HORIZONTAL		MODULAR 2 TANK VERTICAL		MODULAR 2 TANK VERTICAL WITH SSUS-D		MODULAR 2 TANK VERTICAL WITH SSUS-A	
		KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)
0221	Stage/Payload Sep.	7.7	17	7.7	17	7.7	17	7.7	17	7.7	17
0222	Structure & Supports	107.0	236	71.7	158	71.7	158	83.0	183	83.0	183
0223	Thermal	27.2	60	24.5	54	24.5	54	24.5	54	24.5	54
0224	Propulsion Inerts	760.2	1676	206.4	455	206.4	455	206.4	455	206.4	455
0224	Trapped Propellant	87.1	192	30.8	68	30.8	68	30.8	68	30.8	68
0224	RCS Propellant	0	0	0	0	0	0	9.1	20	22.7	50
0224	Mixture Ratio Tol.	0	0	0	0	0	0	0	0	0	0
0224	Pressurant	29.9	66	7.7	17	7.7	17	7.7	17	7.7	17
0225	Reaction Control	0	0	0	0	0	0	13.6	30	27.2	60
0226	Data Mngmnt/Communication	2.7	6	2.7	6	2.7	6	2.7	6	2.7	6
0227	Guidance/Navigation	20.4	45	20.4	45	20.4	45	20.4	45	20.4	45
0227	Ignition Control	4.5	10	4.5	10	4.5	10	4.5	10	4.5	10
0228	Electrical	27.2	60	22.7	50	22.7	50	22.7	50	22.7	50
	Contingency	19.5	43	15.4	34	15.4	34	17.7	39	19.5	43
	Stage Inerts	1093.4	2411	414.5	914	414.5	914	450.8	994	450.8	1054
	Δ V Consumables	2412.2	5318	594.2	1310	594.2	1310	585.1	1290	571.5	1260
	Stage Ignition	3505.6	7729	1008.7	2224	1008.7	2224	1035.9	2284	1035.9	2314
	Booster Adaptation		--		--		--	1754.5	3868	3770.3	8312
	Total Stage Weight	3505.6	7729	1008.7	2224	1008.7	2224	2790.4	6152	4806.2	10627
		M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)	M.	(FT.)
	Length ~ Installed Horiz./Vert.	.88/-	2.89/-	.88/	2.89/-	-/1.70	-/5.58	3.56/ 2.18	11.68/ 7.15	3.61/-	11.84/-
	Stage Diameter	4.11	13.48	2.62	8.60	2.79	9.15	2.79	9.15	1.78	5.84
	Number of Propellant Tanks	8		2		2		2		2	
	Booster Adaptation	-		-		-		SSUS-D		SSUS-A	

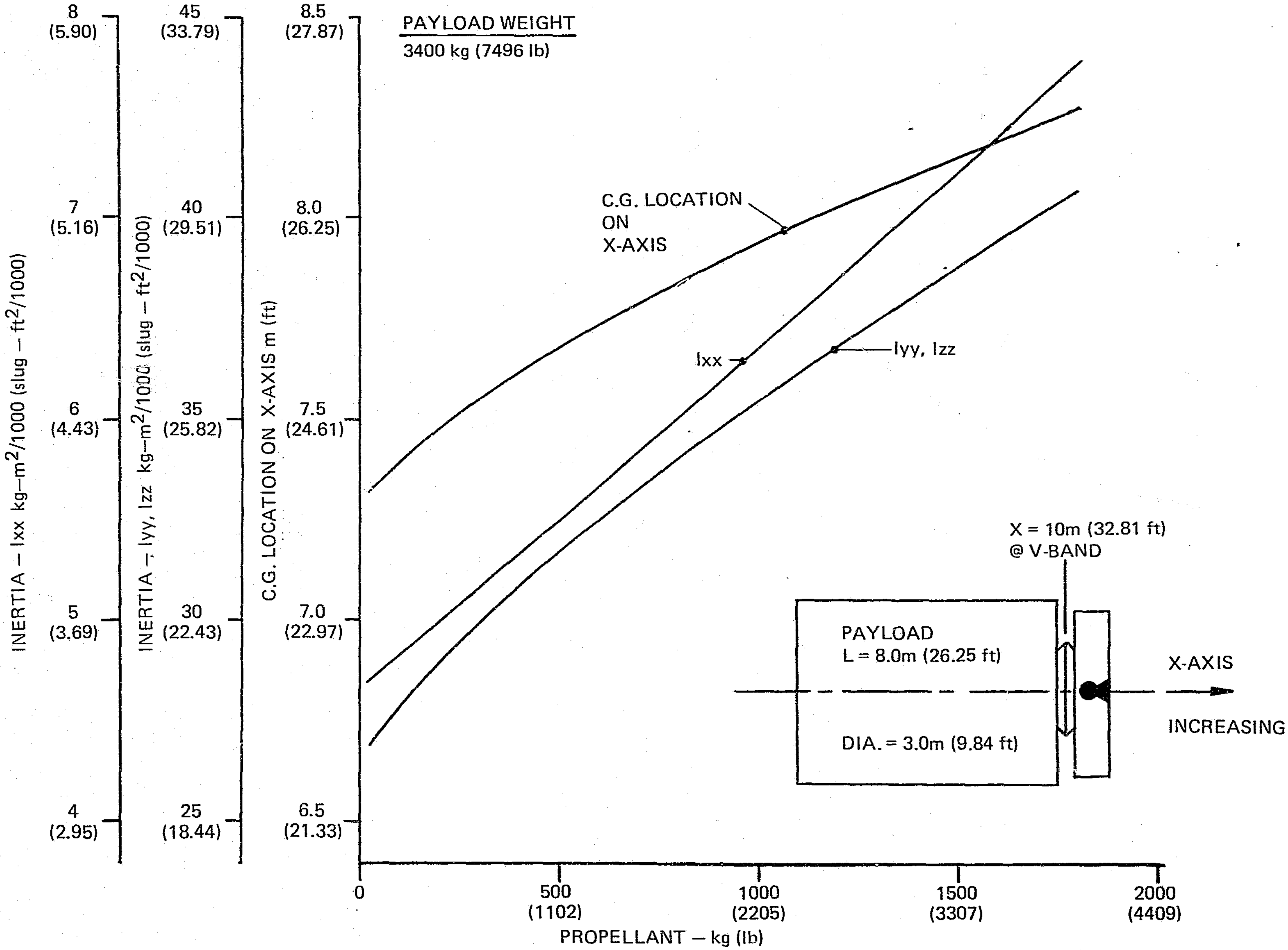


FIGURE 4.35 TYPICAL 8 TANK BIPROPELLANT SYSTEM INERTIA

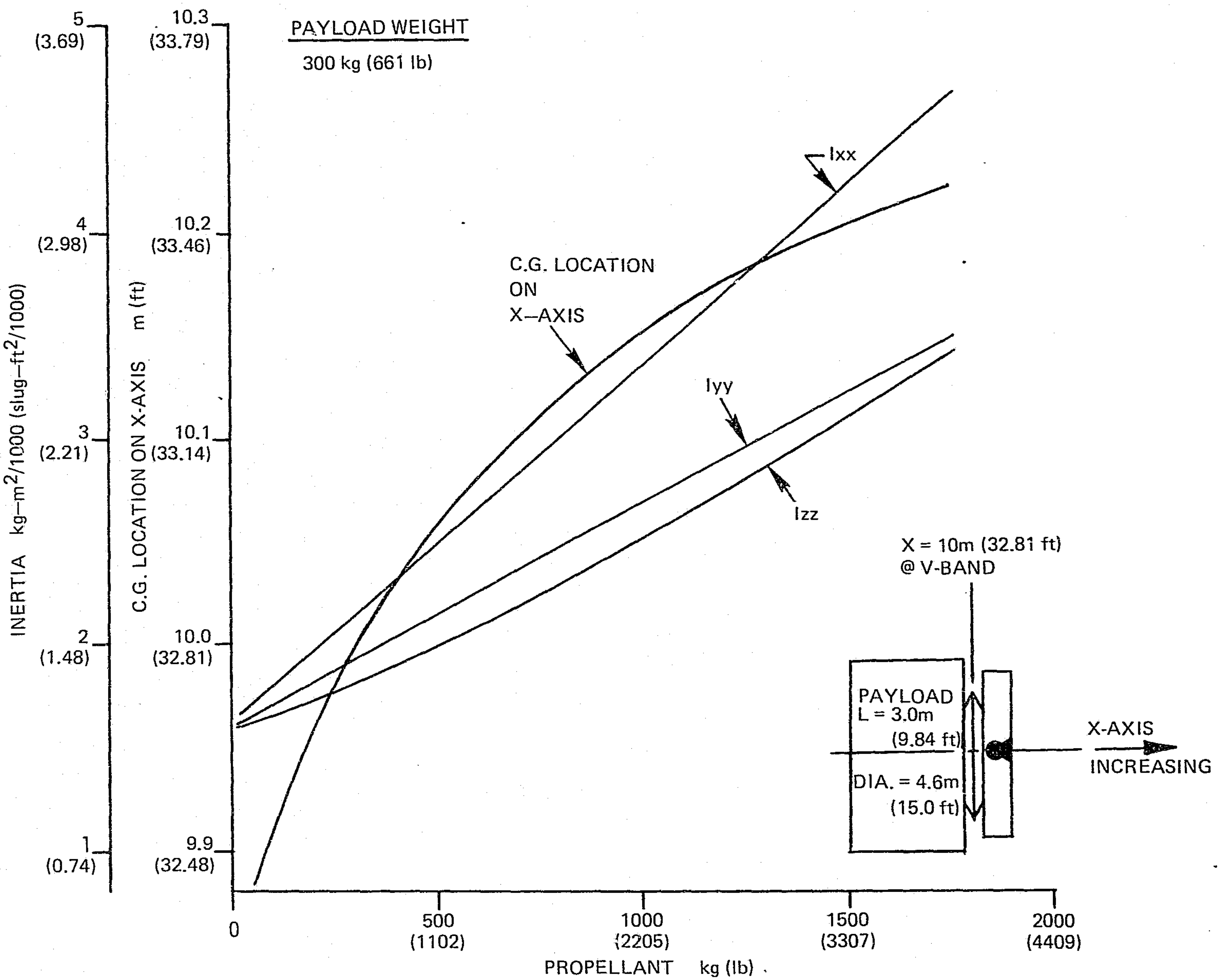


FIGURE 4.36 TYPICAL 8 TANK BIPROPELLANT SYSTEM INERTIA

4.7.1 Bipropellant Configurations

The performance of the liquid bipropellant configurations is presented in Figure 4.37. The 8-tank horizontal and 12-tank vertical modules have about the same performance and could handle 99% of the missions in the Low Energy Regime; all except mission #48, the Scout San Marco D[1] mission. The 4-tank horizontal and vertical modular versions and the 50% off load of these can handle 86% and 85% of the model, respectively. The 50 percent propellant loaded configuration produces performance approximately that of the 100 percent loaded 2-tank monopropellant and the 100 percent loaded TRS resulting in a sound basis for comparison. Refer to Figure 4.37 and Volume IV, Figure 7.2 for performance comparison.

4.7.2 Monopropellant Configurations

The performance of the liquid monopropellant configurations is presented in Figure 4.38. The 8-tank module captures 96% of the model while the two tank horizontal and vertical modular versions capture 85% of the model.

4.7.3 Adaptations

The performance of adaptations of the existing/planned STS delivery systems (SSUS-D and SSUS-A) using various LES bipropellant and monopropellant concepts is presented in Figure 4.39. All of these configurations produce a sufficient velocity increment to handle all of the mission model, except mission #49, but were used only for those missions which require in excess of 2000 to 3000 m/sec. There is not a significant difference in adaptation performance between bipropellant and monopropellant delivery stages. The adaptation performance is influenced primarily by the use of either SSUS-A or SSUS-D as the booster stage of these configurations.

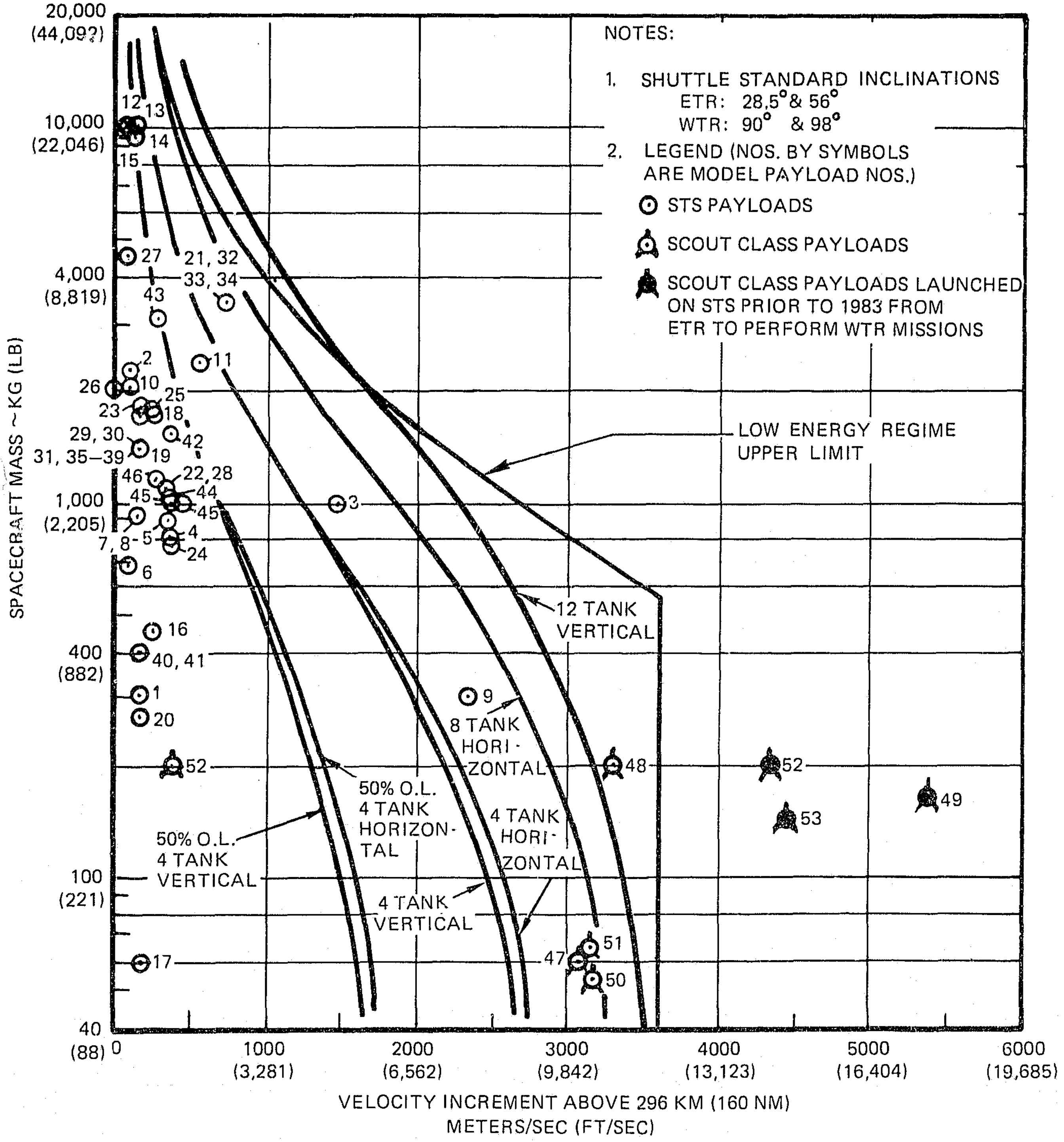


FIGURE 4.37 MODULAR BIPROPELLANT LOW ENERGY STAGE PERFORMANCE

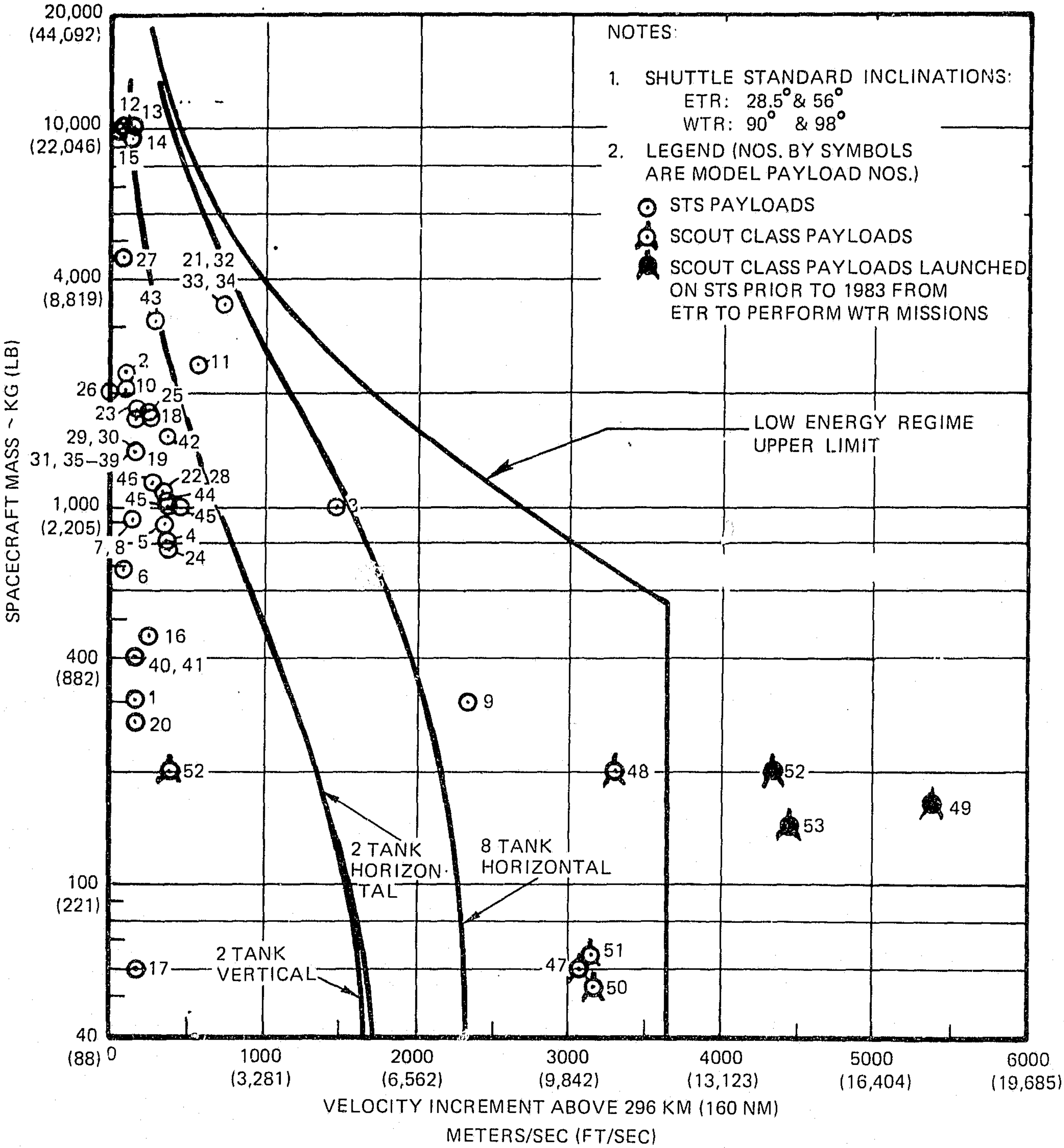


FIGURE 4.38 MODULAR MONOPROPELLANT LOW ENERGY STAGE PERFORMANCE

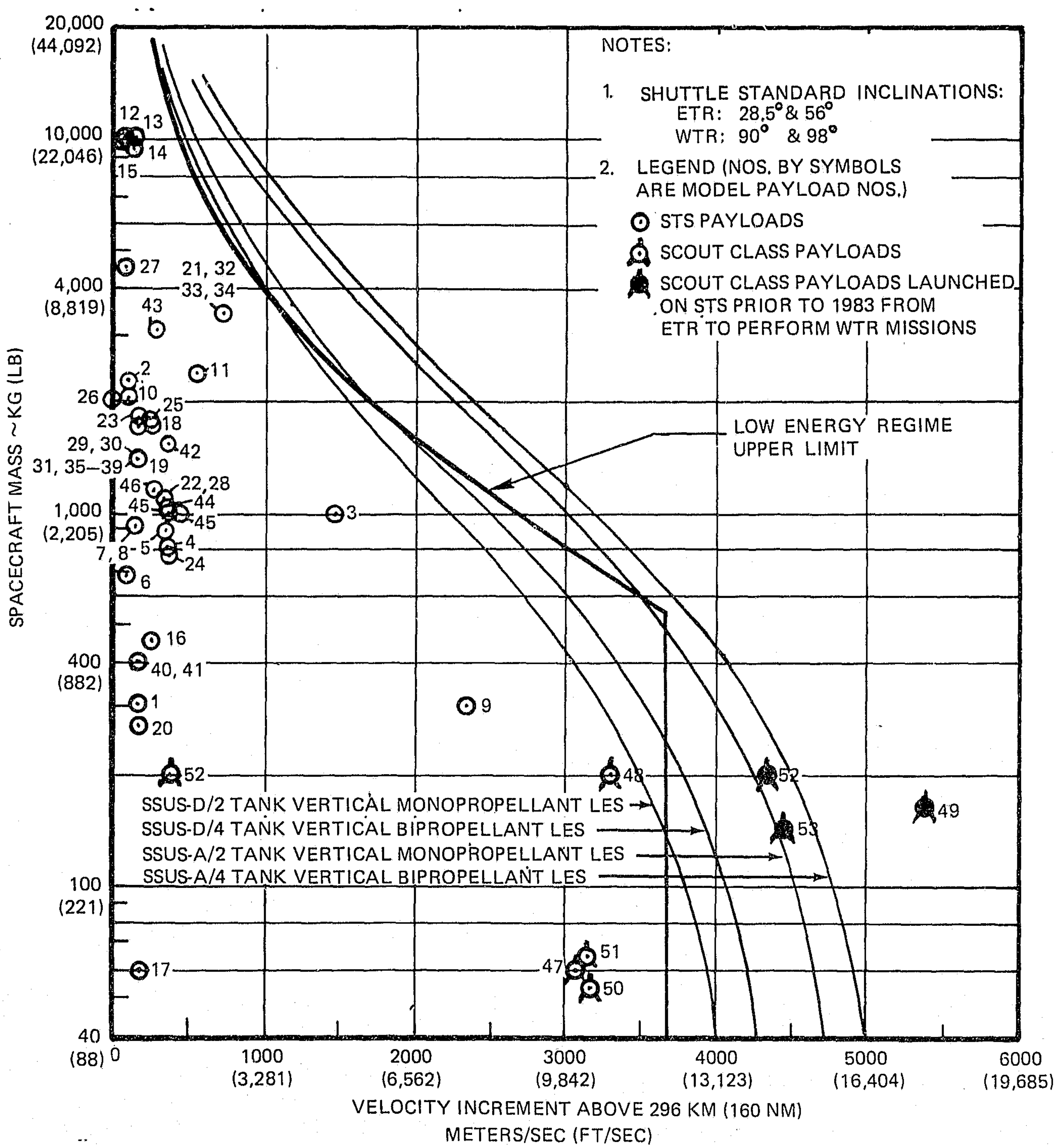


FIGURE 4.39 ADAPTATIONS PERFORMANCE

4.8 INTEGRAL PROPULSION SYSTEM

The purpose of this special task was to conduct a brief investigation of the use of an integral propulsion system which depends on the spacecraft to provide common functions instead of separately providing these functions as a part of the Low Energy Stage. These functions were guidance and attitude control commands, power, communications and data handling. This investigation covered two areas of interest: Cost variances and configuration evaluations. Three options were selected to scope the potential cost variances. Two options used the LES 8-tank bipropellant concept and one was based on the LES components integrated with the spacecraft structure. The configuration integration investigation considered three options using the MMS and LES 8-tank bipropellant configuration. The MMS was used because it was the more mature spacecraft design specified in the payload model (paragraph 4.1).

4.8.1 Cost Variance

The basis for this investigation was the Low Energy Stage Study payload model. This data was used to determine the number of propulsion system designs and contractors involved for each integral option approach. The brevity of the available spacecraft descriptions limited the depth of the investigation. However, the intent of this task was achieved by considering capability and cost deltas from the bipropellant low energy stage approach. The following two considerations expressed in the payload model report were important to this investigation.

(1) The LES Model covers the years 1980-1991. As with all payload models, data on near-term missions (e.g., missions within a current five-year planning cycle) was much more reliable than data on missions projected for the later years (1985 and beyond). Missions defined in the model for the mid-to-late 1980's should properly be viewed as examples of the kind of endeavors that might take place at a certain projected level of future activity. Specific definitions should not be weighed too heavily or be regarded as totally inflexible in determining the requirements of supporting propulsion systems.

(2) Just as the 487 Payload Model is geared to reflect STS requirements, the LES Payload Model is geared to reflect the requirements for a low energy stage. Both models tend to occasionally modify the intent of the individual mission planner in producing mission definitions that comply with groundrules and governing assumptions about future NASA policy. In the case of the LES Payload Model, there were instances where a mission was based upon a non-baseline spacecraft definition. This occurred when the baseline, as viewed by the mission planner, included an integral propulsion system that would not be needed if a low energy stage were developed and used. Thus, although the LES Payload Model includes updated and sometimes more definitive data than the 487 Payload Model, it also incorporates some assumptions that make it peculiar to the low energy stage study application. Use of the LES mission data for other applications should be reviewed on a case by case basis.

Options selected for this investigation were:

(1) One propulsion system design tailored to the mission requirements of the LES payload model and attached to the spacecraft structure.

(2) Several propulsion system designs, each tailored to a specific class of the LES payload model requirements and attached to the spacecraft structure.

(3) Propulsion systems provided by a spacecraft contractor with propulsion and spacecraft components integrated on a common structure.

The cost variance estimates for these options included the impact on the spacecraft as well as the stage. Some stage cost reductions were off-set by increased spacecraft contractor responsibility because the supporting cost cannot be eliminated from the program. These supporting costs were: integration, documentation, interface control, integrated test and simulations, training, management, ground support equipment and operations support cost.

The cost for these items were esculated because they will not be a common one time expense between the many spacecraft contractors.

The payload model consists of 103 spacecraft requiring a low energy stage for transport from the Shuttle orbit to the spacecraft operational orbit.

With these considerations, the following trend results present a comparison of the options based on the bipropellant low energy stage propulsion system and mission scenario.

- Option (1)

This option consisted of the bipropellant LES design with the battery, telemetry, guidance, antennas, and power control equipment removed and considered to be a part of the spacecraft. Therefore, the LES bipropellant propulsion approach remained the same with the spacecraft providing these functions through interfacing hardware.

Cost variance considerations were:

a) LES

It was assumed one contractor would develop the propulsion stage based on a single design for the payload model spacecraft requirements.

- Production recurring cost reduction of the LES equipment deleted was $718,640 per stage with the total for the payload model being $74,019,920.
- DDT&E cost reduction estimate was $10,000,000.
- Shuttle user cost reduction estimate was zero because there were no configuration changes.
- Supporting recurring cost reduction estimate was $17,894,000.

b) Spacecraft

A review of the LES payload model indicated 15 different classes of spacecraft. It was assumed a different contractor would supply each class of spacecraft. Therefore, the propulsion stage designs for the LES payload model would involve 15 different spacecraft contractors.

- Production recurring cost increase estimate was zero because equipment to provide LES functional capability was basic to spacecraft design.

- DDT&E cost estimated cost increase was $1,000,000 per spacecraft contractor. This increase results because of propulsion system software development, management and interface responsibility that must be performed by the spacecraft contractor. The total increase for 15 spacecraft contractors was $15,000,000.
- Shuttle user cost increase estimate was zero because there were no configuration changes.
- Supporting recurring cost increase estimate was 25% of the $17,894,000 LES reduction. This was an increase of $4,473,500 for each spacecraft contractor because the responsibility was considered as an add-on to an existing spacecraft task. The total cost increase for the 15 spacecraft contractors was $67,102,500.

Table 4-XXVIII shows a summary cost comparison between LES and Option (1). The LES cost reduction of $102M was reduced by an $82M increase in spacecraft cost. Thus, a total program cost reduction potential of $20,000,000 exists for Option (1). This potential savings does not consider that some planned spacecraft may not include, as a basic spacecraft design requirement, equipment that will provide the LES functional requirements. It was estimated that to add or make major modification to spacecraft equipment to provide the necessary LES functions would cost $5,000,000 (DDT&E and equipment) per spacecraft class. If this cost impact was incurred on 4 of the payload model spacecraft classes, the predicted cost savings of $20,000,000 would not be realized.

Option (1) retains the following attractive features of the recommended LES concept:

- Stage design provides coverage of the Low Energy Regime. Therefore, for many of the future spacecraft, performance capability will accommodate growth without requiring stage modifications and associated DDT&E cost impact.

TABLE 4-XXVIII

OPTION (1) COST VARIANCES

ELEMENT OF COST	LES COST VARIANCE FOR THE PAYLOAD MODEL $M	SPACECRAFT COST VARIANCE FOR THE PAYLOAD MODEL $M
PRODUCTION	- 74	0
DDT&E	- 10	+ 15
SHUTTLE USER CHARGE	0	0
SUPPORTING RECURRING COST	- 18	+ 67
TOTALS	-102 Cost Reduction	+ 82 Cost Increase

- The one stage design produced by one prime contractor results in only one DDT&E cost.
- The use of one prime contractor and subcontractor team results in lower recurring stage costs.
- The use of the same stage for many missions results in a higher reliability achievement over the mission model life because of design maturity.
- The one common propulsion stage complements the prediction of achieving a lower space program cost when using the Shuttle STS for spacecraft delivery to its operational orbit.
- Stage design can be made modular so that deleted equipment which may not be a part of a spacecraft may be added to fulfill a mission requirement.

o <u>Option (2)</u>

The LES bipropellant configuration was selected for this option because of its short length and associated Shuttle user charge reduction. The cost impact of propulsion system designs for different classes of spacecraft were considered because it would permit designs based on more definitive spacecraft requirements. This results because the mission planning cycle would be near term (within 3 year cycle). Therefore, several propulsion system designs could be made smaller and thus at a lower cost.

A review of the LES mission model considering the spacecraft weight and velocity requirements indicated a split into four different class propulsion systems appeared reasonable. The cost variance between LES and this option were very similar to Option (1). Therefore, these cost guidelines are not repeated for this option. The cost value of the changes are different because four LES designs produced by the same contractor were considered in lieu of one design as in Option (1).

Cost variance considerations were:

a) <u>LES</u>

- Production recurring cost reduction estimates for deleted equipment was the same as Option (1) - $718,640 per stage for a total of $74,019,920. The four different designs result in a lower production run per design which was estimated to increase production cost by 11% for a total of $24,185,842. In addition, there will be minor changes in propulsion system size which was estimated at having no impact on cost. Therefore, the total reduction for the payload model was $49,834,078.

- DDT&E cost reduction estimate was the same as Option (1) - $10,000,000 for the first design. However, there will be DDT&E cost increases for other three designs. It was estimated the increase would be $1,000,000 per design based on the requirement spread. Therefore, a DDT&E cost increase of $5,000,000 was incurred.
- Shuttle user charge reduction was estimated for the payload model at 450 inches at $40,000 per inch for a cost reduction of $18,000,000.
- Supporting recurring cost was estimated to be the same as Option (1) - $17,894,000.

b) Spacecraft

It was assumed that with one contractor supplying four different propulsion system designs the impact on the spacecraft contractors cost will remain the same as Option (1).

- Production recurring cost change was zero.
- DDT&E cost increased $15,000,000.
- Shuttle user cost was not changed.
- Supporting recurring cost increased $67,102,500.

Table 4-XXIX shows a summary cost comparison between LES and Option (2). The LES cost reduction of $81M was off-set by an $82M increase in spacecraft cost. No cost improvement was achieved using this option.

Option (2) has the following features:

- Stage design point more closely tailored to better defined spacecraft requirements.
- Reliability decrement because of reduced design maturity on each design.

TABLE 4-XXIX

OPTION (2) COST VARIANCES

ELEMENT OF COST	LES COST VARIANCE FOR THE PAYLOAD MODEL $M	SPACECRAFT COST VARIANCE FOR THE PAYLOAD MODEL $M
PRODUCTION	- 50	0
DDT&E	+ 5	+ 15
SHUTTLE USER CHARGE	- 18	0
SUPPORTING RECURRING COST	- 18	+ 67
TOTALS	- 81 Cost Reduction	+ 82 Cost Increase

- <u>Option (3)</u>

This option considers a propulsion system more tailored to the spacecraft contractor or mission planner requirement as opposed to a defined propulsion stage the spacecraft must be tailored to fit. Thus, the propulsion design will be tailored to definitive spacecraft performance and configuration requirements. A review of the LES payload model was made considering one spacecraft contractor for each class of spacecraft and the velocity and weight requirements. From this review it was concluded that as a minimum 31 different propulsion system designs would result.

Without a common stage to force spacecraft design discipline, the many spacecraft contractors will produce a variety of different propulsion systems integrated around the spacecraft structure. Each propulsion system design will necessitate a separate DDT&E with only 1-5 propulsion systems produced per spacecraft contractor resulting in no production cost break because of volume. In addition, the integration, documentation, training, interface control, data management, GSE, management and operations cost will increase because of the many different contractor/subcontractor teams. Once the spacecraft becomes the driver for the propulsion system design, the Shuttle user charge will increase because achieving a short compact design may appear un-realistic to many different contractors. Also, the trend will be to set more precise propulsion performance requirements which will increase component price and DDT&E cost because of proliferation of component size, characteristics and cost.

In view of the above, a total propulsion system comparison for the payload model was made between the bipropellant LES and Option (3). It was assumed that Option (3) designs would be a bipropellant type.

Cost variance considerations were:

a) <u>LES</u>

The cost data for LES was taken from Table V of Volume V.

b) <u>Spacecraft</u>

- Production recurring cost for the 31 different propulsion designs was estimated to increase by 30% over the LES because of the very short production runs (1-5 units). The total cost was $241,930,000.

- DDT&E for the 31 different designs was estimated at an average of $12,000,000 each for a total cost of $372,000,000.
- Shuttle user charge was estimated to be the same as LES. It was considered the actual cost will be greater than LES. It is un-realistic to assume that 31 different spacecraft contractors will be so disciplined to propulsion system design that they will achieve a short compact design which is possible for one contractor.
- Supporting recurring cost was estimated at 300% over LES cost of $68.9M because of the 15 different contractor/subcontractor teams. The total cost increase was $206.7M.

Table 4-XXX shows a summary cost comparison between LES and Option (3). This option resulted in a major program cost increase of $540,000,000.

Based on the above results, the minimum total program cost for the LES mission model will be achieved by one common propulsion stage design. The primary reasons for this stems from:

- The design discipline imposed on the many different spacecraft contractors.
- The elimination of the cost of common propulsion system tasks being performed by many contractors as opposed to one contractor.

4.8.2 Configuration Evaluation

The LES payload model was reviewed and the MMS was selected as the spacecraft to be used for the investigation because of its available design data. The MMS contains three basic subsystems housed in modularized containers. These three modules can provide the electrical power, attitude control and stabilization, communication, data handling and command functions for the spacecraft and payload. Combining the MMS modules or the MMS with the LES propulsion modules, structure, and reaction control system, results in a length and weight efficient configuration.

TABLE 4-XXX

OPTION (3) COST COMPARISON

ELEMENT OF COST	LES COST FOR THE PAYLOAD MODEL $M	INTEGRATED SPACECRAFT PROPULSION COST FOR THE PAYLOAD MODEL $M
PRODUCTION	186	242
DDT&E	26	372
SHUTTLE USER CHARGE	189	189
SUPPORTING RECURRING COST	69	207
TOTALS	470	1,010

The MMS/LES concept I consists of a combination of either the LES bipropellant 4 or 8 tank configuration with the MMS as shown in Figure 4.40. In this concept the MMS Attitude Control System (ACS) module, Communication and Data Handling (C&DH) module, and Power module together with the MMS structure are attached to the LES 4 - or 8-tank bipropellant stage. All redundant subsystems in the LES (battery, telemetry transmitter, antennas, inertial stabilization unit, ignition control unit, and power control unit) were removed and the result was a LES weight savings of 54.2 kilograms (119.5 pounds) and a recurring unit cost savings of $718,640. The overall length of this resulting configuration was 2.56 meters (8.42 ft.) long and 0.48 meters (1.56 ft.) shorter than the MMS with the PM-II. This concept requires the minimum change to the MMS and the LES. This conceptual configuration was not compatible with the existing MMS Flight Support System (FSS); however, it should be compatible with the LES cradle.

Figure 4.41 shows the MMS/LES concept II. Again this arrangement utilizes the MMS modules and structure together with the LES structure and tankage. However, in this configuration the LES main thruster and the LES ACS thrusters were relocated from the LES to the MMS. This concept results in an overall length of 2.62 meters (8.59 ft.) and was 0.43 meters (1.42 ft.) shorter than the MMS with the PM-II. The weight savings due to elimination of the redundant subsystem hardware was 54.2 kilograms (119.5 pounds) and the subsystem recurring unit cost is $718,640. A primary advantage of this configuration was that it could be made compatible with the existing MMS/FSS.

The third concept was the MMS Modules/LES concept shown in Figure 4.42. This was the most efficient of the three concepts in terms of weight and length. This concept utilizes the MMS, ACS, C&DH, and power modules on the LES stage in lieu of the equivalent LES subsystems. The resulting configuration was 1.32 meters (4.33 ft.) in length and was 1.727 meters (5.67 ft.) shorter than the MMS with the PM-II. Again this concept saves 54.2 kilograms (119.5 pounds) of subsystem weight and $718,640 of subsystem recurring cost. This concept would require modification to the LES conceptual design to provide MMS module mounting provisions and it will also require the design of a payload transition adapter. The present MMS/FSS can

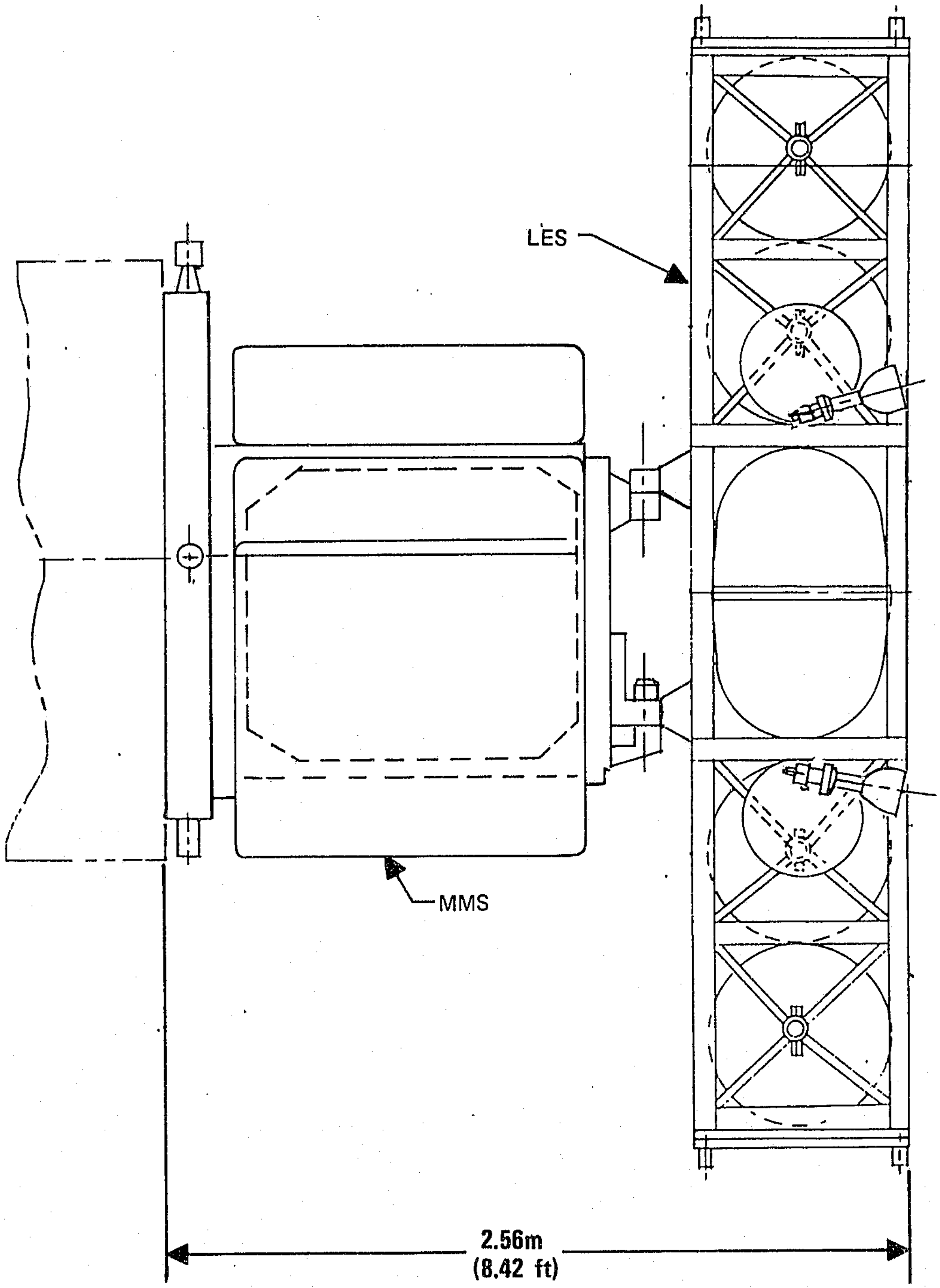


FIGURE 4.40 MMS/LES CONCEPT I

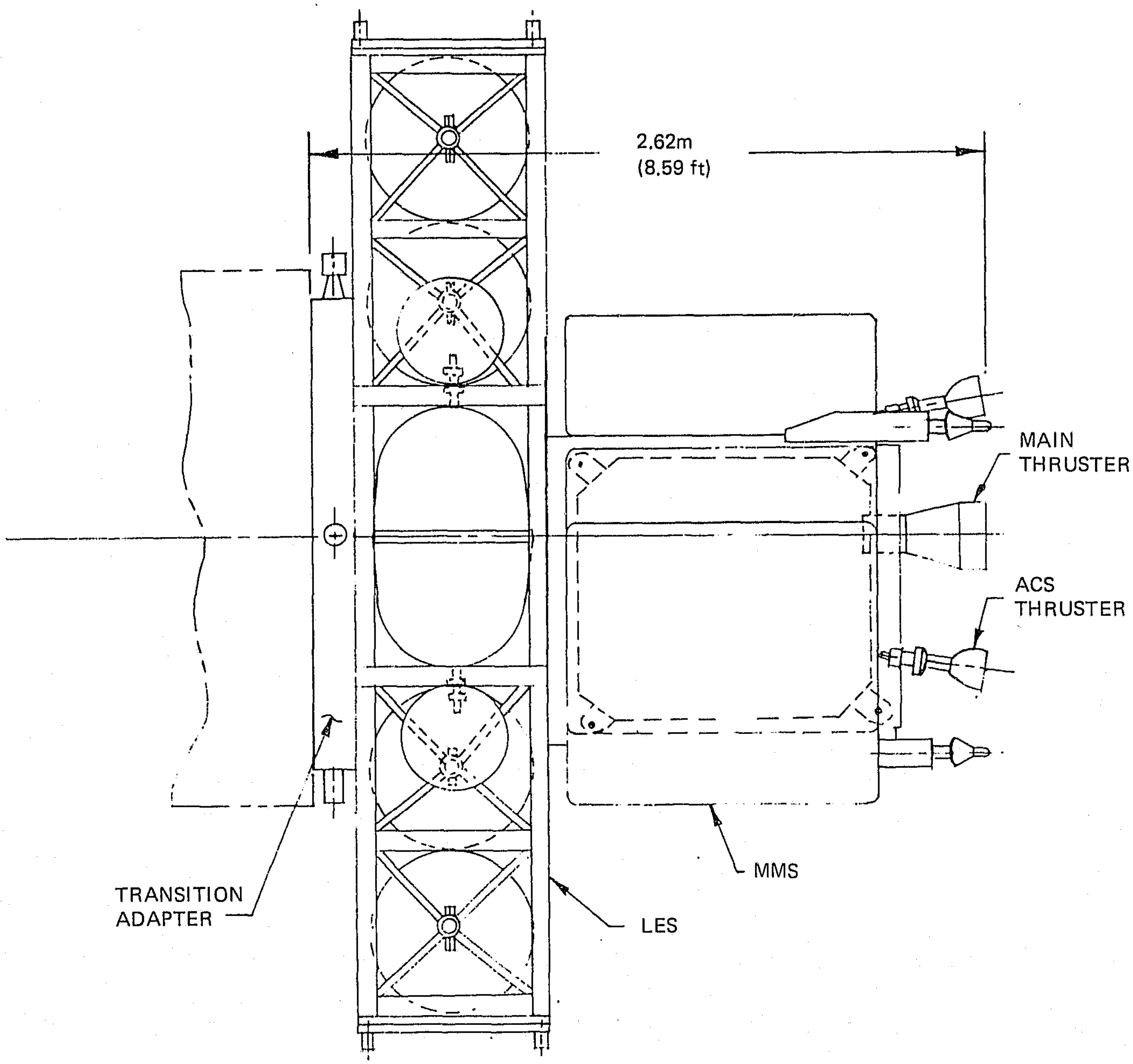


FIGURE 4.41 MMS/LES CONCEPT II

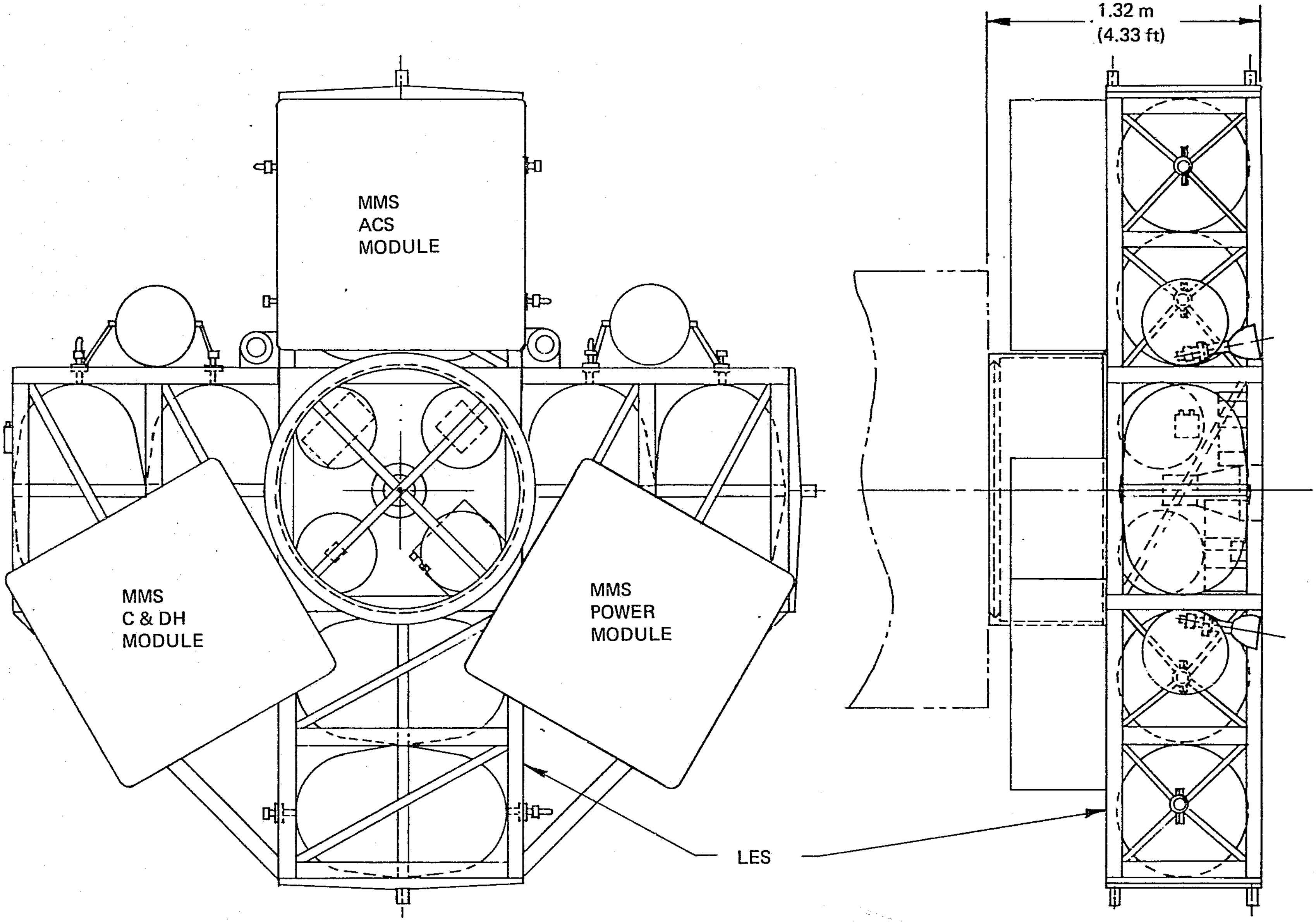


FIGURE 4.42 MMS MODULES/LES CONCEPT III

not be used with this conceptual configuration. However, the LES conceptual design launch cradle could be compatible assuming that the change in the mass distribution due to the removal of the LES subsystems and the addition of the MMS modules can be accomodated.

Each of the concepts shown was a low energy stage that can be used in either of two modes. First it can be used as a stage that performs all payload delivery functions independent of the payload, and second it can be used as a propulsion system with all the physical characteristics of the low energy stage except for those components whose functions were provided by the payload.

Table 4-XXXI shows a summary comparison of the Integral Propulsion System concepts.

4.9 IMPACT ON PAYLOAD DESIGN TRENDS

Payloads of the past have been sized and shaped by the space available on expendable launch vehicles (ELV). These ELV launched payloads traditionally have high length/diameter ratios. They fit the available space in the ELV and the volumetric efficiency is generally 90% or higher.

The advent of the Orbiter with its 18.288 m (60 ft.) long by 4.572 m (15 ft.) diameter cargo bay has relieved the ELV form factor constraints on payload designers and presents a completely new set of form factor criteria. The need for high volumetric efficiency in packaging payloads in the Orbiter cargo bay is made very real by the user charge policy. How well the space aboard each Orbiter flight is used will determine the efficiency of the Space Transportation System.

4.9.1 Measurement of Orbiter Cargo Bay Packaging Efficiency

The Orbiter cargo bay volume is 300.241 cubic meters (10,603 cubic ft.) and the allowable cargo weight for the standard 28.5° inclination launch from ETR is 29,484 kg (65,000 lb.) which yields an average density of 98.20 kg/m^3 (6.13 lb/ft^3) if the entire cargo bay were uniformly filled 100% with the allowable payload. A more useful guide for the payload designer is a weight/unit length figure of 1612 kg/m (1083 lb/ft) of cargo bay length for 28.5° inclination and 1414 kg/m (950 lb/ft) for the 56° inclination.

TABLE 4-XXXI INTEGRAL PROPULSION SYSTEM SUMMARY COMPARISON

CONCEPT	CONCEPT LENGTH (METERS/INCHES)	LENGTH SAVINGS OF CONCEPT VS MMS/PMII (METERS/INCHES)	WEIGHT SAVINGS RESULTING FROM REMOVAL OF REDUNDANT SUBSYSTEMS WEIGHT (KG/LBS)	MMS FLIGHT SUPPORT SYSTEM COMPATIBILITY	
				MMS	LES
MMS/LES CONCEPT I	2.565/101	0.48/19	54.2/119.5	No	Yes
MMS/LES CONCEPT II	2.61/103	0.43/17	54.2/119.5	Yes	No
MMS MODULES/LES CONCEPT III	1.32/52	1.727/68	54.3/119.5	No	Yes

For payloads launched from WTR the allowable Orbiter cargo weight is reduced to: 16,783 kg (37,000 lb) for 90° inclination orbit operations and to 14,969 kg (33,000 lb) for 98° inclination orbit operations. The corresponding weight/unit length figures are 918 kg/m (617 lb/ft) and 819 kg/m (550 lb/ft).

A review of the LES payload/mission model (Table 4-II) reveals that approximately one-third of LES missions will be launched from ETR and the remaining two-thirds from WTR. A payload designed for a WTR launch may be optimized in packaging efficiency if the volume and length relationship is adjusted until they match or approximate closely the user's charge factor due to weight. For example, using a packaging density of 160 kg/m^3 (10 lb/ft^3) average for current spacecraft industry practice, the payload of 1000 kg (2205 lb) mass destined for a 90° WTR launch works out to dimensions of 1.088 m (3.57 ft.) length and 2.704 m (8.87 ft.) diameter and an L/D ratio of .402. This is a new shape for spacecraft payload designers and requires a large diameter flat type of stage to keep the L/D ratio near the optimum and will guide the payload/stage/cradle installations in the Orbiter into the horizontally mounted direction with vertical ("pop-up") deployment. The trend makes spin stabilization by predeployment spinning in the cargo bay difficult and tends to favor a 3-axis stabilization method of guidance and control. To better understand the problems associated with spinning in the bay prior to deployment a study of the loss of cargo bay volume incurred by spinning payloads deployed in a vertical manner was performed.

Figure 4.43 shows the results of the study. The study showed that the most effective way to use the space of the cargo bay was to provide for horizontal installation of a payload sized to use the full diameter of the bay and to reduce the length of the payload/stage combination as much as possible. Direct mounting of the payload to the Orbiter longerons and keel with a flat, large diameter delivery stage cantilevered from the payload structure appears to be the best method for achieving high volumetric efficiency. This method will require deployment from the bay to be in a direction normal to the longitudinal axis of the cargo bay to avoid increased

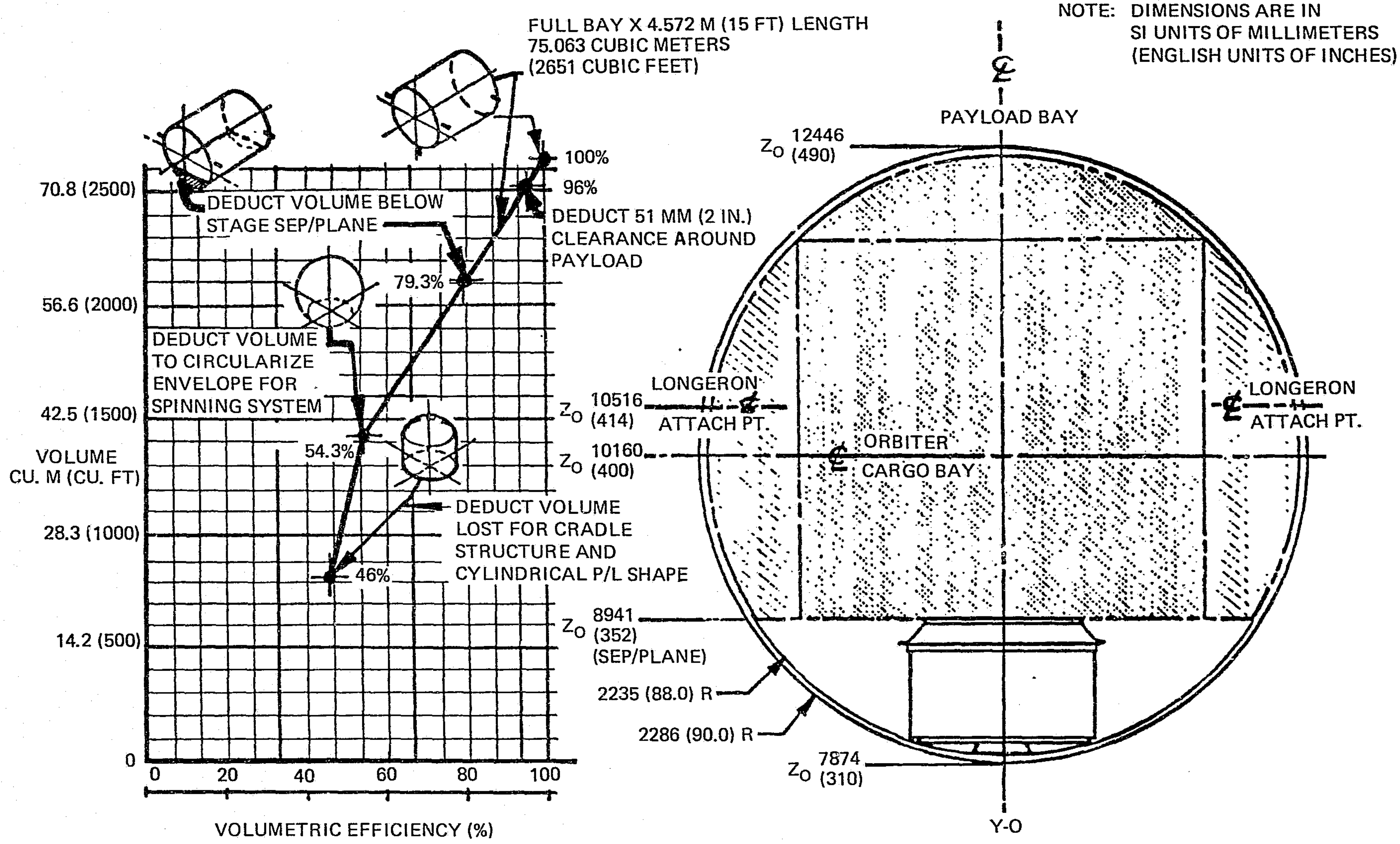


FIGURE 4.43 CARGO BAY PACKAGING VOLUMETRIC EFFICIENCY

user charge due to additional swing clearance for predeployment erection.

4.9.2 Effect of LES/ASE Cradle Designs on Payload Design Trends

Installation design studies during the early phases of Tasks 3 and 4 showed that the greatest volumetric efficiency for the payloads to be carried by the Orbiter could be obtained by the direct mounting method where the payload fills the bay and the stage is very flat and attached directly behind the payload. A review of the LES mission model (Table 4-II) shows that only 8 missions and 25 payloads (19% of the model) are larger in diameter than 4 meters (13.12 ft.) and are currently planned for direct mounting. Another group of large diameter payloads are those of 4 meter (13.12 ft.) to 3.6 meter (11.81 ft.) diameter). This group makes up 7 missions and 24 payloads (18% of the model). The majority of these large payloads are scheduled for launch in 1984 or later and appear to have been able to take advantage of the Orbiter bay form factor criteria to achieve good volumetric efficiency.

As the payload diameter becomes lower than about 3.5 meters (11.48 ft.) it was found structurally inefficient to direct mount the payload to the Orbiter. Some form of intermediate payload support such as an ASE cradle or a Spacelab type of pallet was required for cost effective installation. Further studies showed that the optimum limit for direct mounting of payloads occurred at about 4 meter (13.12 ft.) diameter. At this payload size limit an effective and lightweight ASE cradle could be utilized as a base for fillers and adapters to be added for accomodating smaller diameter payloads, but yet leave the basic cradle available for the expected larger diameter payloads. In order to support a multiplicity of payload lengths with the same basic cradle structure a design concept evolved that permitted varying the cradle length by using telescoping rod assemblies that would permit placement of the support frames at the payload c.g. and the stage c.g. and to use these rods as the reactive base for a deployment force at 90° to the Orbiter X-axis.

4.9.2.1 Effect on Payloads for Vertical Installation

Vertical payload installations using the LES/ASE vertical cradle are shown by Figure 5.7. Payloads would be required to furnish only the V-band coupling flange and their own electrical umbilical mounted on the starboard side of the payload in line with the payload c.g. plane.

The vertical cradle arrangement is a cost effective manner of mounting smaller payloads in the Orbiter cargo bay. Orbiter payloads should be installed vertically if the length of the spacecraft and stage exceeds the diameter of this combination and if the length of the spacecraft, stage, ASE cradle combination is less than the cargo bay diameter.

4.9.2.2 Effect on Payloads for Horizontal Installations

The horizontal payload installations developed during the LES study are shown by Figures 5.4, 5.5 and 5.6 for cradle mounted payloads and by Figure 5.2 for the direct mounted payloads. All cradle mounted payloads over 200 kg (441 lb) mass would be required to provide three mounting trunnions at their outer diameter at 3, 6 and 9 o'clock locations in the plane of their center of gravity (c.g.). These payloads would also furnish mating deployment socket fittings (two fittings for payloads between 200-2000 kg (441-4410 lb) and three fittings for payloads heavier than 2000 kg (4410 lb) at approximately the 5 and 7 o'clock positions at a station plane that will pass through the combined payload/stage center of gravity. The remaining payload furnished interfaces required consist of the payload half of the V-band type separation coupling flange and the payload electrical umbilical at the 2:30 o'clock location in the plane of the payload c.g. and mounted at the payload outer diameter. Should the payload require some electrical service from the LES, an electrical umbilical would be provided by the payload at the 1 o'clock location and just inside the V-band coupling flange.

5.0 TASK 4: INTERFACE ANALYSIS

The purpose of this task was to develop preliminary conceptual designs for Airborne Support Equipment (ASE) for the selected propulsion modes of Task 3 and to provide physical and electrical interface definitions for these propulsion modes between the spacecraft and the ASE and between the ASE and the Orbiter. The developed concepts were evaluated to determine the impact on the elements of total launch cost for individual combinations of payloads, stages and ASE using the revised low energy payload model (paragraph 4.1).

5.1 AIRBORNE SUPPORT EQUIPMENT CONCEPTUAL DESIGN

A conceptual design data base was established for ASE from references 24 through 39. In addition, existing or planned ASE cradle concepts were examined for compatibility and adaptability to conceptual LES. Data were then collected on ASE weight, size and length required for installation and payload deployment from the Orbiter cargo bay. Table 5-I summarizes this data and Figure 5.1 shows the general appearance of the more applicable systems studied.

The larger twelve and eight tank versions of the LES would not adapt to existing/planned ASE cradles without extensive redesign of either the stage or the cradle. The smaller four and two tank versions of LES were adaptable to TRS and MMS cradle concepts, however, additional components would be required to provide support of the LES and cantilevered payloads. Table 5-II lists existing/planned ASE cradle assemblies and the percentage of payloads of the LES payload model adaptable to them. Since only approximately 50% of the LES mission model payloads could be captured by any of the existing/planned cradle systems a modular cradle concept was developed.

The results of this evaluation were used to establish the following groundrules.

- ASE cradles should not add length or width to the stage/payload combination in such a manner that the cargo bay length user's charge increases.
- ASE cradle weight should be minimized consistent with supporting the mission payload/stage combinations.

TABLE 5-I APPLICABLE EXISTING OR PLANNED ASE CRADLE DESIGNS

STAGE/CARRIER	TOTAL ASE WEIGHT kg (lb)	CRADLE SIZE LENGTH-m/DIA-m (LENGTH-ft/DIA-ft)	PAYLOAD WEIGHT LIMIT kg (lb)	PAYLOAD SIZE LIMIT LENGTH-m/DIA-m (LENGTH-ft/DIA-ft)	ORBITER BAY LENGTH REQUIRED (1) m (ft)
MMS/PM-II EXPENDABLE	1905 (4200)	2.130/4.52 (7.00/14.83)	(2) 3629 (8000)	(3) 5.200/1.957 (4) (17.06/6.42)	3.378 (11.08)
TRS - 2 TANK RETRIEVABLE	1302 (2870)	2.230/4.52 (7.33/14.83)	(5)	(5)	2.23 (7.33)
TRS - 4 TANK RETRIEVABLE	1302 (2870)	2.230/4.52 (7.33/14.83)	(5)	(5)	2.23 (7.33)
SSUS-A (HORIZONTAL)	1724 (3800)	2.603/4.52 (8.54/14.83)	1996 (4400)	6.261/4.52 (20.54/14.83)	2.731 (8.96)
SSUS-D (HORIZONTAL)	1724 (3800)	2.603/4.52 (8.54/14.83)	1087 (2400)	6.261/4.52 (20.54/14.83)	2.731 (8.96)
SSUS-D (VERTICAL)	1021 (2250)	2.185/4.52 (7.17/14.83)	1087 (2400)	2.567/2.185 (8.42/7.17)	2.185 (7.17)
SPACELAB (6) PALLET	653 (1440)	2.875/4.46 (9.43/14.63)	3500 (7716)	2.875/3.63 (9.43/11.91)	2.875 (9.43)

NOTE:
(1) Total of stage + ASE + swing clearance for erection prior to deployment.
(2) Design payload maximum weight for FSS for MMS.
(3) Longest of presently planned MMS payloads minus length of MMS.
(4) Maximum diameter over mounting trunnions for present design of Flight Support System (FSS) for MMS.
(5) TRS/SD ASE cradle makes no provision to carry payload. Payload must provide its own mount for use in Orbiter.
(6) Data is for one Spacelab pallet. Multiply by 2 for two pallets and by three for three pallets.

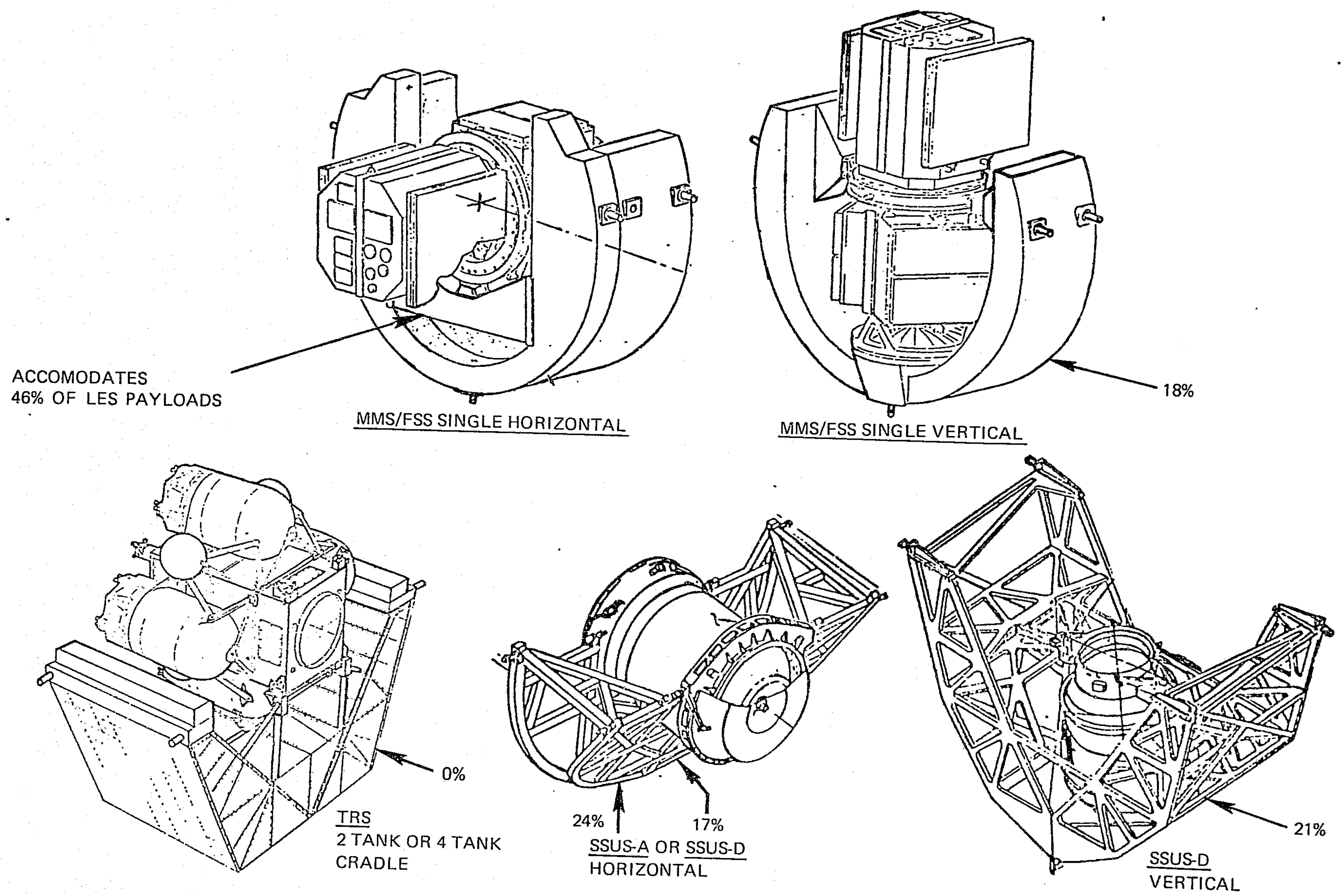


FIGURE 5.1 GENERAL ARRANGEMENT OF SOME ASE CRADLE SYSTEMS

TABLE 5-II

POTENTIAL OF ADAPTING EXISTING/PLANNED ASE TO NEW LES

EXISTING CRADLE ASSEMBLIES	CRADLE ARRANGEMENT	PERCENTAGE OF LES PAYLOADS ACCOMMODATED
MMS/PM-II	VERTICAL	18
MMS/PM-II	HORIZONTAL	46
TRS - 2 OR 4 TANK	HORIZONTAL	0
SSUS-A	HORIZONTAL	24
SSUS-A	19° PRE-TILTED	24
SSUS-D	HORIZONTAL	17
SSUS-D	43° PRE-TILTED	17
SSUS-D	VERTICAL	21

- ASE cradle designs should be capable of easy reconfiguration.
- ASE cradle should be modular so that horizontal and vertical installations can be carried and launched safely with a single set of ASE cradle components.
- ASE cradle should be designed to permit payload/stage deployment without increasing Orbiter cargo bay length requirement.
- Deployment should be at a velocity of 0.3 meter/sec (1.0 ft/sec) to 0.6 meter/sec (2.0 ft/sec) with a maximum tip-off angle of approximately 1° plus an allowance of 0.1° additional for the design clearance envelope.
- ASE cradle designs should use as many components as possible from developed or planned ASE cradles. Especially desired are cost savings through increased use of the remote controlled latching mechanisms used on the Orbiter and the MMS/FSS.

5.1.1 Design Constraints

LES mission model ASE cradle assembly design constraints were identified and defined using the above groundrules. In addition to these groundrules the following design constraints were used for ASE conceptual design:

- ASE shall be modular in design and capable of configuration flexibility by various arrangements of interchangeable parts.
- A 3-axis stabilized guidance system is used for all stages.
- Stage propulsion systems incorporate prepackaged sealed storable propellant tanks. Dumping provisions are not required since the tanks will be design qualified.

- Largest stage the ASE cradle must accommodate horizontally is an eight tank stage of approximately 1.0 meter (3.28 ft.) length by 4.0 m (13.12 ft.) diameter arranged in a flat cross. Largest stage the ASE cradle must accommodate vertically is a four to twelve tank stage of approximately 1.5 m (4.92 ft.) length by 4.0 m (13.12 ft.) width by 2.5 m (8.20 ft.) height arranged in a fore and aft axis stacked arrangement.
- Deployment from the Orbiter cargo bay shall be by ASE cradle contained mechanical means with a deployment velocity of 0.305 mps (1 fps) minimum and 0.610 mps (2fps) nominal for design. Tipoff angle at deployment shall be 1.1° maximum. Deployment envelope shall not increase Orbiter cargo bay length requirements, however, during deployment the 0.076 m (3 in.) clearance at each end may be used as part of the deployment envelope.
- ASE cradle installation shall have alignment repeatability of 0.50° (3 sigma) error with each of the Orbiter's three major axes.
- ASE cradle shall make maximum use of existing/standard components such as Orbiter type interface trunnions, manual or remote operation latching mechanisms and remote controlled latching mechanisms used on the MMS/FSS.
- For adaptation stages, use SSUS-A or SSUS-D horizontal cradles for adaptations of SSUS-A or SSUS-D as horizontal booster stage and SSUS-D vertical cradle for adaptations of the SSUS-D as a vertical booster stage.

5.1.2 Modular Cradle Concepts

The design approach selected for an ASE cradle assembly is a modular concept using the fewest number of components to achieve the required configurations. This concept provides the capability to handle all but the largest of the planned LES payloads.

A 4.0 m (13.12 ft.) payload envelope positioned at the top of the cargo bay leaves adequate space for cradle structure. Payloads larger than 4.0 m (13.12 ft.) in diameter are considered for direct mounting to the Orbiter structure through the use of small fittings tied directly to the payload structure. The stage (in those cases requiring a stage) is mounted in a cantilevered manner to the aft end structure of the payload through use of a special adapter, bolted onto the payload structure, which is of minimum length and weight. Figure 5.2 illustrates this arrangement and is typical of how the larger than 4.0 m (13.12 ft.) diameter payloads can be handled for Shuttle transportation to orbit for deployment. Separation from the Orbiter prior to perigee burn would be by Orbiter maneuver after stage and payload deployment by the Remote Manipulator System (RMS).

The primary design objectives established for the cradle concepts are:

- The cradle configurations achieveable with the components of a cradle assembly provide handling capability for all of the LES payloads that are equal to or less than:
  - 9.0 meters (29.53 ft.) in length
  - 4.0 meters (13.12 ft.) in diameter
  - 4500 kilograms (9,921 pounds) in mass
- The cradle assembly design minimizes length to avoid adding to the user's charge.
- The design of the major cradle assembly components provides two strength levels; one for heavy payload/stage combinations and one for light combinations. These two levels tailor cradle configuration mass to better match payload/stage mass; thus, reducing weight impact for achieving modularity.

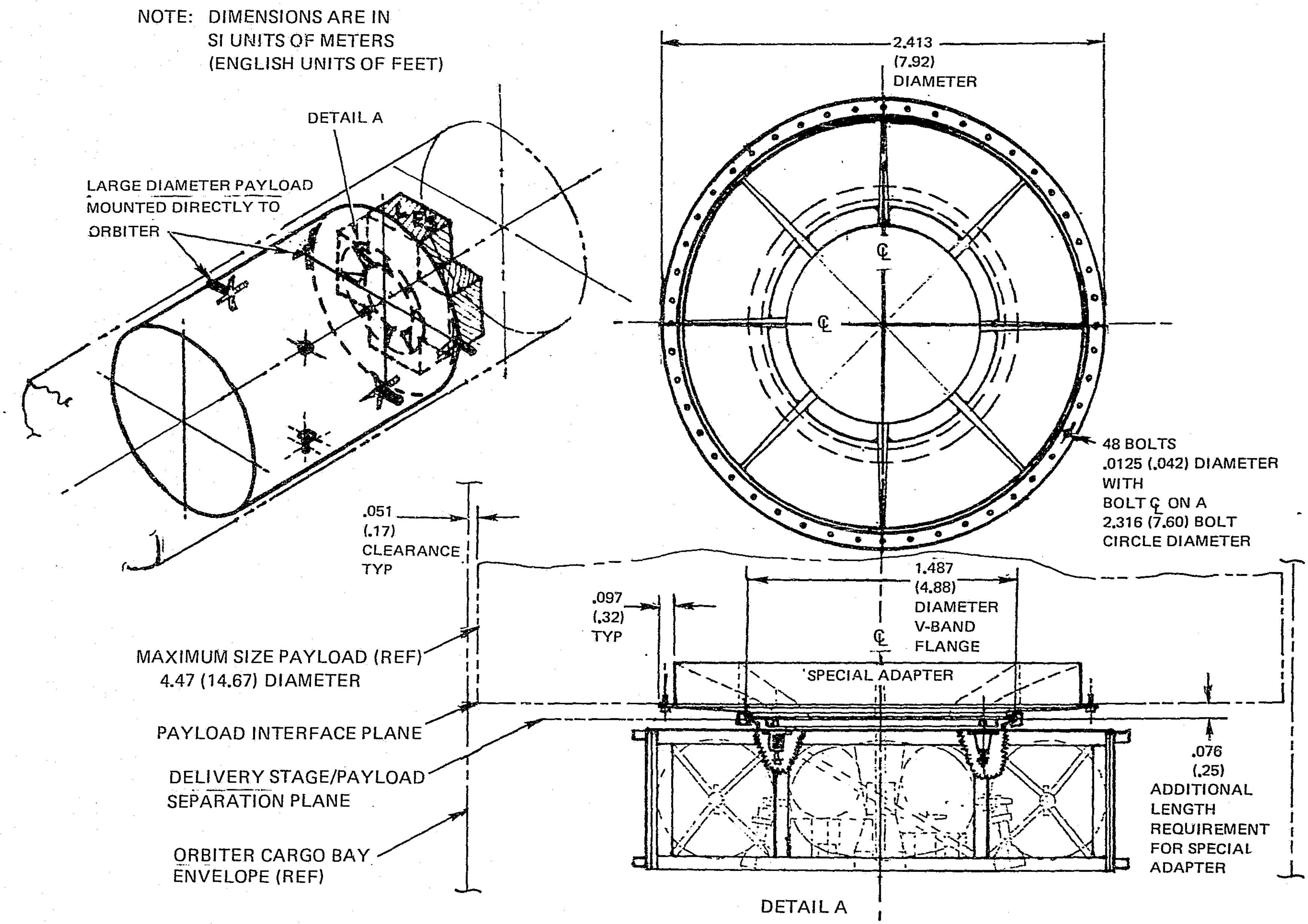


FIGURE 5.2 INSTALLATION OF LARGER THAN 4 METER (13.12 FT) DIAMETER PAYLOAD

- o Maximum use of existing or standard components such as remotely operated latching mechanisms from the MMS/FSS and STS Orbiter programs provides reduction in cost.
- o Payload/stage separation from the cradle and deployment from the Orbiter cargo bay is directed along a path parallel to the Orbiter's +Z axis.

5.1.2.1 <u>LES ASE Cradle Set Definition</u> - A composite envelope of the LES mission model payloads which are equal to or less than the design objectives of 9.0 m (29.53 ft.) length, 4.0 m (13.12 ft.) diameter and 4500 kg (9,921 lb.) mass is shown in Figure 5.3. The larger sizes of this envelope are installed horizontally in the Orbiter cargo bay. Approximately 24 of the mission model payloads with dimensions equal to or less than 2.9 m (9.5 ft.) in length, 1.5 m (4.92 ft.) diameter and 907 kg (2,000 lb.) mass can be installed vertically in the Orbiter cargo bay.

The LES ASE cradle assembly consists of a set of components which are arranged in various configurations to provide four cradle assemblies, individually but not simultaneously, from the parts of the set. Table 5-III is a list of all modular components which make up a cradle assembly set and indicates by item number and quantity required how each component is used. The lighter weight 0.254 m (0.83 ft.) wide cradle I is used for the short horizontal and vertical cradle configurations to save weight because of the generally lower mass and size of these payload/stage combinations.

The conceptual arrangement of the modular components required to handle a small payload carried horizontally is shown in Figure 5.4. In this configuration cradle assembly #1 provides support of the stage configured for horizontal carriage and deployment. In addition, support is provided in a cantilevered manner for a maximum payload size and weight of 1.8 m (5.9 ft.) long by 1.4 m (4.6 ft.) diameter and a mass of 200 kg (441 lb.). The baseline cradle assembly used is for the 8-tank LES. To convert the cradle to handle a 4 tank LES the add-on accessories identified as item numbers 4, 7, 8 and 19 are utilized to fill the void created between the cradle mounting points and the smaller sized stage.

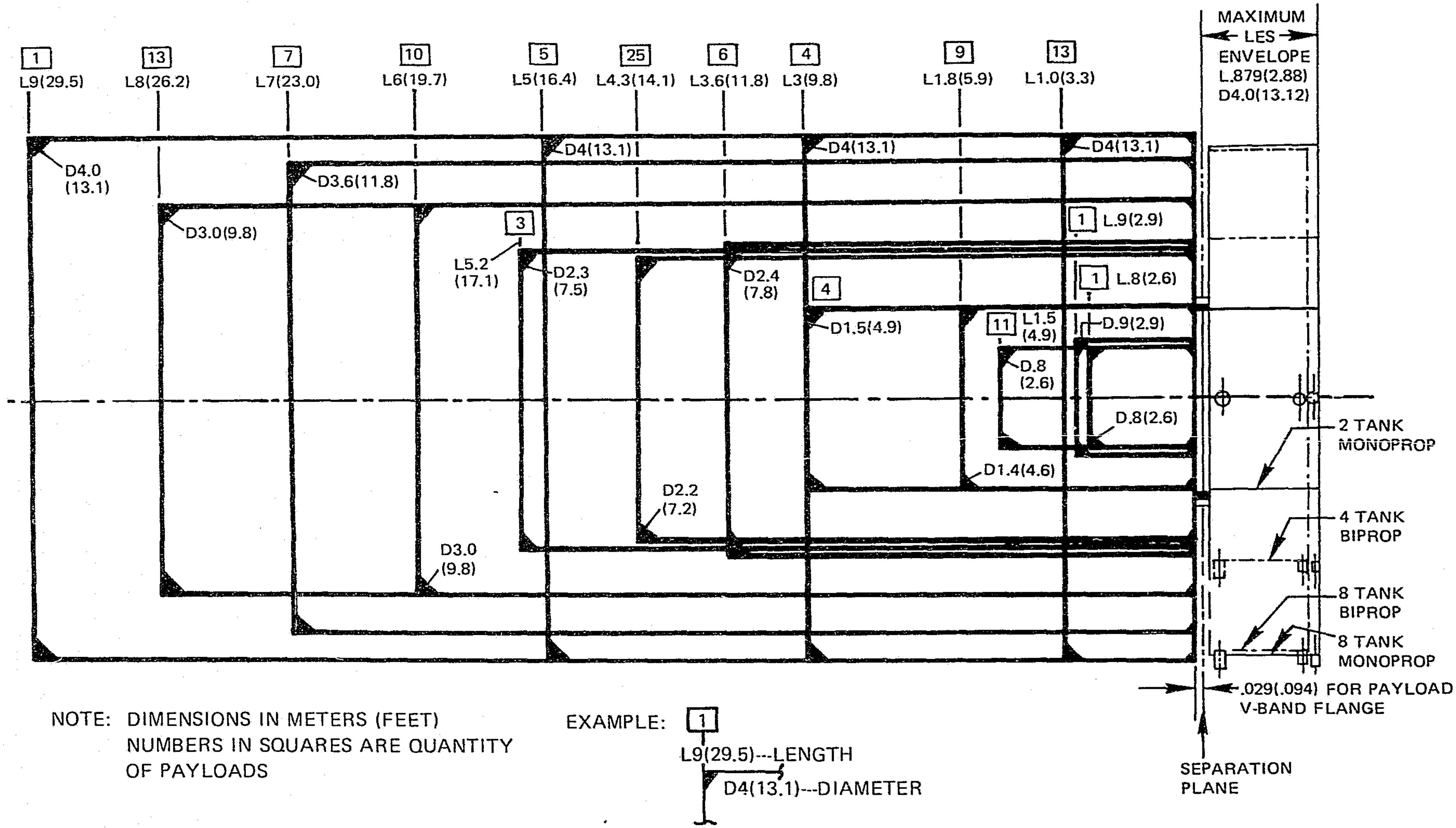


FIGURE 5.3 COMPOSITE ENVELOPE OF LES PAYLOADS AND STAGES

TABLE 5-III LES ASE CRADLE ASSEMBLIES PARTS LIST

PART NO.	PART DESCRIPTION	QUANTITY PER SET	CRADLE ASSY. #1 SMALL HORIZONTAL QUANT/ASSY	CRADLE ASSY. #2 MEDIUM HORIZONTAL QUANT/ASSY	CRADLE ASSY. #3 LARGE HORIZONTAL QUANT/ASSY	CRADLE ASSY. #4 VERTICAL QUANT/ASSY.
1	Cradle I (10" wide)	2	1	2		2
2	Cradle II (14" wide)	2	1	2	2	
3	Walking Beam (14" wide)	1		1	1	
4	Cradle Filler	1	1	1		
5	Cradle Extender - LH	1	1	1	1	
6	Cradle Extender - RH	1	1	1	1	
7	Filler Adapter - LH	1	1	1		
8	Filler Adapter - RH	1	1	1		
9	Filler Adapter - Bottom	1	1	1	1	
10	Deploy Mechanism Base	2	2			
11	Rod Assy - Adjustable 60" to 100"	4	4	(4)	(4)	
12	Rod Assy - Fixed 38"	4			(4)	4
13	Rod Coupler - Fixed 20" Length	4			(4)	
14	Rod Assy - Adjustable 12" to 20"	4		(4)		
15	Rod Assy - Adjustable 30" to 36"	4		(4)		
16	Rod Assy - Adjustable 36" to 60"	4		(4)	(4)	
17	Latch Mech. - Type A (MMS/FSS Type)	8	5	8	6	5
18	Latch Mech. - Type B (Orb Deployable Type)	3			2	
19	Latch Spacer	6	5	6		
20	Longeron Trunnion	4	4	4	4	4
21	Keel Trunnion	2	1	2	2	1
22	Deploy Mechanism #1 - Mechanical Spring Pkg	2	2			2
23	Deploy Mechanism #2 - Multiple Spring Pkg	1		1		
24	Deploy Mechanism #3 - Electro/Mech Mechanism	1			1	
25	Deploy Mechanism - Base-Fixed	1				1
26	Deploy Mechanism - Base-Moveable	1			1	
27	Walking Beam - Pivot Assembly	1		1	1	
28	Latch Shim - Type A	5		3	6	5
29	Latch Shim - Type B	3			2	
30	Vertical Launch - Walking Beam - Assy.	2				2
31	Filler Adapter - Side	4				4
32	Filler Adapter - Bottom	1				1
33	Cross Brace Assembly	4				4
	TOTALS	86	31	38	42	35

NOTE: ( ) OPTIONAL DEPENDING ON PAYLOAD LENGTH

CRADLE ASSEMBLY #1

STAGE	TOTAL ASE WEIGHT
BIPROPELLANT:	
4 TANK	1221 (2692)
8 TANK	1034 (2279)
MONOPROPELLANT:	
2 TANK	1221 (2692)
8 TANK	1034 (2279)

NOTE: DIMENSIONS IN MILLIMETERS (INCHES)
WEIGHTS IN KILOGRAMS (POUNDS)

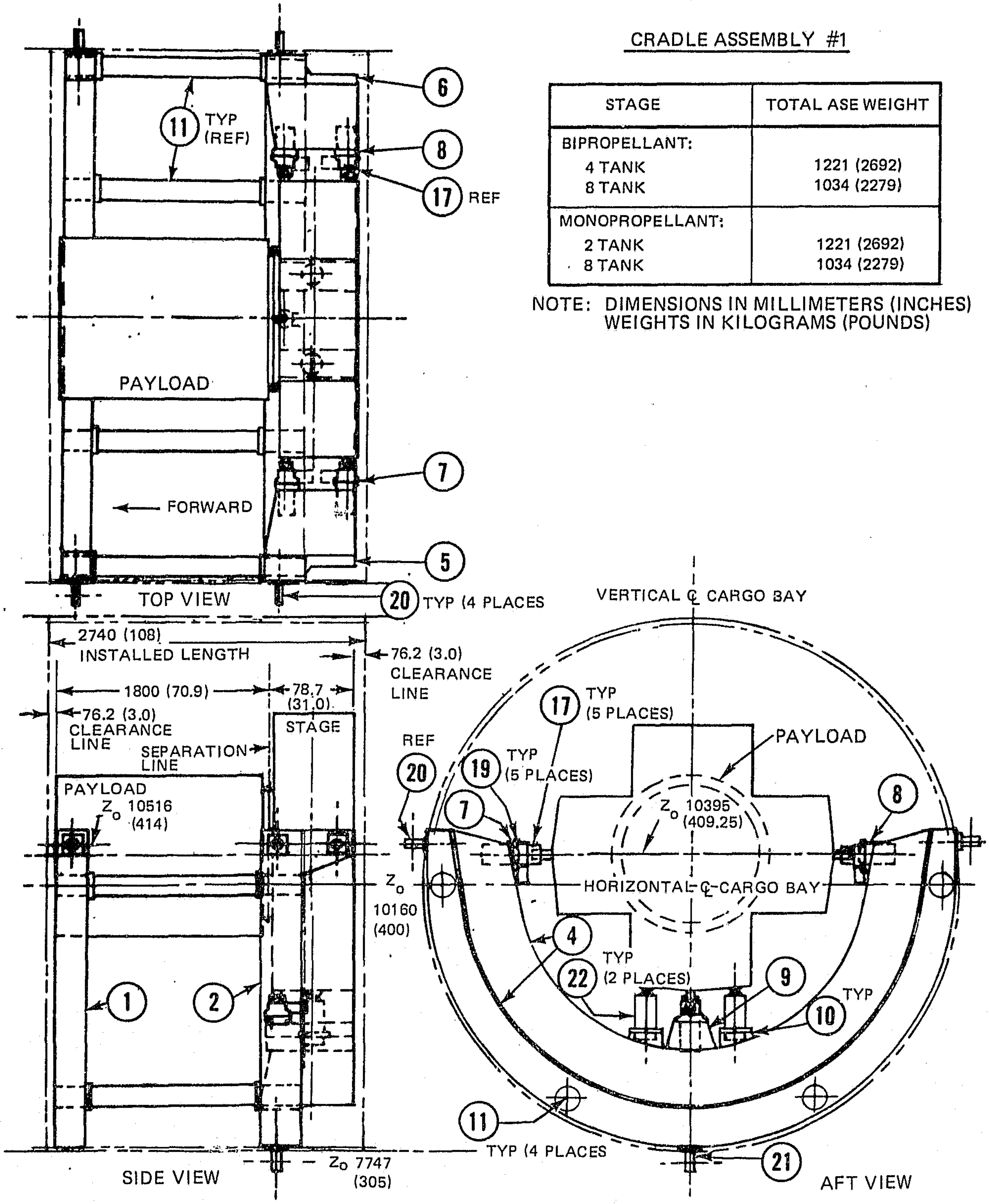


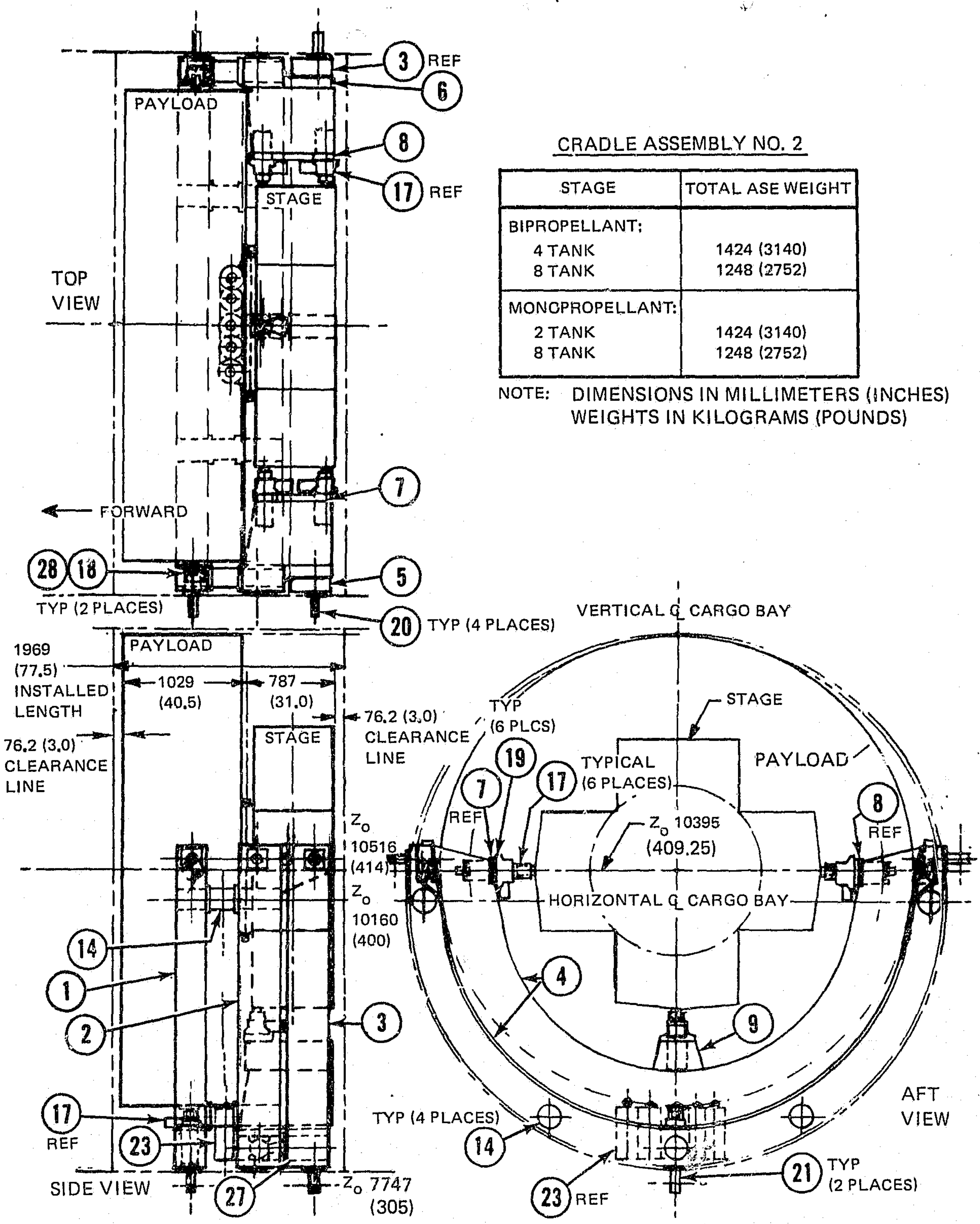
FIGURE 5.4 SMALL HORIZONTAL LES/PAYLOAD INSTALLATION

Cradle assembly #2 for a medium size payload in the horizontal carriage position is shown in Figure 5.5. In this configuration the widest of the cradles (item number 2) is placed under either the stage or the payload, whichever has the greater mass. By assigning a payload maximum weight limit of 2400 kg (5291 lb.) for this cradle configuration the MMS/FSS latching mechanism can be used for support of payloads. Again the add-on accessories are used to accommodate the 4 tank LES. The maximum size of payload this cradle will accommodate is 5.4 m (17.72 ft.) in length and 4.0 m (13.12 ft.) in diameter. This cradle assembly is the configuration used most frequently, consequently it uses the greatest number of adjustable rod assemblies. Rod assembly settings for this cradle vary from 0.304 m (1 ft.) up to 2.032 m (6.67 ft.).

The larger size payloads are accommodated by cradle assembly #3, shown in Figure 5.6. For this configuration the payload is much heavier than the stage and uses the widest of the support cradles at its center of gravity. The longest payload of 9 m (29.5 ft.) requires the use of rod assemblies with a length setting of approximately 3.48 m (11.42 ft.) which is beyond the capability of the largest of the four adjustable rod assemblies. This configuration requires the use of the rod coupler (item number 13) and fixed rod assembly (item number 12) to extend the rod assembly length. A moveable base (item number 26) is used to support the deployment mechanism (item number 24) which is a scissors jack type device used to deploy large payloads.

The conceptual arrangement of modular components required to achieve a vertical four tank stage cradle configuration (cradle assembly #4) is shown in Figure 5.7. Other arrangements of this cradle configuration will handle other vertical LES configurations. The installation dimensions and weights for these alternate vertical cradle arrangements are noted in Figure 5.7.

The vertical cradle arrangement is a cost effective manner of mounting smaller payloads in the Orbiter cargo bay. Orbiter payloads are installed vertically if the length of the spacecraft and stage exceeds the diameter of this combination and if the length of the spacecraft, stage, ASE cradle combination is less than the cargo bay diameter.

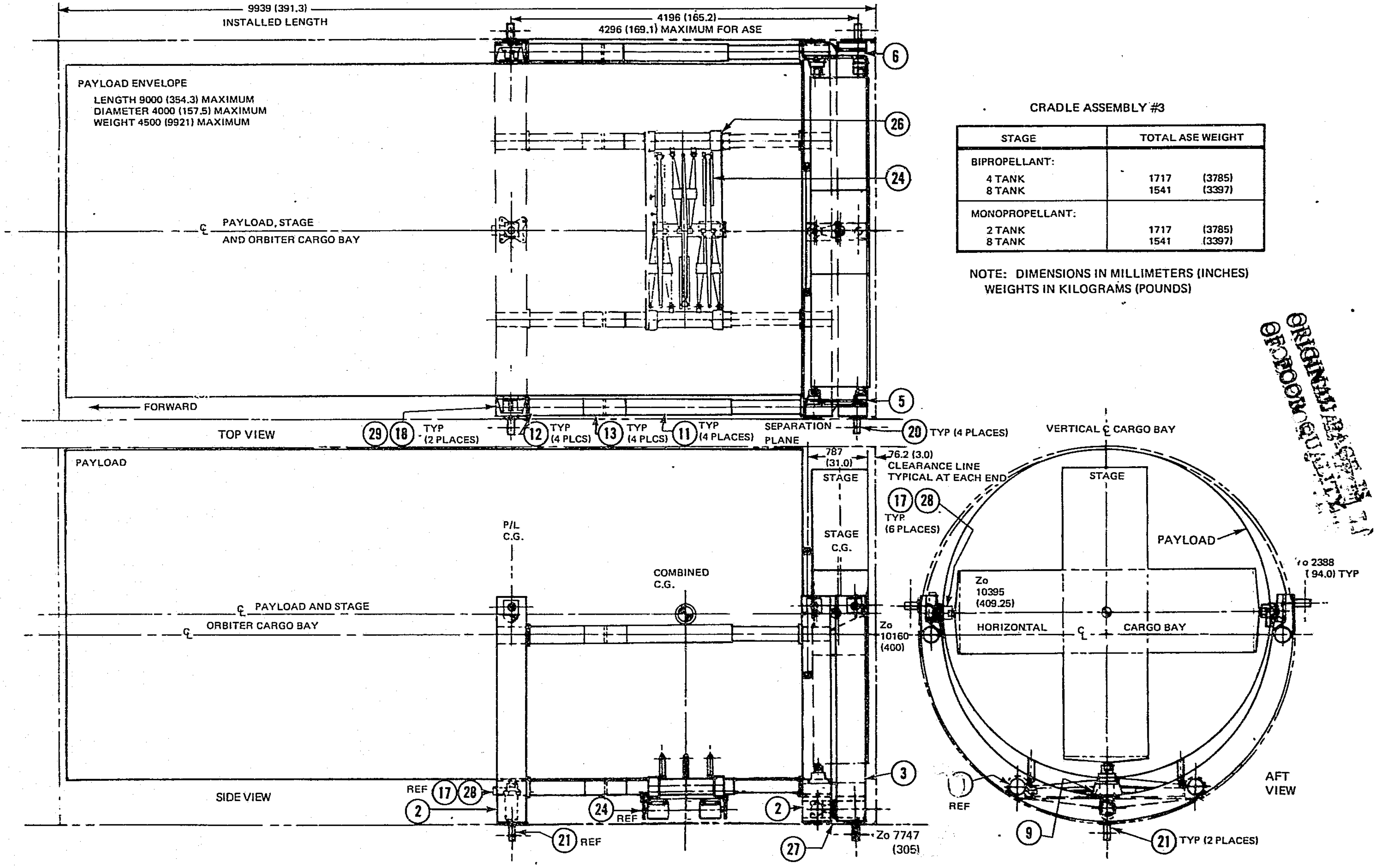


CRADLE ASSEMBLY NO. 2

STAGE	TOTAL ASE WEIGHT
BIPROPELLANT:	
4 TANK	1424 (3140)
8 TANK	1248 (2752)
MONOPROPELLANT:	
2 TANK	1424 (3140)
8 TANK	1248 (2752)

NOTE: DIMENSIONS IN MILLIMETERS (INCHES)
WEIGHTS IN KILOGRAMS (POUNDS)

FIGURE 5.5 MEDIUM HORIZONTAL LES/PAYLOAD INSTALLATION



CRADLE ASSEMBLY #3

STAGE	TOTAL ASE WEIGHT	
BIPROPELLANT:		
4 TANK	1717	(3785)
8 TANK	1541	(3397)
MONOPROPELLANT:		
2 TANK	1717	(3785)
8 TANK	1541	(3397)

NOTE: DIMENSIONS IN MILLIMETERS (INCHES)
WEIGHTS IN KILOGRAMS (POUNDS)

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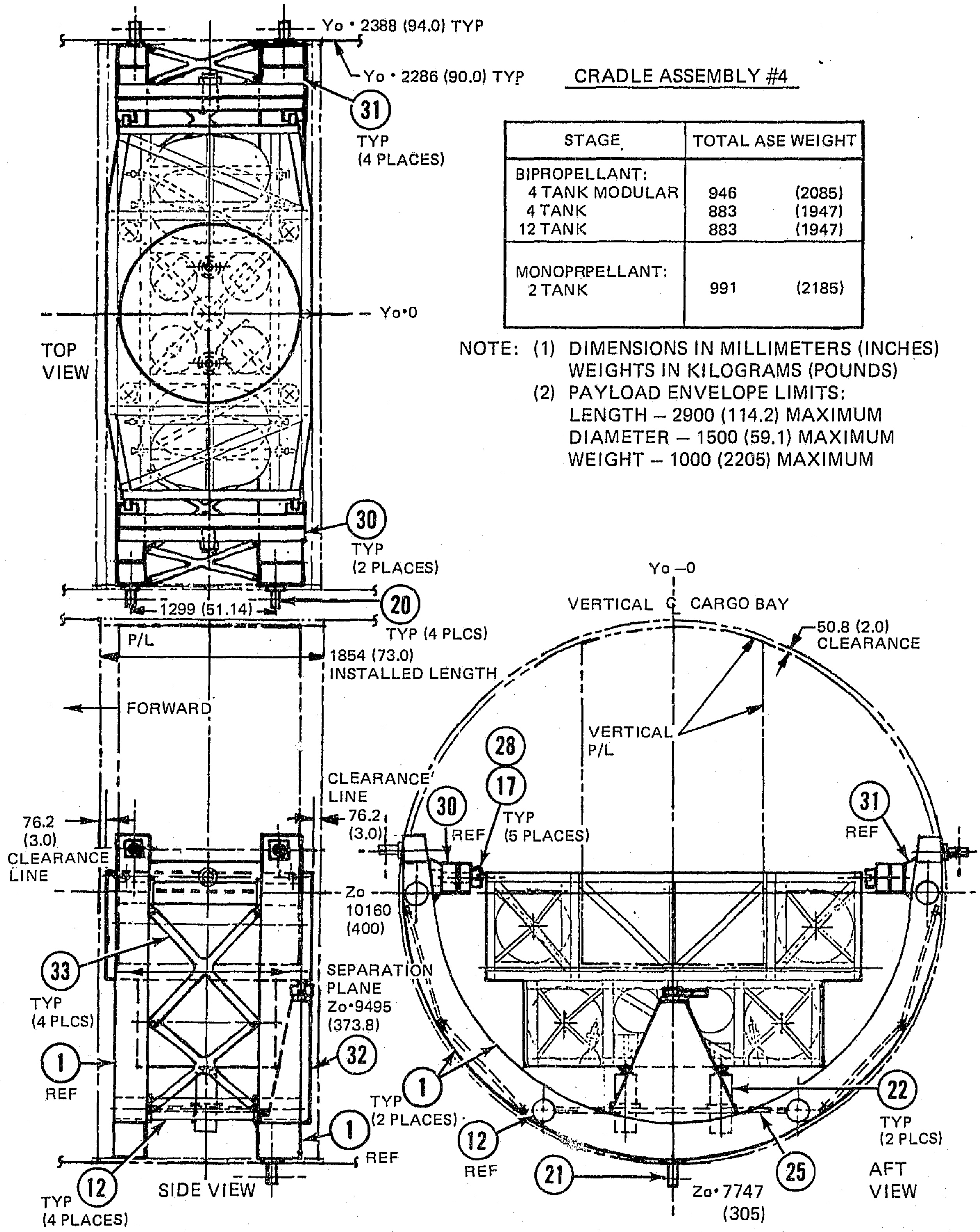
FIGURE 5.6 LARGE HORIZONTAL LES/PAYLOAD INSTALLATION



CRADLE ASSEMBLY #4

STAGE	TOTAL ASE WEIGHT	
BIPROPELLANT:		
4 TANK MODULAR	946	(2085)
4 TANK	883	(1947)
12 TANK	883	(1947)
MONOPRPELLANT:		
2 TANK	991	(2185)

NOTE: (1) DIMENSIONS IN MILLIMETERS (INCHES)
WEIGHTS IN KILOGRAMS (POUNDS)
(2) PAYLOAD ENVELOPE LIMITS:
LENGTH – 2900 (114.2) MAXIMUM
DIAMETER – 1500 (59.1) MAXIMUM
WEIGHT – 1000 (2205) MAXIMUM

FIGURE 5.7 VERTICAL LES/PAYLOAD INSTALLATION

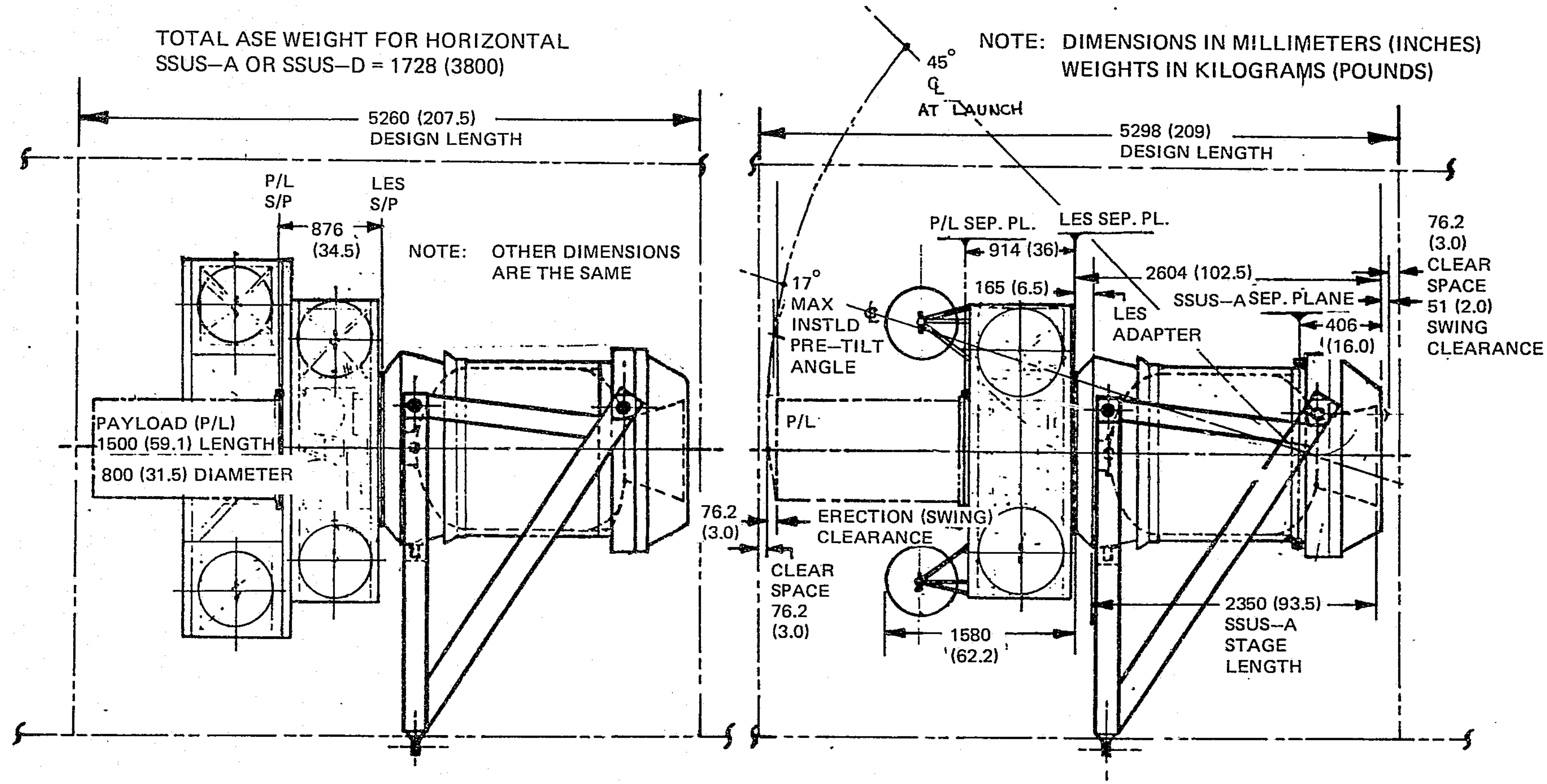
Deployment of all of the LES cradle mounted payload/stage combinations from the Orbiter is provided by mechanical or mechanical/electrical devices. The smaller payloads of up to 2000 kg (4409 lb.) mass are deployed by separation springs selected to match the payload/stage mass. The remaining large payload/stage combinations are deployed by means of a scissors jack type of deploy mechanism driven by two redundant electrical motors through suitable gear boxes. In both cases the force is directed through the combined center of gravity of the payload and stage to eject the Orbiter payload in the +Z axis direction.

5.1.2.2 SSUS Cradle Assemblies Adapted to LES - Figures 5.8 and 5.9 illustrate cradles for adaptations of SSUS and low energy stages for horizontal and vertical arrangements. Both cradle adaptations are the results of an analysis performed to physically integrate the combined mass of payloads and LES to adapt to the SSUS cradle assemblies. The mass combinations investigated were less than the published design capability of the documented SSUS-A and SSUS-D cradle assemblies as defined in References 33 and 39 respectively.

Typical modifications required to the SSUS cradles are: (1) electrically deactivating the spin table functions - deployment would utilize the existing mode of spring separation and (2) bypassing the spin-up requirements through procedural changes.

5.1.2.3 Interface Requirements Definition - The mechanical interface requirements for the LES and cradle assembly configurations are as follows:

- Orbiter to Cradle Assembly - The standard interface attachment points and hardware selected for the cradle assembly mate with the attachment latches and fittings planned for the STS Orbiter as defined in Reference 24.
- Cradle Assembly to LES - The standard interface attachment selected for use between the ASE cradle and the low energy stages is the remote control latching mechanism being developed for the MMS/FSS. Requirement varies from five for the smaller Orbiter payloads up to eight for the medium sized



4 TANK MODULAR BI–PROPELLANT LES/SSUS–A
MOUNTED IN SSUS–A CRADLE

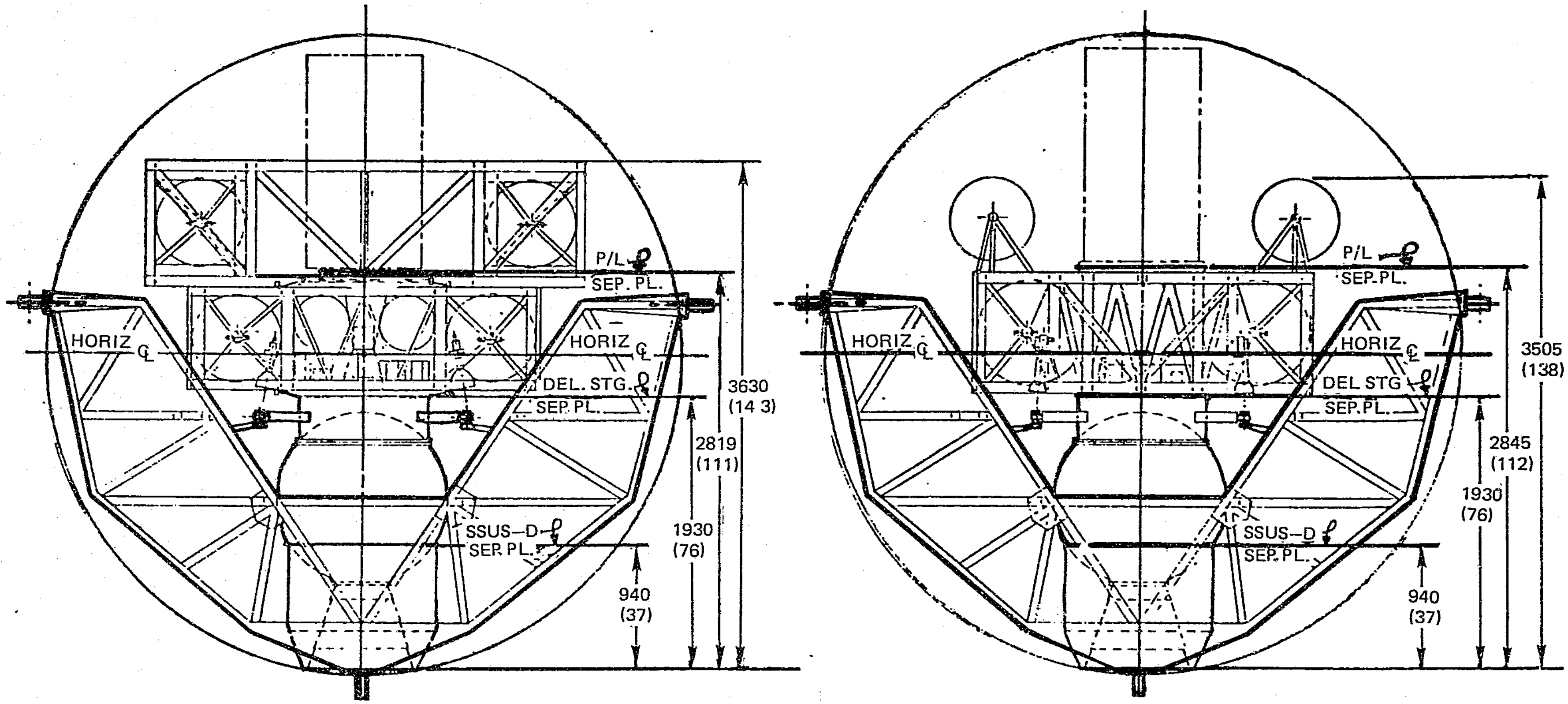
2 TANK MODULAR MONOPROPELLANT LES/SSUS–A
MOUNTED IN SSUS–A CRADLE

FIGURE 5.8 HORIZONTAL LES/SSUS–A ADAPTATION INSTALLATIONS

TOTAL ASE WEIGHT FOR VERTICAL
SSUS–D = 1021 (2250)

NOTE: DIMENSIONS IN MILLIMETERS (INCHES)
WEIGHTS IN KILOGRAMS (POUNDS)

NOTE: SSUS–D VERTICAL CRADLE
REQUIRES: 2184 + 152 CLEARANCE = 2336 OF CARGO BAY LENGTH
(86) + (6) CLEARANCE = (92) OF CARGO BAY LENGTH



4 TANK MODULAR BIPROPELLANT LES/SSUS–D
MOUNTED IN SSUS–D CRADLE

2 TANK MODULAR MONOPROPELLANT LES/SSUS–D
MOUNTED IN SSUS–D CRADLE

FIGURE 5.9 VERTICAL LES/SSUS–D ADAPTATION INSTALLATIONS

payloads. The large Orbiter payloads (over 2400 kg (5291 lb) mass) that use the cradle assembly require five of the MMS/FSS remotely operated latches and three Orbiter deployable type latches.

5.1.3 Design Features

The LES cradle assembly design is modular in construction. Four cradle arrangements that accommodate the LES mission model payloads are assembled from selected components of the cradle assembly set. Other design features which offer cost effective solutions to cradle assemblies are:

- Cradle assembly can be configured for either horizontal or vertical installation depending on the size of the payload and stage.
- Cradle assembly requires no additional Orbiter cargo bay length other than that required to install the payload and the stage.
- Cradle assembly configurations for LES are light weight and short in length.
- Cradles use standard STS latches and support trunnions which are removable and interchangeable with other components of the cradle assembly set. Orbiter deployable type latches are used for cradle support for large payloads. MMS/FSS type deployable latches are used for LES and for intermediate and smaller payloads.
- Cradle assemblies are designed for compatibility with the Orbiter timeline allocations and facilities usage.

5.1.4 Structural and Mechanical Interface

Eight liquid propellant low energy stage configurations, as defined in paragraph 4.5, are supported by four modular cradle assembly configurations. Typical arrangements are shown schematically in Figure 5.10. In the modular cradle approach, separate components of fixed and telescoping tubes are attached to box structure U-shaped frames to make four cradle

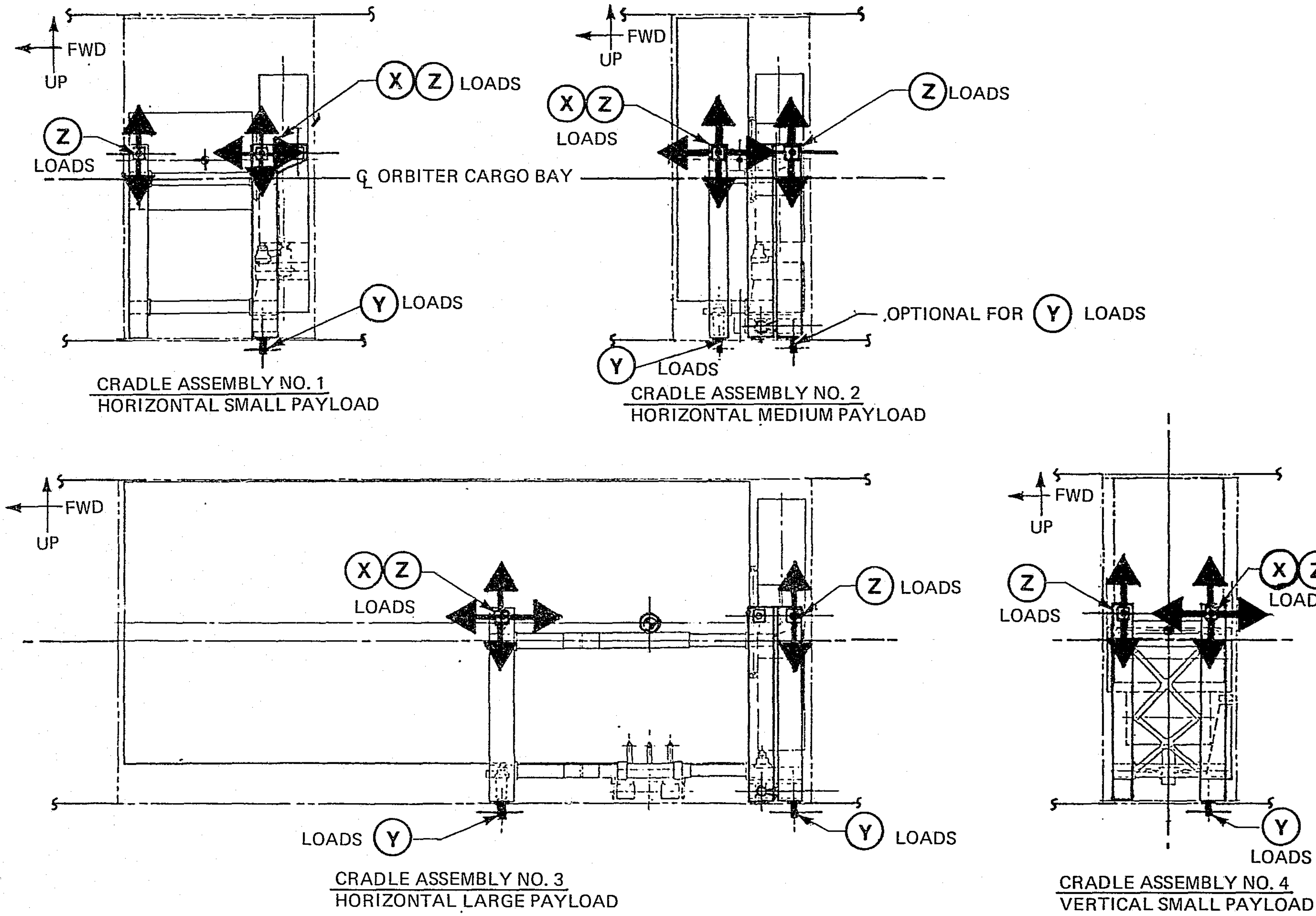


FIGURE 5.10 SCHEMATIC DIAGRAMS OF LES/ASE CRADLE CONFIGURATION LOAD PATHS TO ORBITER

assembly configurations which will support and deploy the low energy payload/stage combinations. The basic structural system employs a 5 point suspension interface with the Orbiter and uses the cradle assembly inherent torsional flexibility to assure low stresses when subjected to Orbiter flexible motion. The LES cradle assembly is designed to support the payload groupings as defined in paragraph 4.1 for 2.0 g handling and shipping loads while outside the Orbiter cargo bay and for 4.5 g_x, 4.5 g_z, and 1.5 g_y for emergency Orbiter landing condition while attached inside the Orbiter cargo bay. These and other design criteria are defined in the program definition and requirements sections of references 24, 37, and 38. LES ASE cradle structural factor of safety for ultimate loads is 2.0 and the ASE will be proof loaded to 1.1; thus STS user charges based on weight factors are not significantly penalized by excessive ASE cradle weights.

Four adaptations of SSUS-A and SSUS-D boosted configurations, shown in Figure 5.8 and Figure 5.9, use existing SSUS-A and SSUS-D ASE cradles. The SSUS cradle assemblies also employ a 5 point suspension system interface with the Orbiter. The combined delivery stage and payload weights for the adaptations are lower than published SSUS-A/D payload capabilities; therefore, the SSUS cradle assemblies are considered structurally adequate.

The horizontally mounted payloads shown in Figures 5.5 and 5.6 use the telescoping modular LES cradle assemblies which have two key features. These are:

(1) The forward U-frame supports the payload at the payload center of gravity.

(2) A walking beam, designed to longitudinally slide and pivot about the Orbiter center of torsional rotation, supports the LES for Z and Y stabilization loads.

The telescoping feature allows the support points to be positioned for large payloads so that the LES structural loads do not exceed the LES structural loads resulting from a design condition wherein LES supports a cantilevered 200 kg (441 lb.) payload as shown in Figure 5.4.

5.2 AVIONICS EQUIPMENT

Typical control, display and avionic ASE is identified and described that provides interface of LES/cradle assembly/avionics to the payload accommodations equipment. These equipment requirements for a typical LES, cradle assembly and avionics consist of the following equipment:

- Control and Monitor Panel
- Cradle Power Control Unit
- Cradle Signal/Data Interface Unit
- Cradle Deployment Mechanism Unit
- High Gain Antenna and Receiver (optional)
- Cable Plant
- Cradle Harnesses

The functional interfaces include:

- Controls and monitoring to activate, test and deploy the LES/spacecraft.
- Caution and Warning monitoring for status of payload and safing controls.
- Cradle functional interfaces for control and monitoring.
- Telemetry antenna and receiver for reception through apogee firing (optional).

The controls, displays, avionic ASE and cable harness as integrated with payload accommodations equipment are shown in Figure 5.11. This approach utilizes the Orbiter payload accommodations equipment where practical with additional equipment added if required. Provisions for interfacing payload accommodations equipment on the Orbiter aft flight deck are available for LES avionics ASE at two distribution panels. These are: (1) Mission Station Distribution Panel (MSDP) and (2) Payload Station Distribution Panel (PSDP). All wiring on the aft flight deck is Orbiter furnished, including that from the PSDP to the payload station and from the MSDP and PSDP to the pressure bulkhead at station Xo 576. The cable harness from the control and monitor panel to the PSDP cable interface is supplied as part of the low energy stage panel. ASE avionics cabling harness will interface with aft flight deck equipment at the pressure bulkhead (Xo 576) as shown in Figure 5.11.

5.2.1 Control and Monitor Panel

The Control and Monitor Panel provides dedicated switching and indicators to activate, checkout, control and monitor the LES and the cradle

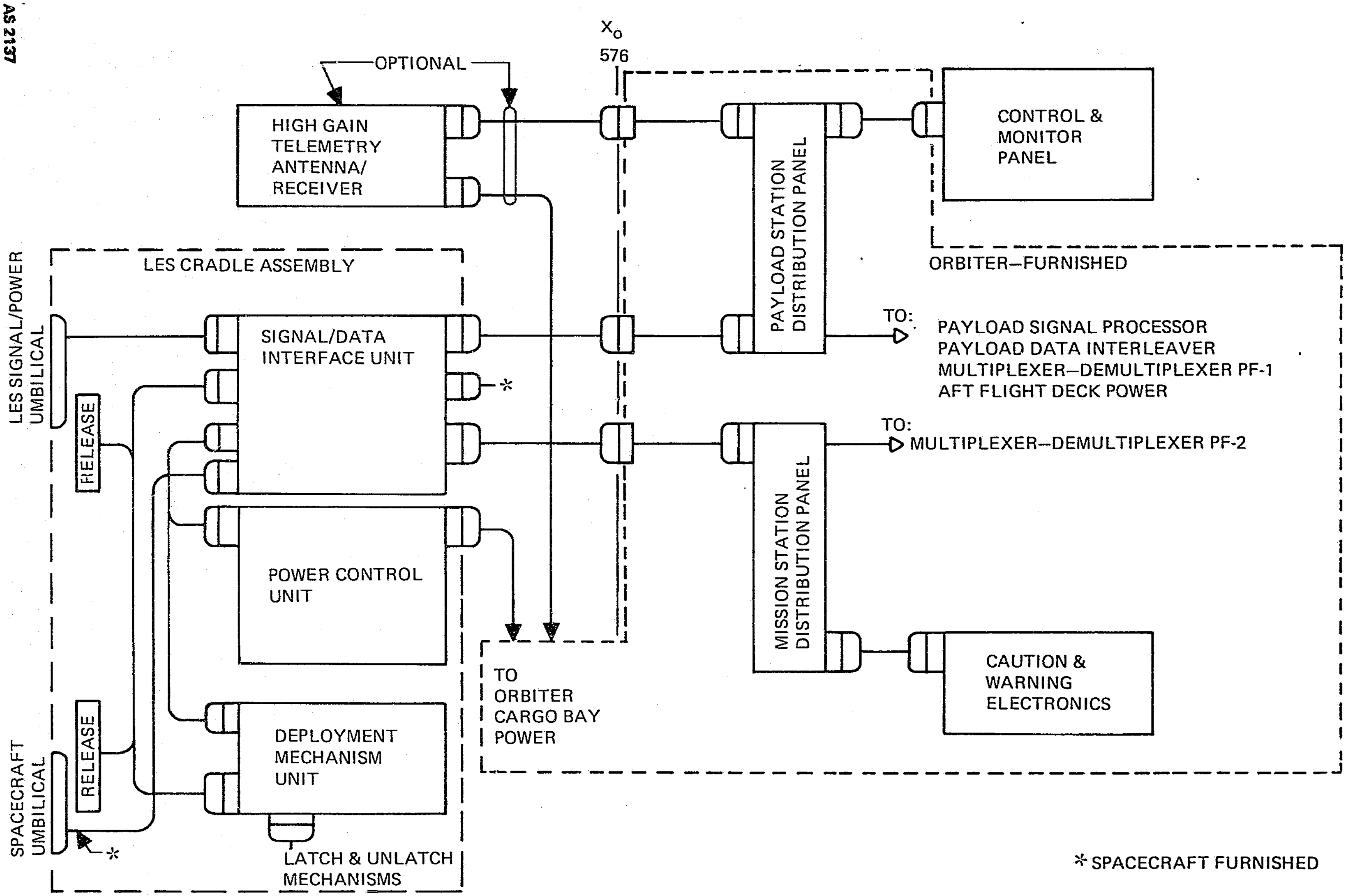


FIGURE 5.11 AVIONICS AIRBORNE SUPPORT EQUIPMENT AND CABLING DIAGRAM

avionics ASE. This panel is installed in the payload station and connected by Orbiter-furnished cabling to the payload station distribution panel. Table 5-IV lists the functions, circuitry, interface type, and wiring requirements for a typical control and monitor panel.

The power control switches generally operate relay drivers in the Signal/Data Interface Unit (S/DIU) which drive relays in the cradle Power Control Unit (PCU) and the LES PCU and Ignition Control Unit (ICU). ASE power application and removal is made selectable by function in order to provide flexibility and conserve Orbiter power. From Orbiter launch to predeployment test LES is in the "OFF" mode. Upon ignition of predeployment test the Orbiter power source is required for external system checkout and deployment. The cradle select switch applies activation control and monitor power to the cradle and the deployment mechanism. Provisions to apply power and separate the LES are designed into the panels. Upon separation these circuits are turned off. The telemetry antenna and receivers, if installed, are turned on before LES separation, left on through apogee firing, and then turned off.

The LES is activated and checked out on external (Orbiter) power prior to battery activation and transferred to internal power inputs for the Inertial Stabilization Unit (ISU) and telemetry loads so that either can be turned on without the other for checkout purposes. LES external/internal power transfer relay contacts are of the make-before-break type for uninterrupted power to the ISU. For safety reasons the LES is separated from its cradle before enabling the ICU, reaction control valves, and propulsion valves and before the propellants are pressurized. A typical sequence up to separation from the Orbiter is:

(1) Apply external power to the ISU and check it.
(2) Apply external power to the telemetry transmitter and verify reception and demodulation.
(3) Activate the battery.
(4) Transfer to internal power.
(5) Turn off external power.

To safe the LES electrical system after battery activation, external power is removed and the battery is discharged through a dummy load in the cradle PCU. The cradle PCU external power relay is opened to remove

TABLE 5-IV CONTROL AND MONITOR PANEL FUNCTIONS

FUNCTION	PROVISIONS			
	SWITCH	PANEL INDICATOR	WIRING	NOTE
ASE CAUTION AND WARNING POWER	1-SPST	1	2	1
ASE CONTROL AND MONITOR POWER	1-SPST	1	2	2
ASE HEATER POWER	1-SPDT	2	3	1
ASE DEPLOYMENT DEVICE POWER	1-SPDT	2	3	2
ASE TM ANTENNA/RECEIVER POWER	1-SPDT	2	3	
ASE DEPLOY EXTEND/RETRACT/INTERVAL	1-1P	3POSITION-ILLUMINATED	5	2
LES EXTERNAL POWER	1-DPST	2	4	2
LES GUIDANCE EXTERNAL POWER	1-DPST	2	3	2
LES TELEMETRY EXTERNAL POWER	1-DPST	2	3	2
LES BATTERY ACTIVATE	1-DPST	-	2	2
LES GUIDANCE TEST/OPERATE	1-DPST	2		
LES GUIDANCE RUN/HOLD	1-DPST	2	USES	
LES TIMER RESET/COUNTING	1-DPST	2	SERIAL	
LES PROGRAM LOAD/COMPARE/VERIFY	1-1P	3POSITION-ILLUMINATED	INPUT/	3
LES TEST SELECT	1-BCD	-	OUTPUTS	
LES TEST INITIATE	1-DPST	-		
LES TEST BUS/GO/NO-GO	1-1P	3POSITION-ILLUMINATED		
CAUTION AND WARNING GO/NO-GO STATUS -				
ASE TM ANTENNA/RECEIVER	-	2	2	1
LES/CRADLE	-	2	2	1
SAFING COMMAND ENABLE -				
ASE TM ANTENNA/RECEIVER	1-DPST	2	4	
LES/CRADLE	1-DPST	2	4	1
SAFING COMMAND INITIATE	1-DPST	1	3	
JETTISON COMMAND INITIATE	1-DPST	1	3	
GUIDANCE SERIAL INPUT/OUTPUT TERMINAL	-	-	4 TSP	2,3
ORBITER SERIAL INPUT/OUTPUT TERMINAL	-	-	4 TSP	3

NOTES: 1. FOR MULTIPLE LES/CRADLES, DUPLICATE FOR EACH LES/CRADLE
2. FOR MULTIPLE LES/CRADLES, BUSS THE WIRING TO ALL LES/CRADLES
3. MICROPROCESSOR PROVIDED AS A COMPONENT PART OF THE PANEL

external power and a battery safing relay in the LES PCU connects the battery to the dummy load. The battery safing relay is needed to deadface the power umbilical for normal operations.

For multiple cradles, a caution and warning selector switch can be added to select monitor signals from the desired cradle for input to the Multiplexer-Demultiplexer (MDM). The switch operates relays in the selected cradle S/DIU to access the monitor signals and the direct caution and warning signal is similarly accessed for display on the GO, NO-GO status indicators.

Safing from the control and monitor panel requires that two momentary push buttons be concurrently depressed - one command enable switch and one command initiate switch. A command enable switch is provided for the LES/cradle and one for the telemetry antenna/receiver. Two command initiate switches are provided, one for safing and one for jettison.

Control and monitoring of the LES ISU is performed through its TEST/OPERATE and RUN/HOLD control lines and its GSE Serial Digital Input/ Output (SIO), which are connected to the control and monitor panel. The SIO is required to load, compare and verify test programs and flight parameters, and to service dedicated controls and indicators on the control and monitor panel. The control and monitor couples its six dedicated guidance switches and ten indicators to the ISU SIO and to a MDM SIO. The MDM SIO is connected to the Orbiter payload accommodations, via MDM installation PF-1, and thus to the general purpose computer and associated payload accommodations.

The ISU and the MDM are each designed to control their respective SIO's, thus any direct interface between them is precluded. Hence, the control and monitor panel provides each SIO with: (1) an interface it can control, (2) input and output registers for both SIO's and for the dedicated switches and indicators on the panel, and (3) a microprocessor to load and read these registers.

5.2.2 Cable Plant

Interconnecting cable harnesses are provided to interface electrical and avionics equipment in the Orbiter. Separate routing of signal, power and radio frequency circuits is employed, if possible, to supplement shielding in the control and elimination of electromagnetic interference. Shielding of sensitive circuits is provided in all the cables.

The cable plant requirements for the Orbiter payload equipment interfaces were shown in Figure 5.11 and are illustrated in Figure 5.12. The power and signal cables for the high gain antenna and receiver are supplied as optional equipment. This cable plant arrangement provides all the electrical interfaces required to integrate Orbiter payload, ASE avionics and LES to Orbiter payload accommodations equipment. Interface provisions are provided for Orbiter payload furnished signal and power cable harness. Two sets of cabling are provided for each operational site. One set for the cargo integrated test equipment and one set for the Orbiter. Both cabling sets are considered flight items.

5.2.3 Cradle Avionics ASE

LES cradle avionics ASE include three packaged units which are installed and becomes an integral part of the cradle assembly. The three units are identified as: (1) Power Control Unit (PCU), (2) Signal/Data Interface Unit (S/DIU), and (3) the Deployment Mechanism Unit (DMU). Each unit is packaged separately to enhance cradle installation and to isolate signal and power circuits to minimize electromagnetic interference (EMI). In addition, three sets of cable harnesses are provided for electrically interfacing the PCU, S/DIU and DMU. First, the PCU power cable provides an interface for power, control and monitoring to LES via the S/DIU and to the latch and unlatching mechanism via the DMU. Secondly, a cable harness is provided between the S/DIU and the DMU for control and monitoring of the deployment mechanisms. Thirdly, the LES umbilical extension from the S/DIU provides stage control and monitoring plus the caution and warning circuit interfaces. The cargo bay cabling plant was discussed in 5.2.2. Provisions will be available for interfacing spacecraft furnished umbilicals.

5.2.3.1 Power Control Unit - The cradle Power Control Unit accepts and transfers Orbiter power to the LES power umbilical, spacecraft umbilical release, signal/data interface unit and deployment mechanism. Complexity is twelve latching relays with readback contacts. Power control and monitor lines are routed to and through the Signal/Data Interface Unit.

5.2.3.2 Signal/Data Interface Unit - The cradle Signal/Data Interface Unit (S/DIU) provides the command/response interface required between the LES

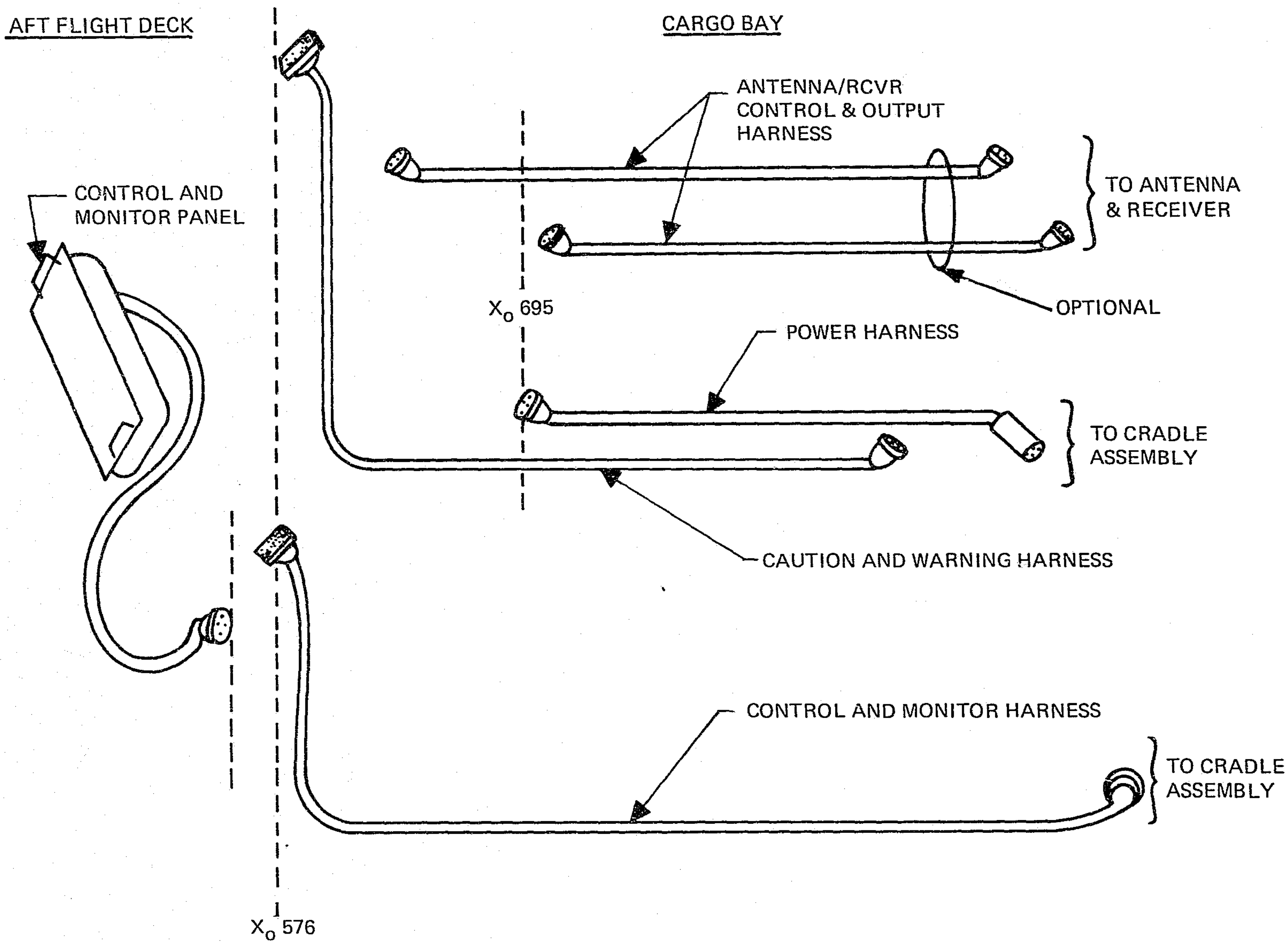


FIGURE 5.12 ASE CABLE PLANT FOR SINGLE LES ORBITER INSTALLATION

and cradle and from these to the control and monitor panel and to existing payload accommodations. Relay drivers are provided for the transfer relays in the ASE and LES power control units. Caution and warning (C&W) sensors can be provided for excitation and signal conditioning that is compatible with the Orbiter-furnished C&W electronics. C&W circuits would be separately powered for continuous operation. Cradle detection is provided to detect the limit positions and latch release states. Cradle cabling harnesses are provided to interface the LES umbilical to the S/DIU and cradle interconnects for avionic ASE.

5.2.3.3 High Gain Telemetry Antenna/Receiver - The high gain telemetry antenna/receiver can be added as optional equipment in order to extend the transmission range to approximately 2050 km (1100 nm) for coverage of the LES apogee firing. The low gain (7 db) Orbiter payload support antenna, payload interrogator and omnidirectional LES antenna provide limited range.

5.2.4 Orbiter Interfaces

Orbiter interfaces for LES and LES ASE are provided that electrically utilize the payload accommodations equipment where practical. LES systems interfaces via the umbilical to the ASE avionics and payload accommodations equipment include the Inertial Stabilization Unit (ISU), Ignition Control Unit (ICU), Power Control Unit (PCU), and telemetry system. The ISU provides pretimed commands, flight control outputs, telemetry outputs, power and support equipment interfaces. The pretimed commands are relay driver outputs to initiate individual ignition and separation events. The flight control outputs include analog signals and solenoid driver outputs for the reaction control valves. Power required to the systems is 28 vdc, uninterrupted while transferring from external power to internal power. The ISU accepts the uninterruptible 28 vdc and performs the required activation sequence.

The ICU provides routing of the electrical power, switches, firing commands and provides safe/arm capability for all pyrotechnics initiators on the LES. The ISU provides the commands to switch power to the pyrotechnic initiators for all events. This unit is inhibited until after separation from Orbiter.

The electrical power system provides electrical energy to all equipment on LES. The PCU provides the capability for switching between external and LES internal power sources during predeployment checkout.

PCM telemetry interfaces are compatible with the Orbiter Engineering Data Handling System. The Engineering Data Handling System on the Orbiter performs S-band reception via the Payload Interrogator and provides hardwired inputs to the Payload Signal Processor (PSP) and the Payload Data Interleaver (PDI). For LES predeployment checkout the ISU Serial Digital Input/Output is serviced, via the umbilical. If telemetry coverage is required through LES apogee firing, provisions are required for increasing Orbiter receiving range.

The ISU is programmed to a PCM telemetry bit rate of 16 kbps for assured compatibility with the PDI which accepts and interleaves up to five simultaneous PCM signals from Orbiter payloads with up to 48 kbps of other signals to form a 128 kbps output. The PDI requires that the payload PCM bit rate be 16 kbps or, when full capacity is not allocated, any one signal could be at a multiple of 16 kbps up to 64 kbps.

To service the Serial Digital Input/Output, the Engineering Data Handling System provides a General Purpose Computer (GPC) with interfaces to the Multifunction CRT Display System (MCDS), onboard storage media and two Multiplexer-Demultiplexers (MDM) designated PF-1 and PF-2. The MDM interfaces the GPC to a Serial Input/Output (SIO), discrete inputs and outputs, and analog inputs and outputs. If the MDM SIO were interfaced to the ISU SIO, the Orbiter GPC and peripherals could be used to support the LES on-orbit checkout and deployment. However, the MDM and the ISU each need control of their respective SIO's and any direct interface between the two SIO's is precluded. Therefore, the control and monitor panel also provides output and input registers for its own dedicated switches and indicators, and for the two SIO interfaces, and services these registers with its own microprocessor. Panel design provisions can also be provided for antonomous control of normal checkout and for use of the MDM link for program load/compare or for data display and readout in event of a NO-GO in checkout.

The MDM SIO involves a four-signal interface, data (half duplex), user-generated work discrete, and two MDM-generated discretes, message in (to MDM), and message out (from MDM). Reference 24 provides a detailed discussion of the MDM SIO.

The Orbiter Caution and Warning (C&W) electronics are interfaced to the LES/cradle assembly via the C&W umbilical cable harness. Orbiter payload accommodations for caution, warning and safing functions are entirely located on the flight deck. Interfaces to this equipment for cargo bay payloads is provided at the station Xo 576 pressure bulkhead connector panel.

Direct wired monitor channels are inputs to the Caution and Warning Electronics unit (CWE). Direct wired safing commands are from switches on the forward flight deck. The Multiplexer-Demultiplexer provides these interfaces for computer processing and operates a sixth discrete monitor channel of the CWE. The computer can present the monitor results to the Multifunction CRT Display System (MCDS). It is expected that the LES will not require C&W interfaces due to the systems remaining in the "OFF" mode during launch, ascent and up to the initiation of predeployment checkout. If C&W interface is required LES would be limited to one hardwired monitor, one hardwired command, and a reasonable number of channels through the MDM. The MCDS would be available in case of anomaly indication to more precisely identify the sensor and to present quantitative readings.

Table 5-V lists typical caution and warning anomalies with their sensing and safing methods that could be required. These analog sensors would

TABLE 5-V TYPICAL CAUTION AND WARNING ANOMALIES

ANOMALY	SENSOR	SAFING
Pressurant-Overheat -Over Pressure	Temp sensor/Tank Pressure at Manifold	Pressurant Onboard Vent Squib Valve
Battery Overheat	Temp Sensor/Battery	Transfer LES to external power and shut down, then discharge battery through dummy load in ASE PCU
ICU Power	Voltage Monitor	
Pyrotechnic Busses	Voltage Monitor	
Inadvertent Act of Ignition Time Sequencing	Relay-existing in Guid (Deploy SW Action will be command to relay)	Remove power from guid sys and return P/L

be used so that once an anomaly is detected the Orbiter backup caution and warning system can extract information for decision making and subsequent analysis. The sensors would be five thermistors, one potentiometric pressure transducer and three voltage monitor points, all located on the LES.

Mission unique ASE circuitry is required to excite the transducers and to interface all monitor channels to the Orbiter Caution and Warning Electronics. Direct-wired anomaly detection for all monitor channels are amplitude detected and the logic outputs from the detectors combine on one line by OR logic such that a warning is generated when one or more detectors are set. Each monitor signal is interfaced through an unloading amplifier and the contacts of a payload selector relay located in the cradle Signal/Data Interface Unit. The payload selector relay is closed from the control and monitor panel.

The three safing functions are: (1) activate pressurant onboard squib valve, (2) transfer LES from internal power to external power, and (3) discharge battery through dummy load. Relay drivers for these functions are provided at the cradle. They are selectable from either the control and monitor panel, the C3A5 panel in the forward cabin, or the Multiplexer-Demultiplexer safing commands.

5.3 SAFETY

Safety of flight crew and Orbiter was a prime consideration in the development of conceptual design integration, test of ASE avionics, and LES cradle assembly. For example, the pyrotechnic circuits must have three failures in order to inadvertently initiate a hazardous event. The initiators for the ignition and reaction control systems are safed on installation (short circuit to the initiator bridge wire) and remain so until after separation from the Orbiter. Propellant and oxidizer tanks are preserviced and sealed. They are to be designed and qualified to avoid tank rupture and to withstand total pressure from the environment. Pressure tanks are also to be designed to withstand total pressure from environment.

5.4 PROPULSION CONCEPT ASSESSMENT

Developed ASE conceptual designs and interface requirements were evaluated to identify the degree of complexity and packaging efficiency and the impact on selected propulsion modes. These impacts are:

- ASE cradle avionics require single umbilical interface to LES from the Signal/Data Interface Unit.
- Orbiter power usage is minimized since LES remains in the "OFF" mode until on-orbit predeployment checkout.

- Control and monitor panel has output and input registers and services these registers with its microprocessor.
- Provisions for spacecraft umbilical interface can be provided via the S/DIU. Each spacecraft umbilical extension cable will probably require unique design.
- Simplified interface can be provided between ASE control and monitor panel to payload station distribution panel.
- Umbilical located on LES is away from LES plume to avoid any possible shorting due to contamination or burn through.
- Separation mechanism for umbilical will be by lanyard or electrical release. Pyrotechnics release was not used since it has prejudical failure mode.
- Umbilical interface is designed to separate with LES.
- Spacecraft interfaces are restricted to basic mechanical and electrical interfaces with each spacecraft providing its own test and checkout equipment and procedures.
- Spacecraft to LES interface is minimized through use of standard adapter - to minimize development cost.
- Thermal impact on stage Orbiter and spacecraft is negligible since heat transfer is controlled by insulation blanket.
- No venting or dumping interfaces to Orbiter are required.
- Orbiter payload physical and functional interfaces are designed to make maximum use of payload accommodations equipment.

- ASE installed in the aft flight deck or cargo bay must be compatible with those environments and materials.
- Orbiter bay envelope clearance and attachments meet the requirements of Reference 24.
- No additional user charge penalty is incurred for cradle length since the cradle is designed for the payload/stage to be mounted and installed within the overall cradle dimensions.
- ASE cradle weight is heavier for 4-tank than for 8-tank LES due to the need to fill the space between the inner cradle contour and the smaller stage structural pickup points with cradle fillers and adapter structures.
- Monopropellant LES is larger in diameter and length than the bipropellant LES and is more difficult to fit into the 4.0 meter (13.12 ft.) space envelope established for LES ASE cradle assemblies. However, the additional stage length of 0.114 m (0.375 ft.), which affects user's charge, will not impact cradle design or cost.

6.0 TASK 5: GROUND AND FLIGHT OPERATIONS

This section identifies and describes support requirements and equipment associated with ground and flight operations. Ground and flight operational requirements such as timelines, support equipment, facilities and personnel are defined for a typical Low Energy Stage and ASE in paragraphs 4.0 and 5.0. The concept of emphasizing buildup, assembly and tests at the factory was used as the basic approach. Operations at launch site facilities include receiving, inspection, assembly and interface tests with the cradle assembly and payload, Orbiter support interface tests, range and status verifications. The approach discussed provides an efficiently controlled processing and launch preparation capability.

Flight operations include monitoring safe status of LES, ASE, and payload from Shuttle launch through predeployment tests and payload separation. The flight sequence begins at deployment and ends with payload separation from LES. Telemetry coverage is to be provided during the entire flight. The data is used for post-flight reporting and to verify performance of LES.

6.1 GROUND OPERATIONS

Typical baseline ground operations were established and evaluated for both the Orbiter Processing Facility (OPF) and the Pad flows as identified in Reference 24. Operations developed here are consistent with the requirements of References 24, 38, and 40 through 44. For the purposes of this study the Pad baseline flow is considered primary and is used throughout as representative of a typical timeline allocation for deriving a ground flow and defining supporting elements for cost determination.

Major task elements were defined and evaluated in terms of initial conditions, sequence of functions under consideration and the output conditions required for the subsequent task element. Once the functions were defined in terms of satisfying the operational concept, work timeline allocations were established. From these timelines and the ground flow, support equipment, personnel and their skills and facilities were established and defined to a level sufficient to derive costs.

6.1.1 Task Elements

Task elements were separated into four categories. These are: (1) Factory Test, (2) Field Test, (3) Transporting and Handling and (4)

Special Handling. Tests at both the factory and field provide verification of the following:

- Electrical Installation
- System Operation and Compatibility
- Test Equipment
- Test Procedures
- Design Change Evaluation
- Malfunction Investigation

Major tests that are identified and discussed are as follows:

a. Factory

- Acceptance Tests
- Bench Tests
- Assembled LES Tests
- Simulated Flight Tests

b. Field

- Systems Test
- Cargo Integrated Tests
- Preflight Readiness Tests

6.1.1.1 Factory Test Requirements

a. Acceptance Tests

Recommended acceptance tests for the Inertial Stabilization Unit (ISU) and PCM signal conditioning are summarized below.

(1) Inertial Stabilization Unit - As shown in Figure 6.1, the ISU acceptance test will be conducted with the package mounted on the dividing head and interfacing with the Ground Support Equipment only. The dividing head is used to accurately reposition the ISU about a given axis while making performance measurements. The following tests are conducted:

- Continuity and Isolation Check
- Input Power and Power Interrupt
- Built-In Test
- Operating Modes Verification
- Processor Performance Test

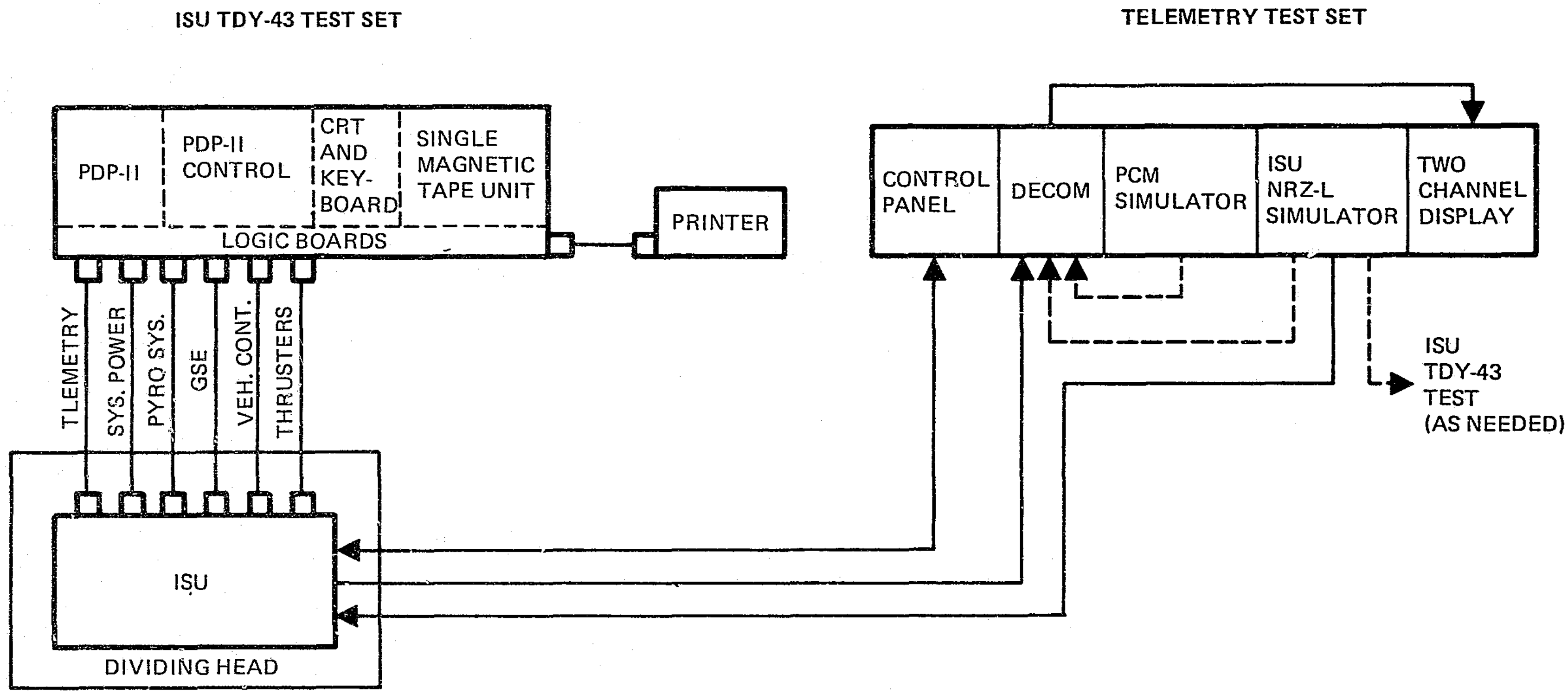


FIGURE 6.1 ACCEPTANCE TEST AND INERTIAL STABILIZATION UNIT BENCH TEST CONFIGURATION

- Gyro Performance Test
  - Alignment Verification
  - Drift Check
- Accelerometer Performance Test
- Stage Control Interface (using simulated loads)
- Pyrotechnic Outputs (using simulated loads)
- Telemetry Interface (using test set loads)
- GSE Interface Test

(2) PCM Signal Conditioning - The telemetry acceptance test configuration is shown in Figure 6.1. The following tests will be required for the PCM Signal Conditioning:

- Continuity and Isolation Test
- Input Power
- A/D Converter Calibration
- Sensor Power Supply Checks
- Sensor Simulation and Signal Conditioning Verification
- Output Waveform Verification
- Internal/External Slaving Mode Check
- Channel Crosstalk Check

b. Bench Tests

A Bench Level Test is recommended for the ISU even though it is largely a repeat of the acceptance level tests. The primary difference between the Bench and Acceptance level tests will be the incorporation and verification of the mission related software parameters. A bench test for the PCM Signal Conditioning is not required. The following are bench level tests for the Inertial Stabilization Unit:

- Input Power and Power Interrupt
- Built-In Test
- Processor Performance
- Operating Modes
- Gyro Performance
  - Alignment Verification
  - Drift Check
- Stage Control Interface (using simulated loads)
- Pyrotechnic Outputs (using simulated loads)

- Telemetry Interface (using test set loads)
- GSE Interface

Selected functions for this test will be repeated in the field as an ISU verification test prior to conducting the cargo integrated tests.

c. Assembled LES Tests

An assembled LES test is required to demonstrate LES compatibility using actual hardware and to verify LES integrity. For these tests, the avionics equipment such as the Inertial Stabilization Unit, Ignition Control Unit (ICU), Power Control Unit (PCU), etc. will be installed on the stage. The test configuration of Figure 6.2 is required to demonstrate LES compatibility using actual hardware and to verify LES integrity. The following are tests for the Assembled LES Test:

- GSE Interface
- Input Power and Power Interrupt
- Build-In Test
- Processor Performance
- Operating Modes
- Telemetry in Vehicle End-to-End Calibration
- Gyro Performance
  - Alignment Verification
  - Drift Check
- Accelerometer Performance
- Stage Control Interface (using simulated loads)
- Ignition System Tests (using simulated loads)
- Telemetry System Checks
- Guidance Power Switching
- Mission Event Timing Test (using LES Flight Program and simulator loads)

Ignition system tests are typical of the scope of these tests. Ignition system functional tests will be conducted by connecting squib simulators to the initiator connectors and operating the ICU through its flight sequence. Outputs to the squib simulators will be monitored to demonstrate that firing signals occur at the proper time. Ignition capacitor bus voltages will be monitored by oscillograph in order to demonstrate integrity of the capacitor

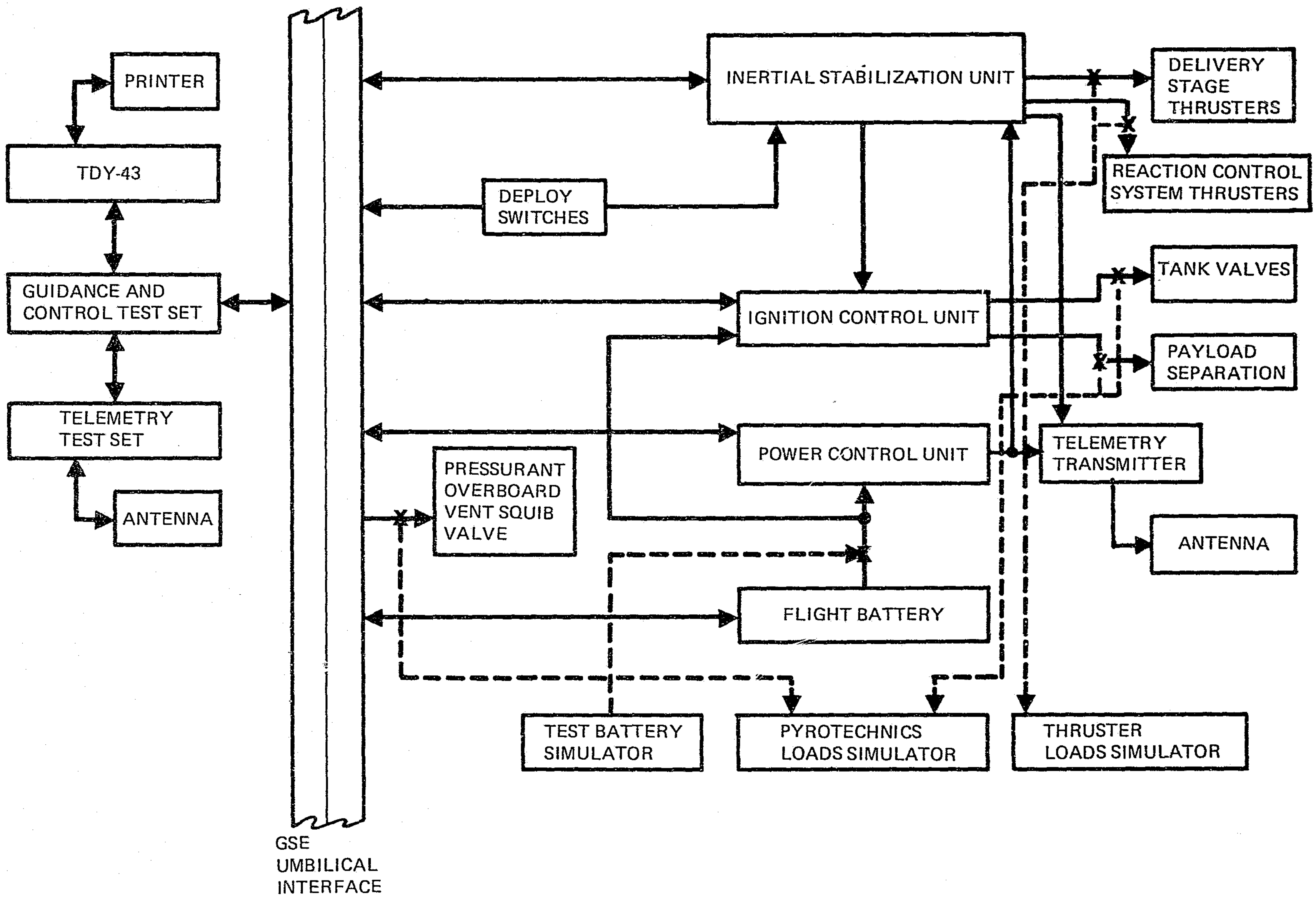


FIGURE 6.2 ASSEMBLED LES TEST CONFIGURATION USING TEST SETS AND SIMULATORS

banks by observing their recharge characteristics. The ignition system will not operate on external power; therefore, the stage will be provided power by a battery simulator which replaces the flight battery for these tests. It may not be practical to operate the deploy switches which initiate the flight sequence of the guidance system during these tests; therefore, circuitry bypassing these switches will be provided which enables the initiation of the flight sequence by the test set.

d. Simulated Flight Tests

The simulated flight test is conducted for final acceptance of the assembled LES. During this test, all systems are operated on internal power and the umbilicals are physically separated. All avionics equipment will be installed in the LES with the exception of the Flight Battery which will be remotely located. Provisions for de-energizing the systems will be provided on the Battery Simulator. The following tests are conducted in the simulated flight test:

- Thrust Motor Circuit Verification (using simulated loads)
- Ignition Pyrotechnics System Verification (using simulated loads)
- Inertial Stabilization Unit Verification
- Telemetry System Verification (hardline and RF)
- RFI Test
- Simulated Flight Test

6.1.1.2 Field Test Requirements

a. Systems Test

During this test all systems will be activated to verify system performance, individually, and in conjunction with other systems to verify functional operation of LES systems after mating to cradle assembly. This test is similar to the factory simulated flight test. The following tests are conducted for the systems functional verification tests.

- Thrust Motor Circuit Verification (using simulated loads)
- Ignition System Verification (using test loads)
- Inertial Stabilization Unit Verification
- Telemetry System Verification
- Payload System Interface Verification
- Simulated Flight Test

b. <u>Cargo Integrated Tests</u>

This test will be performed after the LES has been interfaced with the cargo bay simulator test equipment, and is conducted to verify the operational capability of the LES avionic equipment using the control monitor and display panel (ASE) as interfaced to Cargo Integrated Test Equipment (CITE). This is a dress rehearsal of the actual Preflight Readiness and Predeployment tests to monitor all selected systems and verify critical parameters. All avionics equipment and pyrotechnics are installed in the LES and the pyrotechnics are safed. The pyrotechnic circuits are not activated during these tests but flight and crew safety functions are monitored via the caution and warning interface. The following tests are conducted.

- Avionics Systems Checkout
- Telemetry Ambient Readout
- Inertial Stabilization Unit
- Drift Test
- Guidance Parameter Monitoring
- Telemetry Parameter Monitoring
- Power Interface Verification

c. <u>Preflight Readiness Test</u>

In a manner similar to the cargo integrated tests, the preflight readiness tests will be a repeat of that test except the spacecraft is now installed and interfaced to Orbiter installed ASE avionics equipment. The ISU attitude will be monitored, automatic confidence checks will be made, and continuous performance monitoring, by Payload Specialist, BITE, and by hardline telemetry monitoring, will be accomplished. On-orbit predeployment tests are planned to be a repeat of the preflight readiness tests. It is not anticipated that these tests will vary except for the elimination of operational cycle tests of the deployment mechanisms and physical separation of the payload.

6.1.1.3 <u>Transporting and Handling</u> - Transportation and handling equipment that center around stage assembly and movement at the factory and field level are required. Provisions include stage movement at the factory, and between facilities at the field site. The flow provides major task events and sequences in handling and transporting LES, ASE and Orbiter payload. Preliminary facilities, checkout, integration and support equipment are identified and described in the context of handling, movement, clean room and safety requirements.

6.1.1.4 Special Handling

a. Prepackaged Propellant Tanks

Special handling and transportation provisions are required for prepackaged propellants and the gaseous helium pressure system. Since the safety requirements are essentially the same for bipropellants and monopropellants both are treated the same for this study. For the pressure system task element, the safety criteria for high pressure vessels was used.

Prepackaged propellant tanks are not installed at the factory. Inert tanks will be provided at the factory for mechnical and electrical fit checks. After simulated flight test these reusable tanks are removed and remain at the factory. The prepackaged propellant tanks are to be serviced at a qualified vendor.

The vendor, after filling the Nitrogen Tetroxide (N_2O_4) and Monomethylhydrazine (MMH) tanks, (either 50 percent or 100 percent loaded), seals the tanks without installing pyrotechnic valve initiators. These prepackaged tanks are monitored for 12 to 24 hours for leakage and pressure build-up prior to installation in shipping containers and shipment to the field site. For safety reasons a sniffer leakage check is performed before visual inspection. After passing both checks the tanks are stored in shipping containers and monitored daily until Shuttle launch. Upon assignment the tanks are transported to the assembly area for installation on LES. At all times these tanks are to be handled in compliance with safety guidelines to be established in the safety plan and approved by NASA/DoD.

b. Propellant Pressurization System

The pressurization system requires servicing equipment with accessories. The field site gaseous helium servicing system along with safety guidelines for high pressure vessels and systems were considered in defining task elements, equipment, and sequence. A pressure test of this system will be conducted after assembly in the LES to assure proper regulator lockup pressure and no leakage. This test will also have been performed at the factory.

6.1.2 Task Flow

Before defining and describing a typical LES field ground operations flow a preliminary definition of factory requirements were examined.

6.1.2.1 Factory Flow - A typical factory flow is shown in Figure 6.3, and represents all the major tasks performed from completion of LES fabrication, through assembly tests, simulated flight to final inspection, packaging/crating and shipping.

Manufacturing and fabrication of the LES parts and components and the installation of these into stage structure are accomplished in accordance with engineering drawings and manufacturing procedures. Vendor components are accepted in-plant to receiving and inspection acceptance test specifications prior to installation. After installation of the parts and components into the stage, the LES is checked out and tested to standardized processing procedures. These procedures are used for both factory and field processing. In plant, the procedures include bench testing, LES buildup, assembled LES and systems tests, and preparation for shipment. These same procedures, where applicable, are repeated at the field sites, and include system integration tests, preflight readiness tests and predeployment tests. In addition, these standard operating procedures include field unique requirements that cover testing and maintenance of the Ground Support Equipment (GSE) and Aerospace Ground Equipment (AGE) used to handle, transport and checkout the LES.

After completion of the acceptance and bench level tests, the LES and stage harness are electrically connected in the checkout area. The LES is then assembled with inert fuel tanks, plumbing connected and a functional test of LES wiring and systems performed. The purposes of the functional tests are: (1) to insure that an unsafe condition does not exist in the LES plumbing and wiring and (2) to prevent rework of the LES after it is received at the field site. This factory test checks and verifies all subsystems, power and insures that all subsystems and wiring are functionally correct prior to delivery to the field site.

Engineering data such as drawings, standard operating procedures (SOP's) and acceptance test specifications are controlled through a Configuration Management System which must approve any changes or revisions. The SOP's are to be segregated into individual procedures and tasks which cover

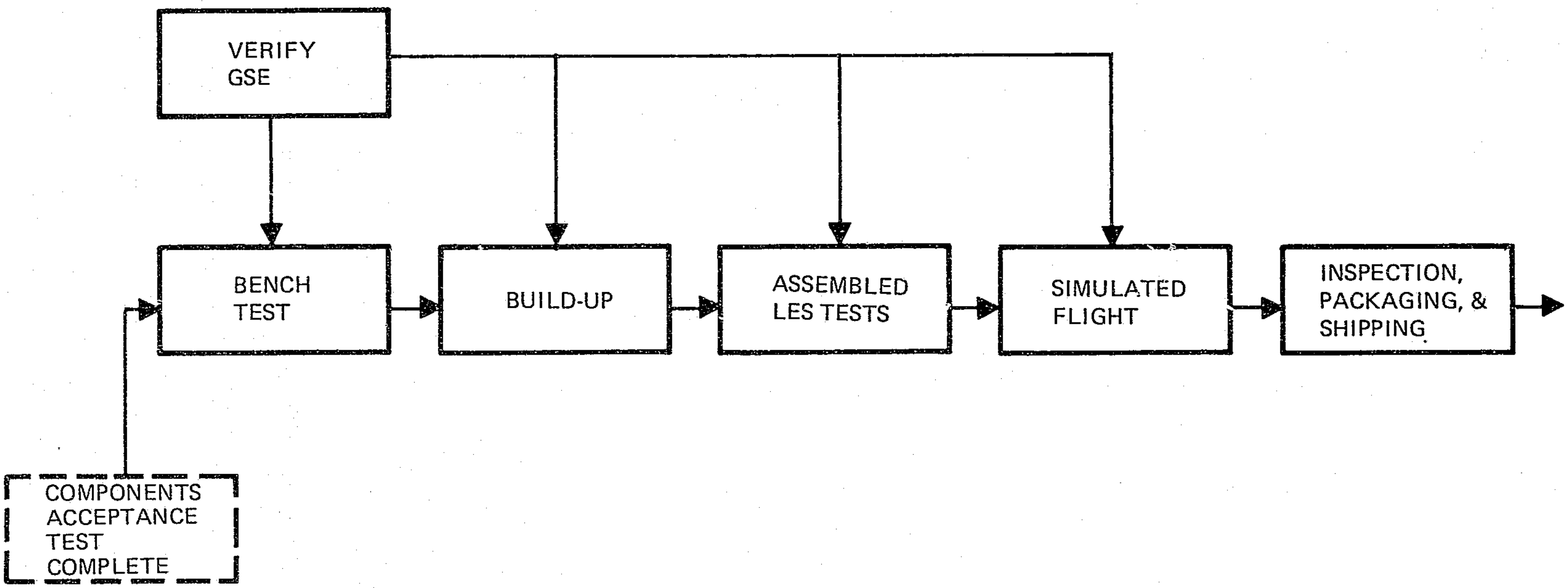


FIGURE 6.3 TYPICAL FACTORY FLOW

buildup, assembly or checkout of each system independently. Procedures are provided and functionally verified at the factory and field sites during the demonstration tests.

6.1.2.2 Field Flow - The typical ground operations flow for LES at the field site shown in Figure 6.4, represents the major activities that are performed from completion of receiving, uncrating and inspection of the LES preserviced fuel tanks and ordnance devices to preparation for and installation in the Orbiter cargo bay. This flow is used to derive timeline allocations for each task and to define support requirements such as personnel, equipment and facilities. A typical timeline for LES critical path requirements processed via the PCR to Orbiter is shown in Figure 6.5. The payload is the responsibility of the payload manufacturer. However, the mating of the payload to the LES will be assisted by the LES field team. Checkout, handling and transportation ground support equipment consist generally of those items listed in paragraph 6.1.3. These equipment are used for operational assembly work as well as for maintenance requirements. It is tentatively planned that the Spacecraft Assembly Encapsulating Facility will be available for most of the processing effort prior to transferring the payload to the Vertical Processing Facility (VPF) and subsequent installation in the Payload Changeout Room (PCR).

After inspection LES is mounted on the turn table device located on the Mobile Flat Bed Assembly (MFBA) transporter. This arrangement provides flexibility and ease of installation of the preserviced fuel tanks. The preserviced fuel tanks are installed on the LES using a chain hoist, a hydroset and sling assembly. Each tank is hoisted one at a time and mounted on the LES and secured. After all tanks are installed, the plumbing is connected and secured in preparation for surveillance leak check. Upon successful completion of the surveillance leak check, the Gaseous Helium (GHe) pressure system is pressurized to low pressure prior to installation on the cradle. A turn-over sling attached to the bridge crane provides the capability to rotate LES to a position for installation on the cradle assembly and the insulating blanket is installed. The LES umbilical and simulators are connected to the test sets in preparation for systems checkout. Test sets are energized and the LES systems are verified operational. This includes

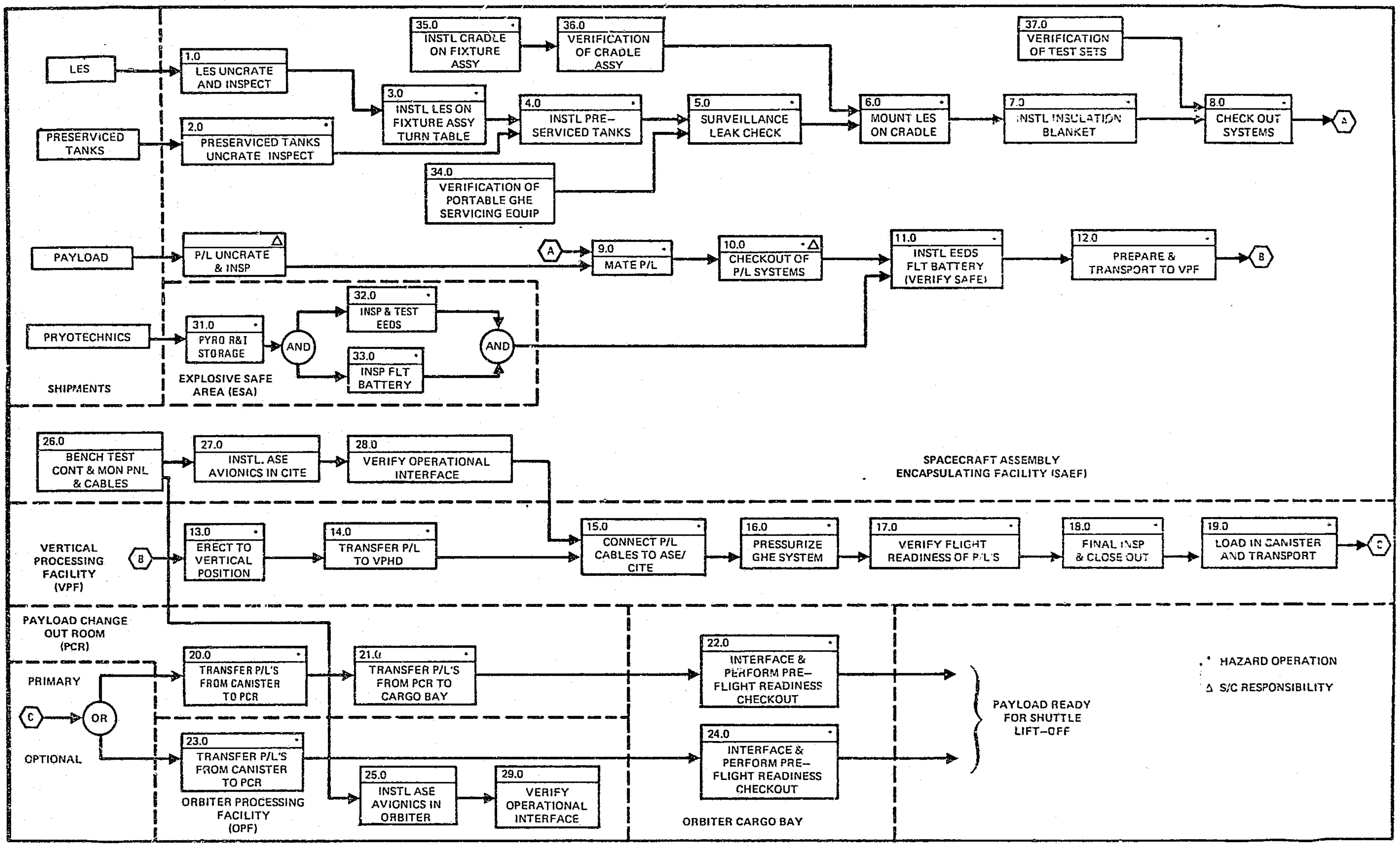


FIGURE 6.4 GROUND OPERATIONS FLOW DIAGRAM

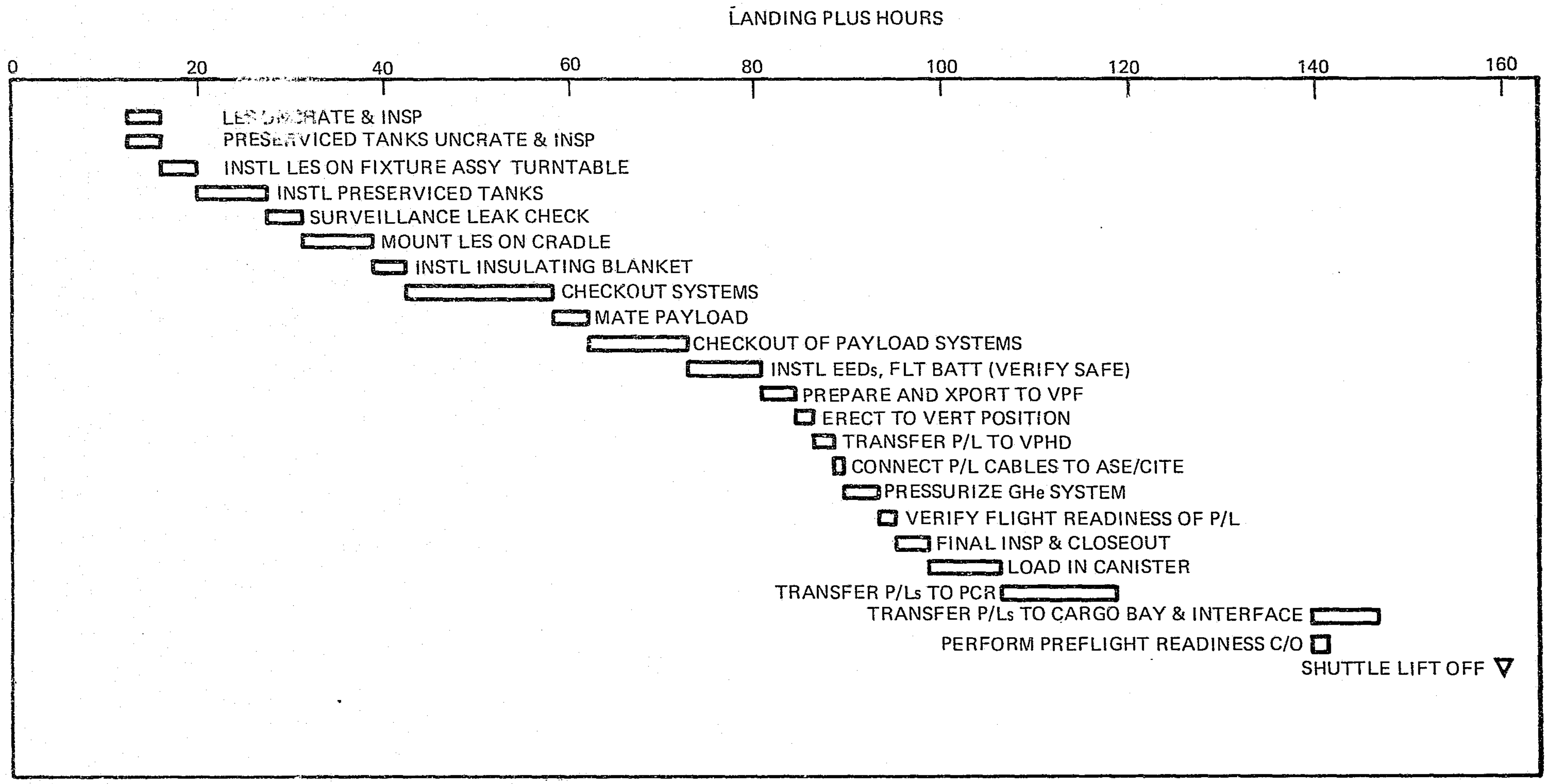


FIGURE 6.5 LES PROCESSING TIMELINE WHEN INSTALLATION OCCURS AT PAD

guidance and control, ignition, pyrotechnic circuits, power control unit, and telemetry using appropriate simulators and test loads where required. The systems and test sets are then de-energized and all support equipment disconnected in preparation for mating the payload.

Mating of the payload will be supported by the LES field team. Upon completion of this activity, the payload team verifies that the payload systems are operational. The payload systems are de-energized and the GSE disconnected in preparation for installation of LES pyrotechnics and flight battery. Door openings in the insulation blanket provide access to these areas. After installation and connection of the electroexplosive devices (EED's), they are verified safe. The payload is secured for transporting to the VPF on the MFBA. The MFBA is a commercial flat-bed trailer modified for LES handling, checkout and transporting. At the VPF the payload is erected to the vertical position using the erector hydraulic actuation system. The hoist sling is then attached to the payload and the bridge crane. The bridge crane is operated to hoist and mate the payload to the Vertical Payload Handling Device (VPHD). The payload cabling interfaces are completed to the Cargo Integrated Test Equipment (CITE) in preparation for the flight readiness tests. Prior to these tests, the portable servicing equipment is connected to the VPF GHe pressure system and LES. The LES pressure system is then pressurized to 3600 psi. The CITE checkout (essentially the same as predeployment tests) of the payload systems are performed to verify the interface and functional operation of payloads as they will be integrated in the cargo bay and aft flight deck prior to canister loading. The ASE avionics provided for these tests are identical to the flight ASE installed in the flight deck and cargo bay. The abbreviated systems checkout in the Orbiter cargo bay are identical to those performed in the CITE.

After completion of the CITE tests, preparations are made for transferring the complete flight cargo to the canister. For this activity, the canister is vertical on its transporter. The payload cargo is transferred to the canister and secured for transporting to the pad. Upon arrival at the pad the canister is positioned in front of the PCR and preparations are made for transferring the payload cargo to the PCR using the Payload Ground Handling Mechanism (PGHM). The payload cargo is transferred to the PCR and the canister returned to storage. In accordance with the scheduled timeline allocation the complete payload cargo is transferred to the Orbiter cargo bay using the PGHM. The necessary interfaces are completed in the Orbiter cargo bay in preparation for in-Orbiter checkout of the payload systems. Upon successful completion of preflight readiness checkout, the systems are de-energized. The LES will remain in this mode until ready for on-orbit predeployment tests. If required, the safe status of selected functions will be monitored via the Caution and Warning (C&W) electronics from completion of Task 22 (Figure 6.4) to deployment. Controls will be provided as required in case of an emergency.

The LES processing flow via the OPF is essentially the same up to loading in the canister except for facilities usage. For the OPF flow the payload cargo is transferred directly to the canister in the horizontal position (normally bypassing the VPF) and from the canister to the Orbiter cargo bay laying down. The GSE for performing these activities are provided as part of the MMSE. The checkout of the systems remains the same.

6.1.2.3 Adaptations - LES/SSUS-D adaptations of new low energy stages with a spinning solid upper stage (either SSUS-D or SSUS-A) were examined for operational interfaces and adaptability to a combined processing flow of the two stages. Preliminary evaluations indicate that both processing cycles can be initiated separately and proceed to the first interface task. For LES this task would be at completion of Task 8.0 of Figure 6.4. LES is prepared and transported to SSUS-D buildup and assembly facility for mating to SSUS-D which has been mated to its cradle assembly located on SSUS-D transporter.

Typical groundrules assumed for LES/SSUS-D adaptations are:

- SSUS-D transporting equipment would be used for transporting the combined SSUS-D cradle, PKM, LES, and spacecraft.
- Existing support equipment from both programs will be used. No additional requirements are anticipated.
- Ordnance installation and checkout will be a combined team effort.

- The team concept (both LES and SSUS-D) applies during each stage buildup assembly, checkout, integration and verification in the Orbiter.
- The two processing flows would parallel each other so that scheduled interfaces are met.

Preliminary examinations indicate that certain activities are not required for adaptations. These are:

- Dynamic balance of SSUS-D not required.
- Modification kit consisting of a signal data interface and power control unit are to be provided by LES program for installation on SSUS-D cradle.
- Existing control and monitoring ASE will be used from both programs. However where spin activities show in procedures, these action items are deleted.

Upon completion of the integration of SSUS-D and LES preparations are made for mating the spacecraft. The spacecraft is transported to the SSUS-D facility, mated to LES, secured, and checked out. Ordnance installation and checkout is initiated and completed in preparation for transporting to and installing the payload in the Vertical Payload Handling Device (VPHD). The adaptation is interfaced to Cargo Integrated Test Equipment (CITE) for cargo integrated test. This test is the pre-Orbiter checkout of the total flight cargo. The interfaces for LES are typical for LES and only minor modifications are required to adapt SSUS-D and LES combined. Assumptions and ground rules identified as applicable to LES/SSUS-D are considered applicable to SSUS-A/LES combination.

A review of expected ground operation requirements for the propulsion module of the MMS and for the TRS identified no significant differences from those identified for the LES.

6.1.2.4 Task Functional Identification - Preliminary timelines for significant major task elements were developed to a level to identify total support requirements. A typical example is presented in Figure 6.6. The timeline of Figure 6.6 is for Tasks 26.0, 25.0, and 29.0 as identified in

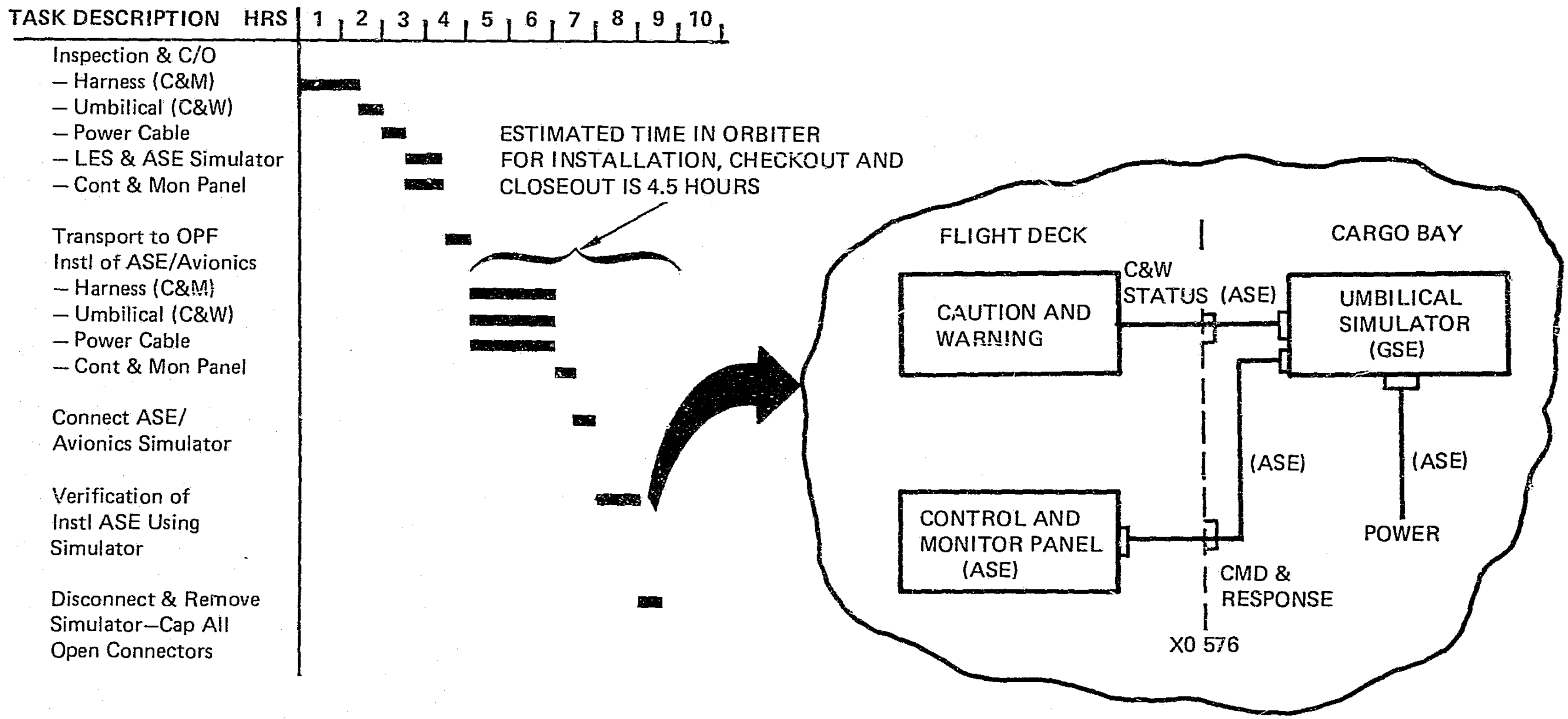


FIGURE 6.6 ASE/AVIONICS CHECKOUT, ORBITER INSTALLATION AND VERIFICATION

Figure 6.4. Included is a simplified block diagram which shows the Orbiter interface checkout arrangement. It is assumed that the payload specialist participates in this checkout as a test conductor. He will provide operational control using the equipment installed in the flight deck. The LES field support consists of equipment, technician, inspector, engineering and other services as required to perform functional verification of flight ASE/avionics in the Orbiter. The same activities and timelines are required for the cargo integrated test equipment flow (Tasks 27.0 and 28.0 of Figure 6.4).

As a result of this timeline evaluation preliminary requirements for equipment, facilities, personnel/skills, test, handling and transporting were identified. Typical examples are:

- General test equipment required for bench acceptance of cable and control and monitor panel.
- Containers required for transporting and storage of equipment.
- Local automobiles required for transporting personnel and equipment from work facility to Orbiter Processing Facility (OPF).
- Facilities space and work benches identified.
- Test equipment selected requires periodic calibration and a controlled environment for storage.
- Procedures for control and standarizing checkout.

The umbilical simulator is used to verify electrical control of LES/ASE launch functions. It simulates electrical loads, caution and warning functions, verifies receipt of commands and provides simulated response. This simulator is portable and may be carried aboard the Orbiter. The umbilical simulator is provided with matching interfaces to all applicable Orbiter bay connectors. Cable harness and logic must allow for checkout with or without LES cradle installation. The conceptual design includes a small microprocessor to control logical sequence tests and voltage load limits. This simulator may also be used to accept the LES/ASE control and monitor panel.

6.1.3 Support Equipment

An analysis of the Ground Operational Flow (Figure 6.4) and a typical LES Processing Timeline when installation occurs at the pad (Figure 6.5) was performed to identify support equipment requirements. These requirements were identified and are listed in Table 6-I under three categories. They are:

- Checkout
- Handling, Transporting and Assembly
- Miscellaneous

The list of equipment was then evaluated for any additional requirements when the timeline allocations for installation at the OPF are used. The results indicate that the list is also typical of the requirements for this flow. Therefore, these equipment are considered representative of requirements to cover both processing flows.

The checkout equipment will provide the capability to functionally verify that LES is operational as it moves through the ground integration and test sequence. Simulators are provided to verify that test sets are operational and to verify interfaces as required during the flow sequence.

The handling, transporting and assembly equipment will provide the capability to move, buildup, assemble and integrate LES and LES components. This equipment also provides the capability, in conjunction with a mobile crane and rental truck, to off-load the cradle assembly at a contingency landing site and return to the launch site. Miscellaneous equipment consists of special tools, test and safety equipment that are unique to LES. This equipment will be provided and available to each task of the processing cycle, both at the factory and field. Provisions for proof loading of handling slings and other load bearing equipment are assumed to be available as GFE.

For the purpose of this study all spacecraft support equipment is assumed to be provided by the spacecraft agency.

TABLE 6-I SUPPORT EQUIPMENT

ITEM	DESCRIPTION	FACTORY REQMT	FIELD REQMT	FIELD REF. FIG. 6.4 TASK NUMBER
	CHECKOUT			
1	Guidance and Control Test Set	X	X	8.0
2	TDY-43 Computer Test Set	X	X	8.0
3	Telemetry Test Set	X	X	8.0
4	Test Battery Simulator	X	X	8.0
5	Pyrotechnic Test Load Simulator	X	X	8.0
6	Thruster Test Load Simulator	X	X	8.0
7	Portable GHe Servicing Cart (with accessories)	X	X	34.0; 5.0; 16.0
8	Audio GHe Spectrometer	X	X	34.0; 5.0
9	ASE/Avionics Simulator	X	X	36.0; 28.0; 29.0
10	Umbilical Simulator	X	X	37.0
11	Cables and Cable Plant	X	X	8.0; 15.0; 17.0
12	Electrical/Electronic Test Equipment	X	X	11.0; 26.0
13	Control and Monitor Panel		X	26.0; 17.0; 22
14	Electro Explosive Devices Test Equipment (GFE)		X	32.0; 11.0
	HANDLING TRANSPORTING AND ASSEMBLY			
15	Shipping Containers	X	X	1.0; 2.0
16	Mobile Flat Bed Assembly		X	35.0; 12.0; 13.0
17	Hoist Sling for Tanks	X	X	4.0
18	Turn Over Hoist Sling for LES	X	X	3.0; 6.0
19	Hoist Sling for Vertical Lift of Payload at the VPF		X	14.0
20	Fork Lift (GFE)		X	1.0; 2.0
21	Truck (GFE)		X	12.0
22	Cradle Assembly		X	35.0
23	Multi-Mission Support Equipment (GFE)		X	14.0; 19.0; 20.0; 21.0
24	Hoist Sling for LES	X	X	1.0
25	LES Handling/Assembly Dolly	X		
26	Hydroset	X	X	4.0
	MISCELLANEOUS			
27	Hand Tools	X	X	All as Required
28	Safety Equipment	X	X	As Required

X - ONE EACH REQUIRED OR AS NOTED

6.1.3.1 Guidance/Telemetry Checkout Capability - An investigation was made to determine the hardware capability needed to support the guidance and telemetry test requirements summarized in paragraph 6.1.1. The guidance system and telemetry system have associated "standard" items of GSE that are either generally available as commercial equipment or fabricated on a standard baseline equipment configuration which will support the LES application. The following discussion relates the capabilities of these proposed equipments and identifies the areas where additional capabilities are required.

The Scout Teledyne TDY-43 computer test set has the features to test the input and output (I/O) of the ISU as well as the functioning of the TDY-43 computer. The system includes a printer so that the test results may be obtained as hard copy for record. This TDY-43 computer test set contains the following items:

- Computer, PDP 11/05-AL with 8K of core memory.
- Cassett Tape Drive, Canberra single drive unit.
- CRT Display, TEC display with 50 characters per line and 20 lines of display.
- Keyboard Control Assembly, TEC terminal keyboard plus 28 function keys for mode control.
- Logic Assembly, Teledyne fabricated circuit board that provides the testing interface to the TDY-43 computer and its I/O circuits.
- Power Supply Assembly, Teledyne fabricated power control and power supply chassis that supplies power to the test set and controls the power from an external source to the unit under test.
- W1 Cable Assembly, TDY-43 test connector cable with buffer and test point box.
- W2 Cable Assembly, Contains cabling from TDY-43 I/O connectors.
- W3 Cable Assembly, Contains the TDY-43 power cable.

With these components, the test set will be able to generally support the following tests:

- Factory System bring-up, initial test of the assembled guidance system in the factory.
- Acceptance Testing, support acceptance test procedures for the guidance system components and the system acceptance test procedure.
- Fault Isolation, trouble-shooting of the TDY-43 computer and other components of the Inertial Stabilization Unit.
- Bench Calibration, collection of data for calibration of the inertial sensors of the guidance system.
- System Bench Testing, provides vehicle simulation sufficient to test the system components supplied by Teledyne.

The TDY-43 test set is not suitable for software generation and modification; however, this capability will be available in a Vought Software Development Center which will consist of the following:

- A CRT Terminal
- A Dual-Diskette Data Storage Unit
- A Printer
- A Telephone Modem for processing the data center CDC6600 computer

The Precision Measurement Equipment laboratory at ETR and WTR is equipped to perform calibration of test instruments required for LES program. Therefore local facilities will be used for calibration of test instruments required for LES.

6.1.3.2 <u>Handling, Transporting, and Assembly Equipment</u> - The capability to integrate, assemble and checkout LES must be maintained at both ETR and WTR. To accomplish this, maximum use of existing and proposed facilities are made. Typical requirements and recommendations are made which provide the capabilities of accomplishing these activities. Typical handling and transporting equipment shown in Figures 6.7 through 6.11 are used individually

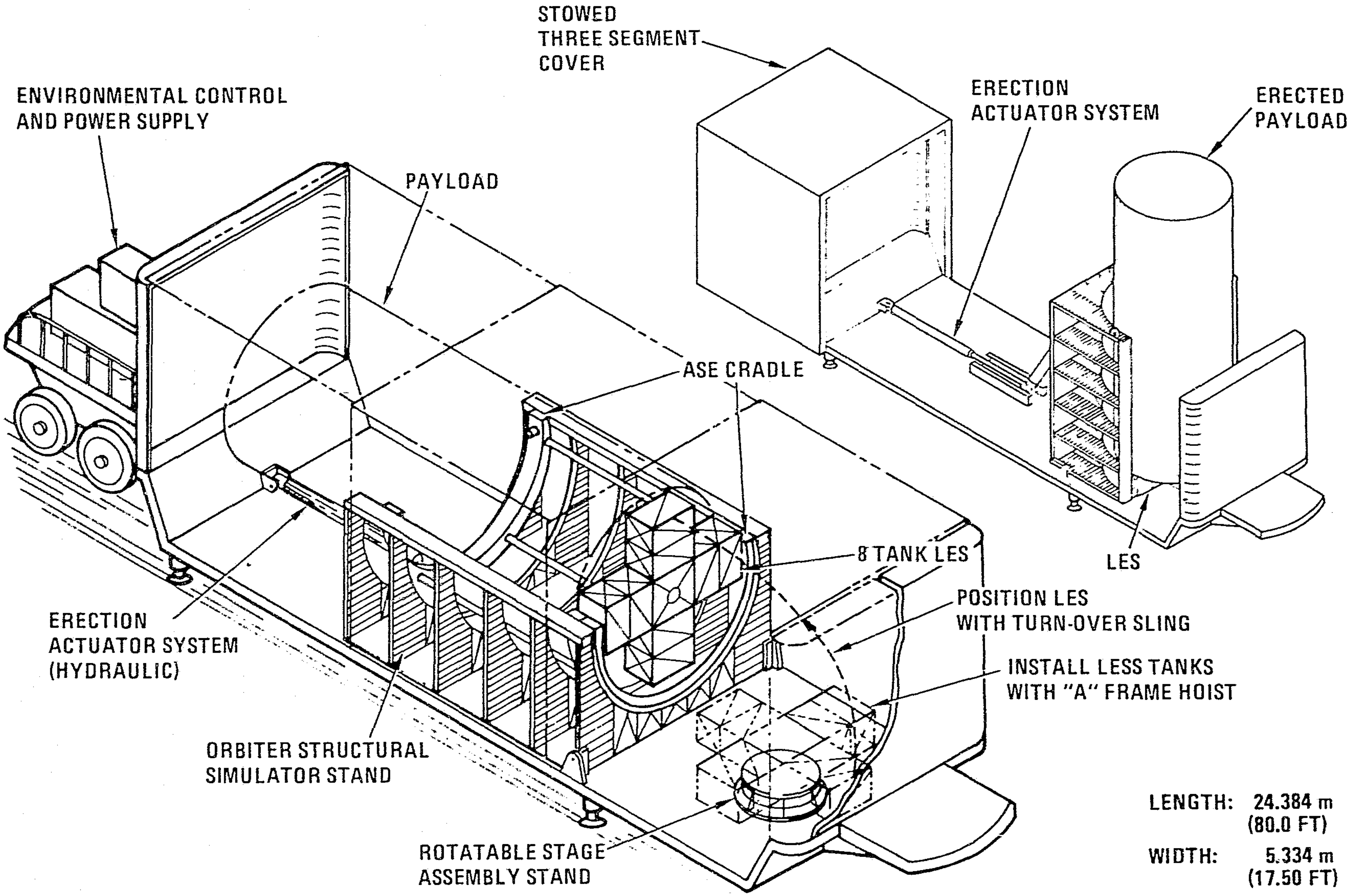


FIGURE 6.7 MOBILE FLAT BED ASSEMBLY

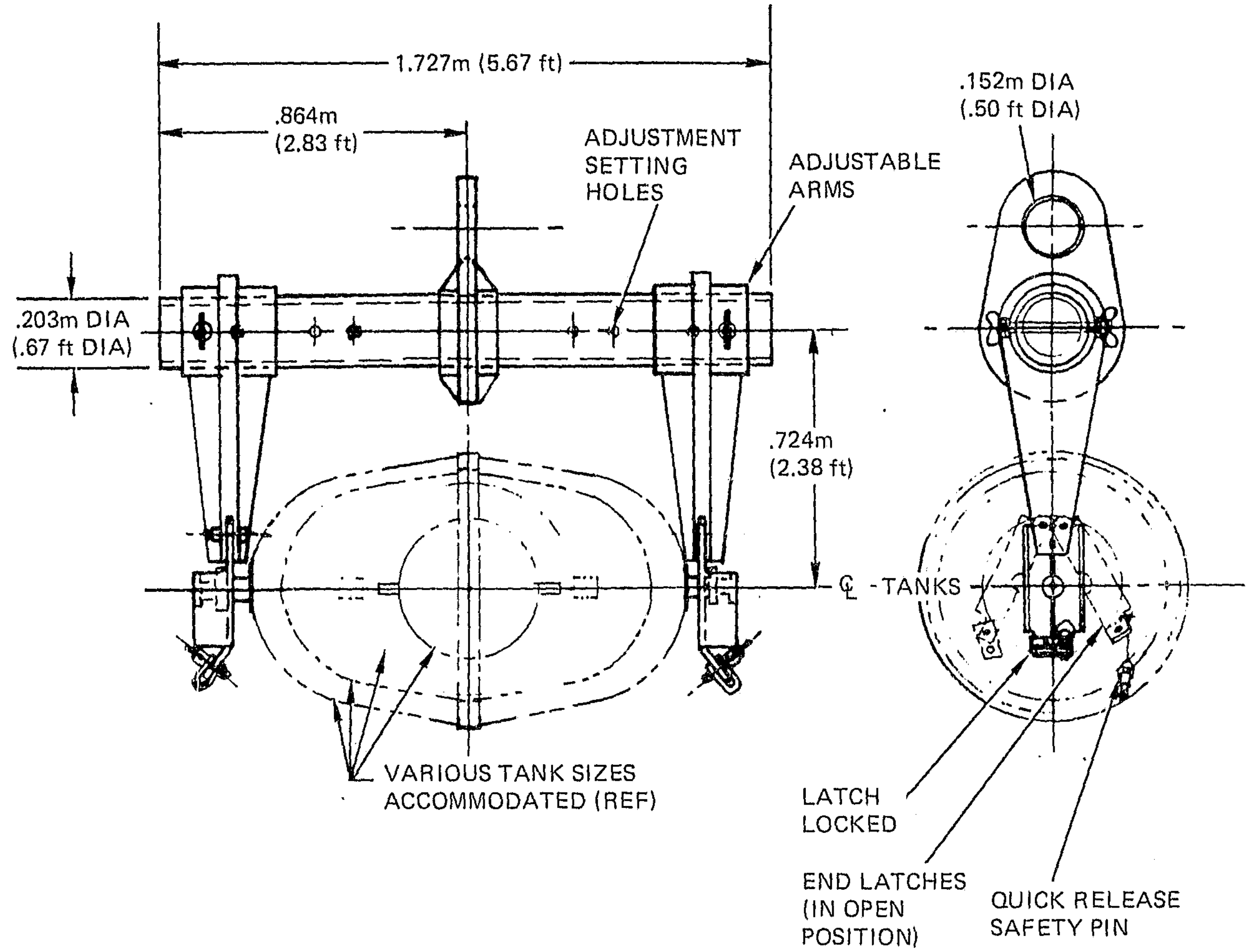


FIGURE 6.8 HOIST SLING FOR PREPACKAGED PROPELLANT TANKS

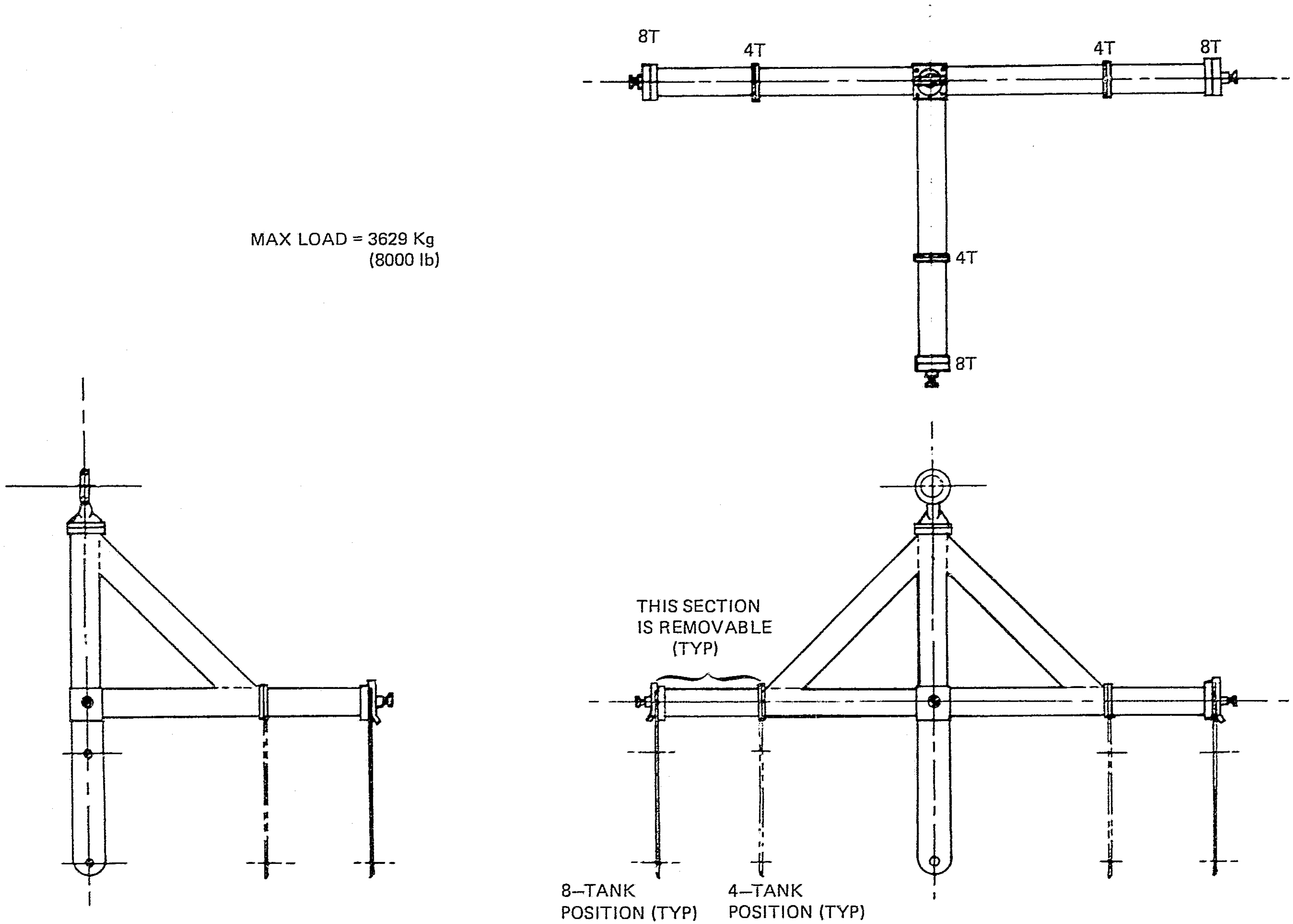


FIGURE 6.9 LOW ENERGY STAGE SLING

MAX LOAD = 4,536 Kg (10,000 lb)

MAX STAGE ENVELOPE:

LENGTH = 1.0m (3.3 ft)
DIAMETER = 4.115m (13.5 ft)

MAX PAYLOAD ENVELOPE:
(WHEN MOUNTED ON STAGE)

LENGTH = 1.8m (5.9 ft)
DIAMETER = 1.5m (4.9 ft)

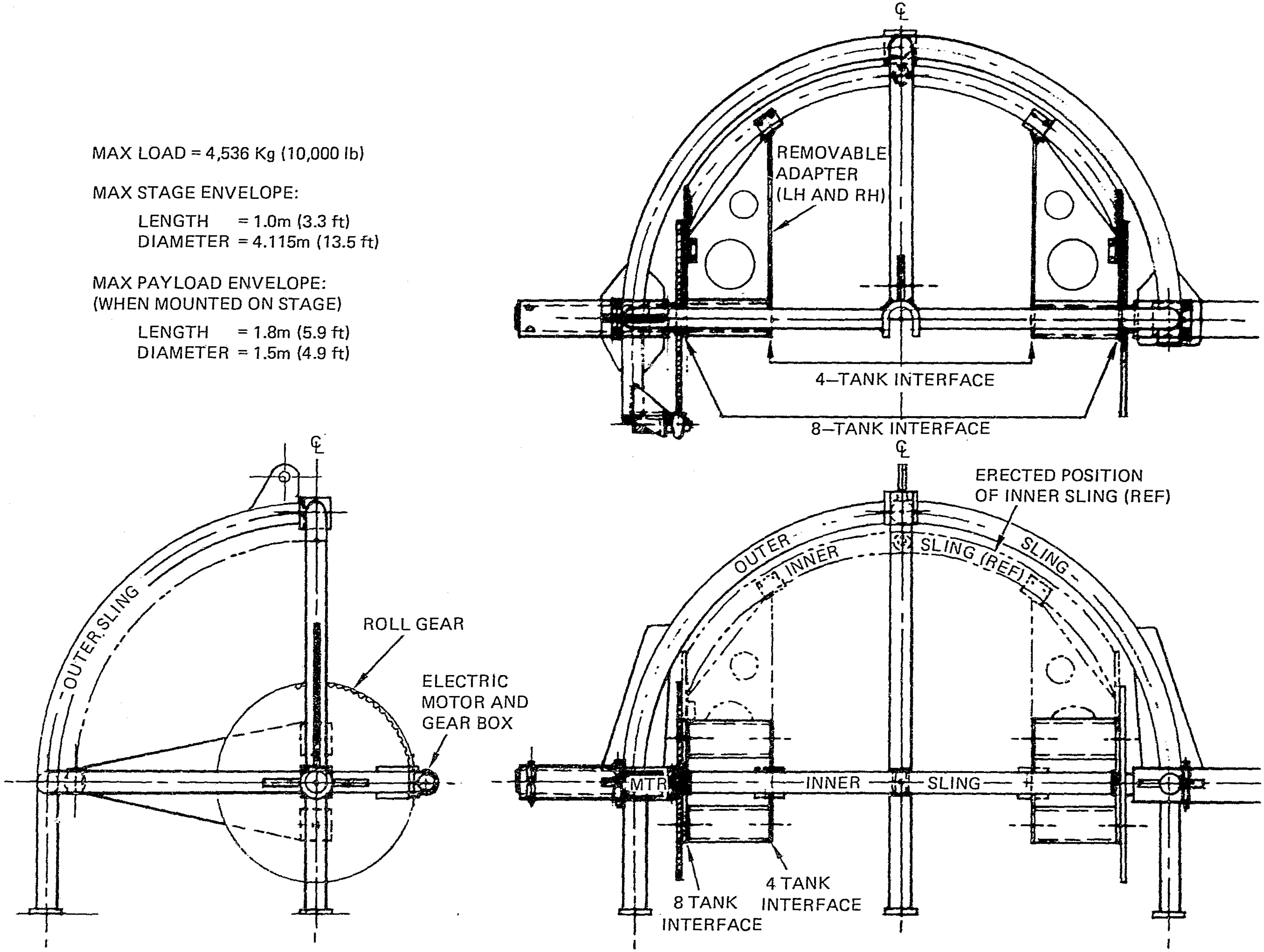


FIGURE 6.10 TURN-OVER HOIST SLING

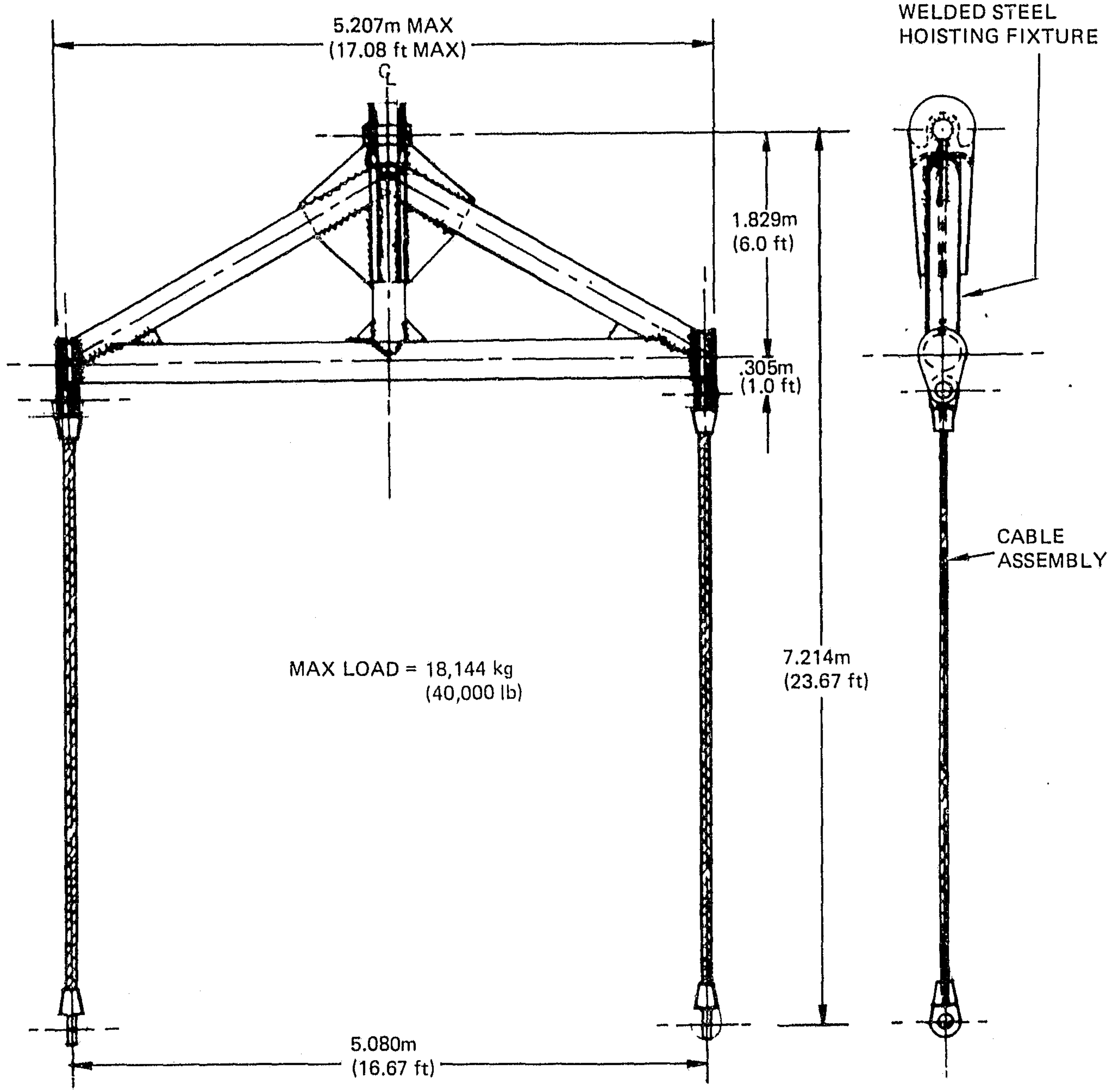


FIGURE 6.11 VERTICAL LIFT HOIST SLING

and in conjunction with Multimission Support Equipment (MMSE) to provide the capability of performing functions such as receiving inspection, assembly, transporting and mating LES, LES/cradle assembly and LES/cradle assembly/spacecraft. Figure 6.7 shows the Mobile Flat Bed Assembly (MFBA). This assembly provides a mobile buildup and integration transporter for the Orbiter payload. Figure 6.8 shows the hoist sling concept for removing, handling and mounting the prepackaged propellant tanks. Figure 6.9 shows the hoist sling for removing LES from its reusable shipping container and placement on the MFBA turn table. Figure 6.10 shows the turn over hoist sling for lifting a complete assembled LES from the turn table on the MFBA, then rotating the stage assembly from the horizontal position to the vertical position and transferring it to the cradle assembly. Figure 6.11 shows the vertical hoist sling used for lifting the Orbiter payload and Orbiter structural simulator stand from the MFBA to the vertical payload handling device. After transfer of Orbiter payload the sling is used to lower the structural simulator back to the MFBA. These figures are typical of concepts defined for cost in Table 7-V of Volume IV.

The configuration of LES and cradle assembly is interfaced with the Systems Test Sets and simulators for systems tests. After system tests the spacecraft is mated using a spacecraft provided sling. Upon successful mating and securing of the spacecraft systems, tests are completed. The next handling activity occurs when all tests and ordnance installations are completed. After securing Payload and MFBA, a GFE truck is attached to the MFBA for transporting the payload to the VPF. The payload is elevated to the vertical position with the self-contained erector actuation system. A hoist sling attached to the overhead crane provides the lifting capability for interfacing the payload with the VPHD. Upon completion of this activity, the remaining handling and transporting task uses MMSE such as the canister, canister flat bed, Payload Change Out Room, and Payload Ground Handling Mechanism.

6.1.4 Hazard Operations

Preliminary analysis indicates that certain tasks during ground processing and in-Orbiter flight operations will be classified as harzardous. Table 6-II identifies these potential hazards with comments.

TABLE 6-II GROUND AND FLIGHT OPERATIONS HAZARDS

OVERPRESSURE:	HELIUM PRESSURE TANKS . Designed to withstand total pressure from environment . Provided with squib valve on-board venting BATTERY . Design to withstand maximum pressure expended with safety margins FUEL TANKS . Tanks not under pressure until activation of RCS after separation at T + 40 minutes
CONTAMINATION:	FUEL LEAKAGE TO PAYLOAD ORBITER/THRUSTERS . Use preserviced fuel tanks . Use brazed or welded fittings . Isolate fuel with squib valve to maintain metal-to-metal seal . Fire squib valves after deployment . Avoid tank rupture by design margins
OVERHEAT:	FUEL TANKS . Provide insulation to control tank environment . Provide proper ullage HELIUM PRESSURE TANKS . See Overpressure THRUSTERS AND LINES . Charge lines and thrusters after deployment BATTERY . For battery overheat transfer to external or deploy payload - flight only
PREMATURE SQUIB/ IGNITION FIRING:	ELECTROEXPLOSIVE DEVICES TYPE NSI-1 . Fail safe design - requires three failures to fire . Ignition system not armed until after separation at T + 30 minutes

6.1.5 Integrated Interfaces

A series of factory and field validation tests are proposed on the flight LES hardware, GSE, ASE, ETR and Orbiter interfaces equipment. These activities will be performed in a logic sequence from hardware fabrication through a field compatibility demonstration test. The end result will be to certify the interfaces and readiness of the GSE, ASE, Procedures, and Facilities for operational missions.

6.1.5.1 Factory - Typical major factory activities consist of qualifying new hardware designs and procedures and performing an integrated LES system test. Some of these activities are as follows:

Stage Structure - A structural load test is to be performed on the stage structure to demonstrate its capability to withstand the expected launching loads. The assembled stage structure with mass simulated components installed will be mounted in a structural test jig. The critical points on the stage structure will be instrumented to measure strains and deflections and the loads applied by hydraulic jacks. A combination of axial shear and bending loads for the critical design will be applied in increasing increments until the design ultimate load is achieved. At each increment, measurements of strain and deflection will be recorded and this data will be examined to verify structural design analysis. The stage structure will be tested to failure. It will not be available for compatibility demonstration, however, another structure used for environmental testing will be available for demonstration.

Payload Adapter - A structural load and stiffness test, vibration and mechanical shock test and a separation test is to be conducted on the payload adapter. The structural load test will demonstrate the capacity of the adapter to withstand the expected transporting, handling and flight loads, and will provide data to determine the stiffness of the adapter for use during environmental testing. This test will be similar to the one conducted on the stage structure. The section will be instrumented with strain gages and deflectometers and will be subjected to a combination of axial shear and bending loads applied via hydraulic jacks from zero to the expected transporting, handling and flight loads. Instrumentation data will be recorded at each load step and subsequently analyzed to verify the structural design analysis and the expected stiffness of the adapter.

An environmental test of the adapter will be conducted by exposing it to vibration and mechanical shock. The base of the separation system will be attached to the vibration exciter system with a simulated payload attached at the forward end. This simulated payload will have mass characteristics similar to an actual flight payload. A resonance survey will be conducted to determine the resonances of the structure and response associated with each. Vibration testing follows with inputs to induce the expected dynamic flight loads. Functional monitoring requirements will include accelerometers to obtain transmissibility data, strain gage output measurement, and electrical continuity measurements of the interface connectors during environmental exposure.

A separation test will be conducted on the payload adapter. This test will simulate separation of the payload from the adapter section by allowing a mass simulated payload to eject and free-fall from a constrained adapter section. Two tests will be conducted using the flight explosive separation system while the third test will be conducted with one of the redundant systems disabled.

A structural proof load test will be performed on all new mechanical load bearing GSE. This test will be conducted at twice the design operating loads and will verify overall structural integrity prior to final use.

6.1.5.2 Field - The major field activities at ETR consist of contractor equipment interface tests, proof loading handling/transporting equipment, and a demonstration/validation ground test.

Site Integration - As part of the initial site activation and prior to mating the LES with the Orbiter, the Orbiter mock-up simulator will be used to functionally verify the LES ASE avionics and cradle assembly equipment interfaces. Simulators will also be used to functionally test the system test equipment located in the SAEF-2 or other facility as determined by availability of facilities. Prior to first use, and periodically thereafter, all mechanical load bearing equipment will be proof loaded to twice its design load.

Validation Test - A demonstration/validation test is proposed for LES as the final step to certify flight readiness for operational missions. The LES will be a fully operational configured system except inert tanks will be installed. The LES will be the same one that was used for integrated system factory test consisting of equipment required for flight sequencing and control, telemetry and those systems necessary to validate functional mechanics of the operational LES. The LES system and cradle assembly are to be configured with additional instrumentation to measure handling and transporting environmental compatibility. This integrated demonstration could be a stand-alone test, depending on the LES configuration and facilities availability, and primarily would be used for demonstration and validation in the following ways:

- Verify physical interfaces of the LES/ASE and Prototype Spacecraft or dummy Mass P/L with the ETR receiving, handling, assembly, transporting, VPF, canister loading and PCR.
- Verify processing timelines-LES, ASE, P/L and integrated payload.
- Verify processing operational procedure documentation.
- Verify range facilities, hardware and support service and interfaces; Telemetry, Communications.
- Verify LES operational interfaces with Checkout Equipment, Cargo Integrated Test Equipment, facilities, and safety.
- Verify timeline allocations.
- Verify environment compatibility.
- Verify range instrumentation interfaces.
- Verify post flight operations - refurbishment and post flight data interfaces.

Operational LES - All efforts, including LES, ASE and GSE design, analyses and fabrication, ground tests and checkout, system integration, and demonstration tests are directed toward achieving successful operational flight tests.

6.1.6 Support Requirements

All prospective users of National Range Services or facilities must submit their requirements via the Universal Documentation System (UDS). The UDS is a standardized documentation system that is accepted and used at all National Ranges and certain Service Ranges. It provides a formal, common method of language and format for stating requirements and preparing support responses. This system is the means by which the ranges formally accept such requirements and prepare official plans for support of users' program.

For the purpose of this study certain range services such as telemetry, communications, facilities and the like were estimated using past programs as a guide. These cost are identifiable in Table 7-VI of Volume IV.

Site activation will include administrative activities, coordination of facilities and services, receipt and checkout of support equipment and implementation of the off-site spares program. Integration of the LES program to an off-site base will probably consist of two major efforts. These are:

- Integration of users administrative activities. This involves establishing personnel procedures, orientation of new personnel to range requirements and regulations and establishing lines of communication with range support elements.
- Integration of system operational activities. This involves development of safety procedures consistent with range requirements, scheduling of assembly and test operations requiring range support to meet the availability of the supporting elements and performing compatibility checks (both mechanical and electrical) to ensure system interface.

Spares and repair requirements are divided into two categories:

- Those functions occuring in-plant for an operational program.
- Those functions required for off-site support.

For the purposes of this study, past programs were used to identify percentage spares/repair requirements on a per unit operational cost.

Preliminary requirements for a training program were estimated. These included task and skills analysis to define the various LES related tasks to be accomplished and the skills required to perform those tasks of preparation, integration, installation and checkout. It also included maintenance related tasks as required. The payload specialist was included as part of this training program. However, a detailed study will be required to determine and define the various related payload tasks and training equipment requirements and development of training literature to conduct a new training program.

Each task of Figure 6.4, along with the timelines of Figure 6.5 were evaluated for manpower and skill loading at the field site base. A typical field team manned with the proper cadre of personnel/skills is estimated to be 18. These personnel will have the capability to provide the necessary services and expertise to assemble, checkout and prepare the LES/cradle assembly/GSE and ASE and to assist in the total integration of payload in the Orbiter cargo bay for deployment and its intended mission. This cadre of personnel includes administrative, engineering, technicians and inspectors. The mixture of skills shown in Table 6-III provides the proper number and skills to perform scheduled prelaunch efforts and to maintain the site equipment in an operational condition.

TABLE 6-III PRELIMINARY PERSONNEL REQUIREMENTS

SKILL	NUMBER
SUPERVISOR	1
ADMINISTRATOR	1
ENGINEER	2
OPERATIONS ENGINEER	2
TECHNICIAN	8
INSPECTION	2
QUALITY	1
LOGISTICS	1
TOTAL	18

The field team will also provide services as required to define mission integrated requirements for field prelaunch and launch functions that meet operational objectives within the constraints imposed. Additional launch support from the home plant is estimated at 30 man days per launch to provide technical and monitoring services.

6.1.7 Support Facilities

The facilities for processing LES, LES ASE and spacecraft are shown in Figure 6.12. It is assumed that ground safety approval for use of these facilities will be provided or similar facilities suggested as alternates.

The hub of payload activities is planned for the Spacecraft Assembly Encapsulating Facility (SAEF). If this facility is not available, an alternate facility such as a hanger with suitable space for LES Test Sets, Mobile Flat Bed Assembly and other GSE/ASE will suffice. In either case the facility assigned for this program is to be designated as an Ordnance Payload Assembly Facility (OPAF). With this designation ground safety approval for handling the preserviced fuel tanks and performing ordnance installation is possible. Certain facility modifications may be required to adapt the facility for LES program support equipment and to accommodate the spacecraft. These could include facility modifications to provide work space, enclosed area for test sets, storage for equipment, clean room requirements, and spares or repair parts.

The Explosive Safe Area (ESA) provides storage and a testing area for Electro-Explosive Devices (EED's). It is planned that all EED's assigned to LES will be bench tested in this facility with ground safety approval using government supplied test equipment. In addition, the squib activated batteries are stores in either the ESA or in the spares storage area until ready for use.

The Hypergolic Maintenance Facility (HMF) may be required to receive, inspect and store preserviced fuel tanks. As a normal flow, the preserviced fuel tanks, with ground safety approval, could go direct to the LES assigned OPAF for uncrating, inspection and installation. If flight schedules are such that more than one set of tanks are required on site, then the HMF is assumed as the receiving and storage area for these tanks until ready for use.

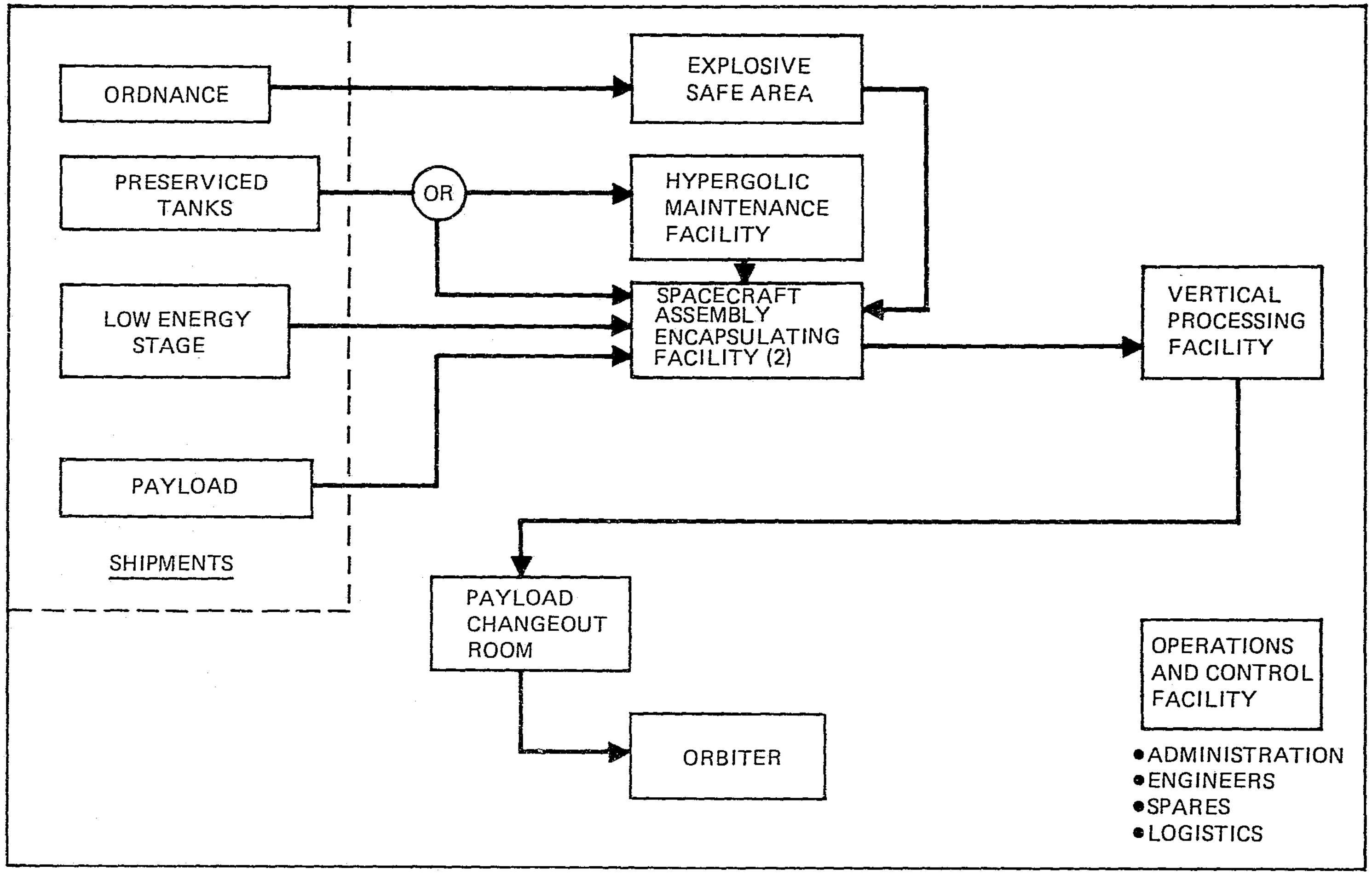


FIGURE 6.12 SUPPORT FACILITIES

The Vertical Processing Facility (VPF) is common to all payloads integrated into the Orbiter cargo bay standing up. This facility is assumed to be capable of accepting LES Mobile Flat Bed Assembly for transfer of payload to the Vertical Payload Handling Device (VPDH). Work platforms to the 55 foot level are provided for interface work and preparations for Integrated Test using the Cargo Integrated Test Equipment (CITE). The CITE , when provided with installed payload ASE/Avionics equipment, provides the capability of performing a preflight readiness checkout of each payload as a totally integrated cargo prior to installation and interface with Orbiter payload accommodation equipment. The flight payload cargo is transferred to the canister and secured for transporting to the Payload Changeout Room (PCR).

The PCR accepts the flight cargo from the canister in a vertical position (standing-up). The transfer of the cargo from the canister to the PCR is made using the Payload Ground Handling Mechanism (PGHM). When the PGHM is in the retracted position work platform space is provided for payload related equipment. When the payloads are transferred and installed in the Orbiter cargo bay personnel and equipment access platforms are available as required to perform payload interfaces, adjustments, and servicing. There is no planned checkout activity for LES while in the PCR.

When a payload is installed via the optional route (laying down) the Orbiter Processing Facility (OPF) enters into the payload flow and the VPF and PCR are bypassed. However, the CITE verification will remain as a required interface test in the flow. After this test the flight payload cargo installed in the canister is transported to the OPF. The flight cargo is transferred from the canister to the Orbiter cargo bay laying-down using the MMSE Strongback.

Operations and Control Facility (O&CF) or other suitable space is required for administrative, engineering, logistics, etc. personnel. Also a controlled space with environmental control will be required to stock minimum spares and repair parts for the LES program.

6.2 FLIGHT OPERATIONS

During the Orbiter launch flight phase to on-orbit operations the LES is off and safe. Selected caution and warning candidates are monitored with provisions provided to recover from an unsafe condition, should it occur.

On-orbit operations consist of preparations for and performing predeployment checkout and deploying the Orbiter payload for its mission.

After deployment, at separation the mission flight sequence begins. Telemetry coverage of the flight is provided by the Orbiter within its range capability. Additional coverage beyond that of the Orbiter will be by ground tracking stations to obtain flight performance data to spacecraft separation and insertion into orbit.

Upon completion of mission assignment the Orbiter prepares for re-entry and landing. After landing the Orbiter is towed to the OPF for recovery operations.

6.2.1 Flight Decision Sequence

On-orbit operations are controlled and monitored from the flight deck by the Payload Specialist. He monitors the "safe" status of the payload from launch to separation. Provisions for recovery from an unsafe condition will be documented in flight procedures. The Payload Specialist will control and monitor predeployment tests and deployment of the LES and Orbiter payload. He will also enter and verify flight trajectory data, enter time of launch and initiate the final deployment sequence which results in separation from the cradle assembly and Orbiter. If an anomaly occurs during predeployment tests, a decision sequence will be followed similar to that shown by Figure 6.13. Even though it will be possible to fault isolate LES to the replacement box level, it will not be possible nor practical to replace the box for two reasons. These are:

- No spares are carried on the flight, and
- Orbiter payload would probably have to be deployed by the RMS and the black box replaced using EVA, if a spare were available.

Neither approach is acceptable nor is it considered safe. Therefore, fault isolation will only provide self test within the capability of the Built-in-Test to verify that the LES does not pose a threat to the safety of the Orbiter or crew and a logical decision can be made to either return or dump the LES and/or spacecraft. However, if the anomaly should be isolated to the payload accommodations equipment, specific instructions will be available in

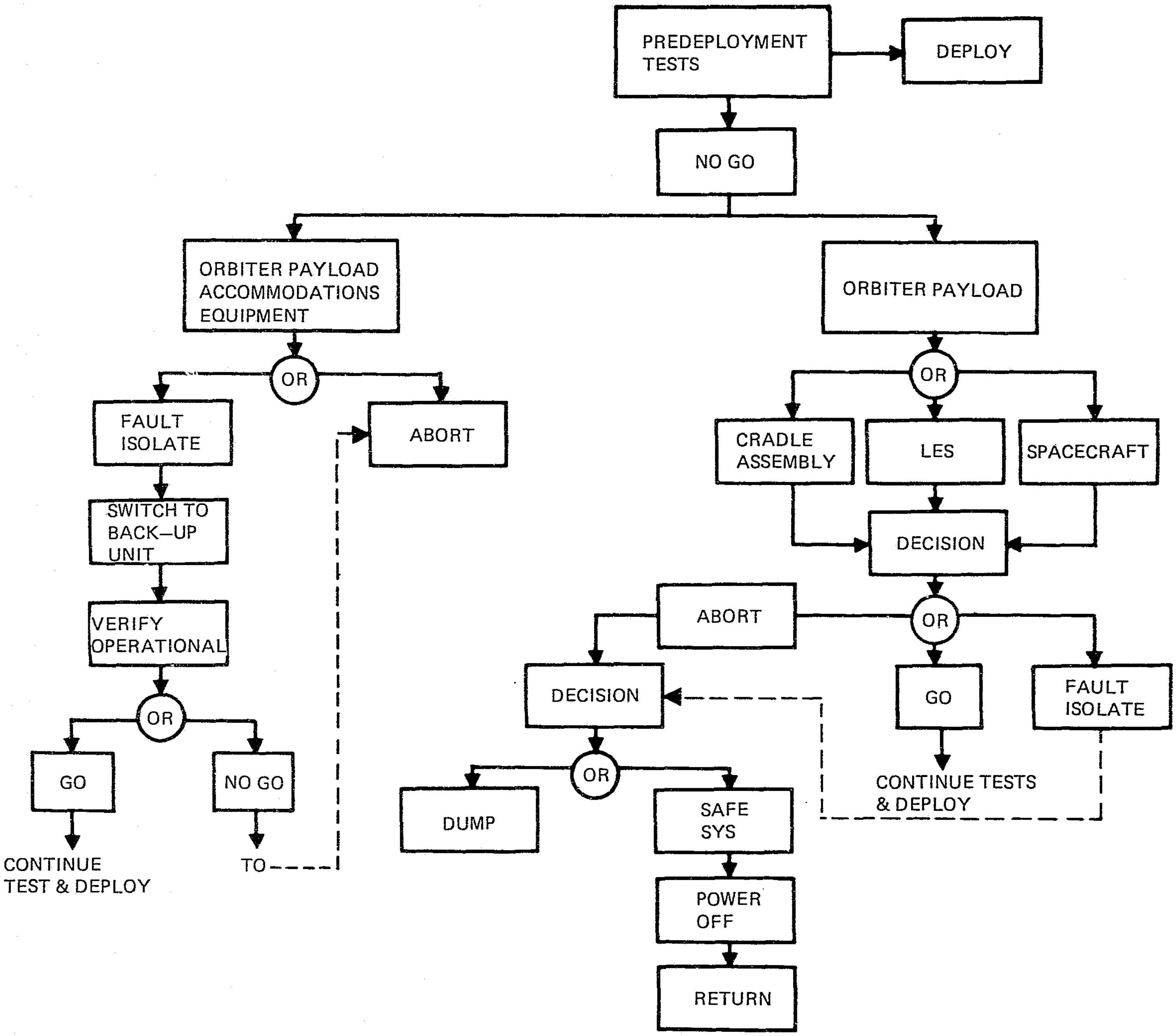


FIGURE 6.13 GENERAL FLIGHT DECISION SEQUENCE

documented procedures that allow an orderly recovery by switching to backup systems or subsystems.

6.2.2 On-Orbit Operation

With the Orbiter payload integrated in the cargo bay and electrically interfaced to the ASE avionics and payload accommodations equipment, the pre-launch predeployment checks are performed. These checks include the Inertial Stabilization Unit (ISU), telemetry and selected power switching functions. The flight battery, Ignition Control Unit (ICU), squib and motor circuits are not included as part of these tests. The squib and motor circuits remain in the "safe" condition because the ICU is inhibited and electrical power is kept off the ICU power bus until just after separation from the Orbiter. After Shuttle orbit has been established, pre-launch activities are initiated. A simplified block diagram of these activities is shown in Figure 6.14. The 3-axis stabilization is accomplished by the digital computer and guidance platform of the ISU. The guidance platform provides 3-axis attitude and directional reference data for flight trajectory control and position data for control event sequencing. Self test (BITE) of the system is continuous after power application. Safe status monitoring via the caution and warning electronics is continuous from the preflight readiness checkout up to umbilical disconnect.

Once the Orbiter payload predeployment checkout is complete and the status is verified to be in a GO condition and the Orbiter is ready, the decision will be made to deploy the LES/spacecraft. Upon separation the deploy switches actuation enables the ICU so that event sequencing can occur for squib firings at predetermined times.

6.2.3 Mission Sequence

After T = 0 the LES entire active flight is conducted under the ICU computer control. After ignition arm, which occurs at T + 30 min, the squib firing sequence is initiated. A typical mission sequence of events is shown in Table 6-IV. Following a time delay of approximately 40 minutes, to allow the LES and spacecraft to travel to a safe distance away from the Orbiter, the Reaction Control System (RCS) is activated. The LES and spacecraft are then oriented to the attitude for perigee burn by the computer. A typical

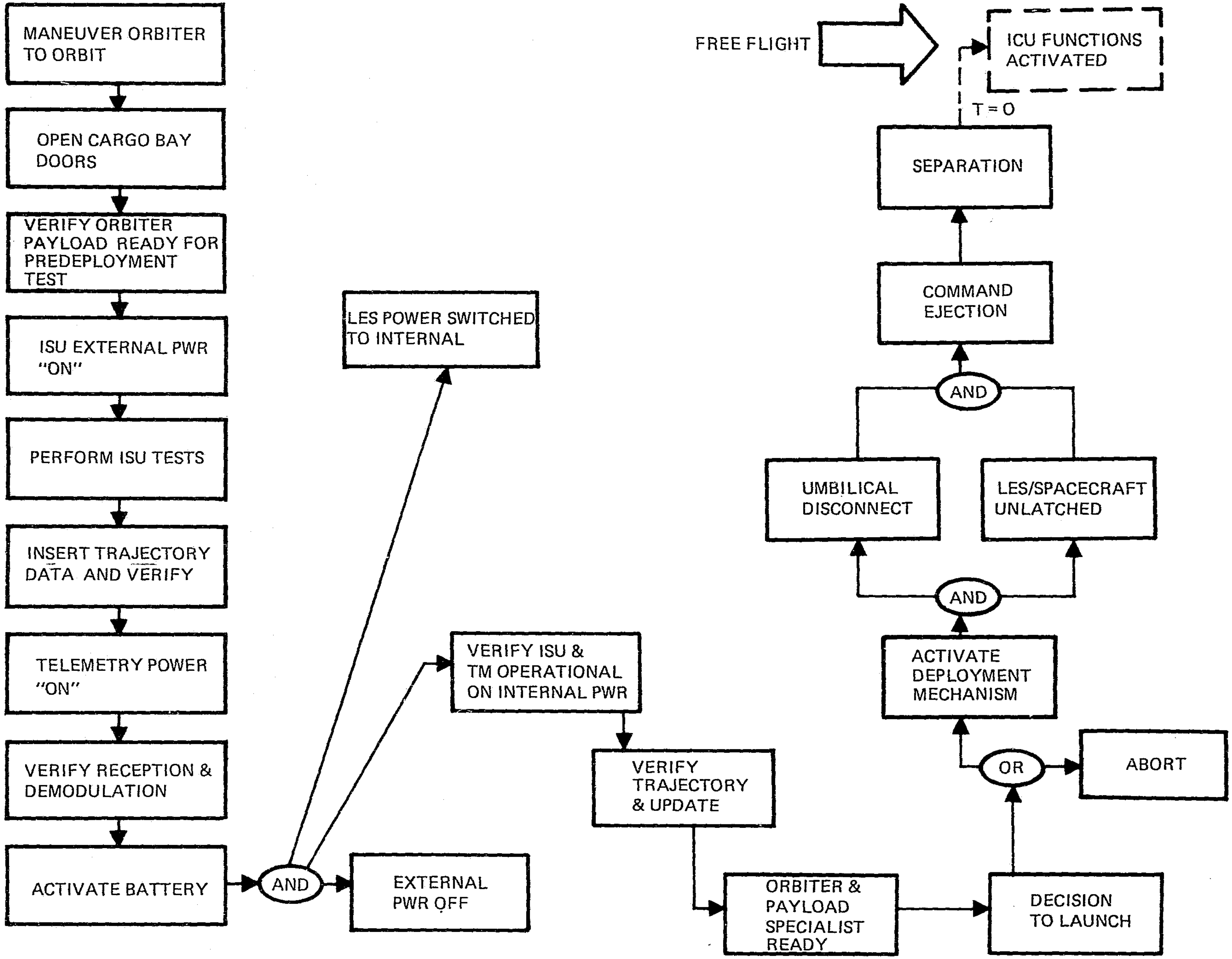


FIGURE 6.14 ON ORBITER PAYLOAD OPERATIONS

TABLE 6-IV SEQUENCE OF EVENTS - TYPICAL MISSION

TIME (MIN)	EVENT
-15	Warm-up Guidance System and Insert Flight Software
0	Deploy at ΔV = 1-2 fps - Guidance On - RCS Off
30	Arm Ignition System
30.5	Fire Pressure Tank Valves
31.0	Fire 1st Set of Fuel and Oxidizer Outlet Tank Valves
31.5	Fire 2nd Set of Fuel and Oxidizer Outlet Tank Valves
32.0	Fire 1st Set of Fuel and Oxidizer Tank Inlet Pressure Valves
32.5	Fire 2nd Set of Fuel and Oxidizer Tank Inlet Pressure Valves
40	Activate RCS and Orient Attitude for Perigee Burn
45	Start Perigee Burn
60	Complete Perigee Burn (RCS Off)
100	RCS On - Reorient Attitude for Apogee Burn
105	Apogee Burn
119	Complete Apogee Burn
120	Spacecraft Separation

separation velocity of 1-2 fps will allow sufficient separation distance in 45 minutes to avoid possible contamination to the Orbiter, Orbiter cargo and bay areas. At T + 45 minutes perigee burn occurs and lasts approximately 15 minutes. The computer then reorients attitude for apogee burn at T + 105 minutes. Apogee burn is completed at T + 119 minutes at which time the spacecraft is separated by firing the separation explosive bolts. Separation clearance is provided by spring energy of the spacecraft "V" clamp adapter.

6.2.4 Abort and Recovery

6.2.4.1 Abort - Abort considerations during the ground processing cycle are considered to be few, if any, since the occurence of problems at this level are considered to be those that are recoverable with no significant impact on Shuttle processing timelines. Each problem that occurs will be evaluated and corrective action taken, on an overtime basis if necessary. The overtime scheduled provides time to correct minor malfunctions and allows transition back to normal scheduled Shuttle timeline commitments. Fault isolation to component (block box) replacement provides this flexibility. Once the LES and spacecraft are installed in the Orbiter and interfaced with ASE avionics and Orbiter equipment, replacement of components is not practical due to accessibility and the fact that ordnance devices are installed. Therefore, abort considerations will probably involve on-the-spot decisions based on an evaluation of the problem, its impact on the Orbiter, Orbiter cargo, the criticality of the launch window, etc. If the problem involves the safety of crew and/or Orbiter, abort procedures will apply. Upon completion of LES preflight readiness checkout, LES is de-energized and placed in a safe mode. The safe status, if required, will be monitored on the Caution and Warning Electronics (CWE). CWE can provide continuous monitoring of candidate LES C&W functions up to separation from the Orbiter. If an abort is indicated by an unsafe condition as noted on the CWE, and the condition cannot be safed, the Shuttle abort procedures are used. Typical abort considerations during on-orbit operations will be system failure, inadvertent actuation of ICU and/or initiators, battery fails to activate, deployment mechanisms fail to operate, no power transfer, umbilical disconnect fails, and separation fails to occur. Each abort consideration will be defined in procedures so that immediate action may be taken to protect and avoid any possibility of creating a catastrophic hazard to crew and Orbiter.

6.2.4.2 Recovery - Provisions for personnel and support equipment are provided for recovery operations. The two modes of recovery include: (1) the return of the Orbiter payload or (2) the return of only the ASE avionics and cradle assembly. If the on-orbit operations are unsatisfactory, for whatever the reason, and the decision is made to safe and return the payload, the ground crew is notified and preparations are made for recovery. The schedule of recovery operations shown in Figure 6.15 begins with the towing of the Orbiter to the OPF and safing and deservicing of the Orbiter. At the completion of this task, access is provided for the LES and spacecraft teams to move in with support equipment and prepare for removing the payload. The spacecraft team removes the spacecraft after safing operations are complete. With the spacecraft clear of the Orbiter and removed from the OPF, the LES/cradle assembly/ASE avionics previously verified safe are removed. The

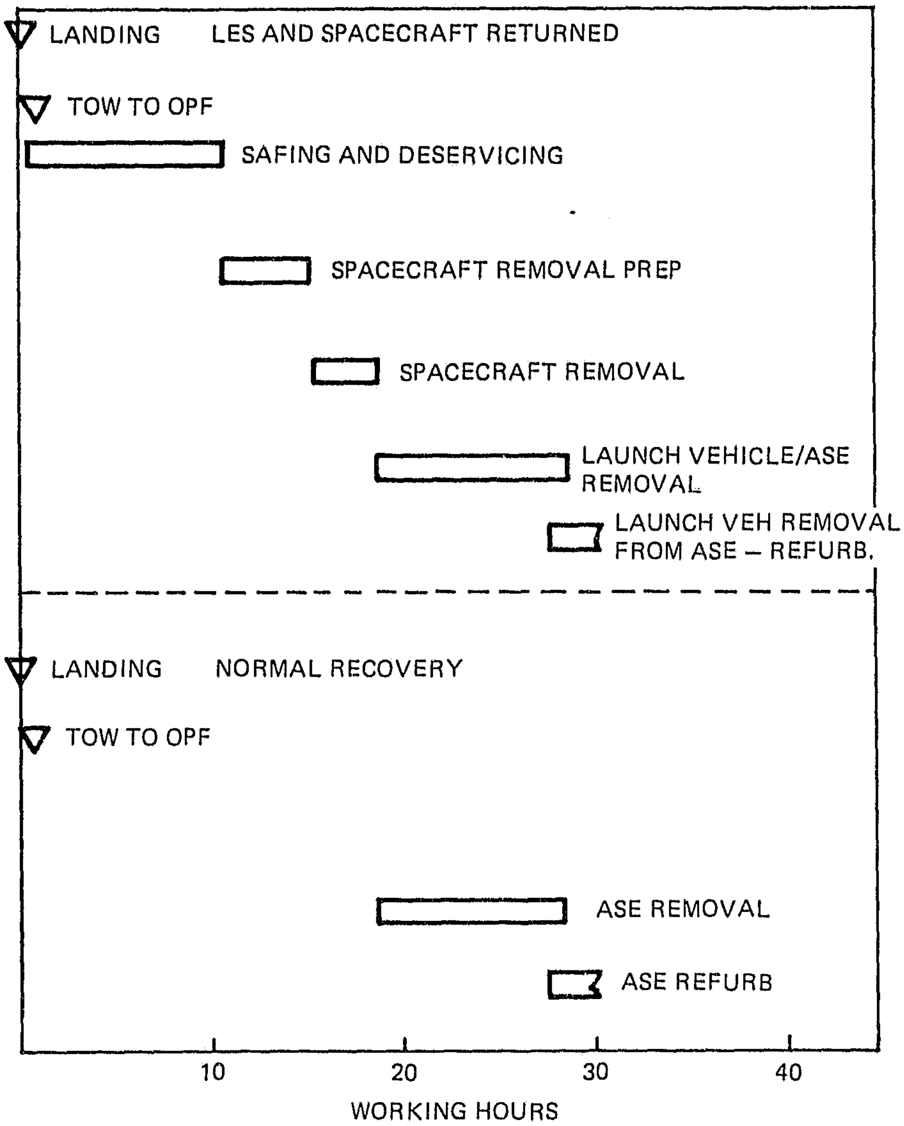


FIGURE 6.15 RECOVERY OPERATIONS

LES/cradle assembly is placed on the mobile flat bed assembly using LES-furnished slings, secured and transported to LES processing facility for refurbishment.

When normal recovery is made, only the ASE avionics and cradle assembly are returned. The removal flow is the same as that for removing the LES/cradle combination except no safing is required. The cradle assembly and ASE avionics are removed and transported to the processing facility for refurbishment and stored for the next flight.

6.2.5 Safety

A safety analysis was conducted on the LES to identify hazards during test, handling and transporting. Typical results are as follows:

- Electro-Explosive Devices (EED's) - The hazards associated with EED's are a result of possible inadvertent actuation. Prevention of an inadvertent actuation is assured by applying proven safety principles in design as well as procedures. Protechnic devices are designed in accordance with applicable standards. The NASA type NSI-1 initiator meets these standards. Initiator leads are shorted and grounded during handling and up to final bench test. After installation they are shorted via the Ignition Control Unit and remain shorted during the CITE, preflight readiness, and predeployment test. The RCS and motor firing circuits are inhibited and shorted. In addition to shorting the initiator firing leads, the Ignition Control Unit (ICU) controls the actuation using the principles of independent arming and firing switches. Means are provided to test the ICU for proper operation prior to installing and connecting the EED's.

- Pressure System - The possibility of an inadvertent release of gas under pressure creates the potential hazard in the gaseous helium fuel pressurization system. Design principles and safe operating methods are applied to prevent the likelihood of this occurring. Design margins for proof pressure and burst pressure are used to insure adequacy of the pressurization system. An inert gas is used to eliminate chemical reaction with the propellants. Furthermore, the system is designed using a relief valve which will prevent leakage from overpressurizing the plubming. Standard safety practices are followed during servicing of the pressurization system to assure protection against overpressurization and potential personnel hazards. Also, warnings are to be incorporated in the procedures and test plans to advise operators of the potential hazards.

- Electrical - The LES electrical system presents the potential hazard of electrical shock and the chemical hazard of the battery electrolyte. Protection from release of corrosive or toxic fluid from the battery is provided by sealing the battery. Electrical shock is avoided by using safe operating principles and by insulating or protecting the hazardous areas. The LES utilizes single point ground to minimize ground-loop hazards.

- Handling/Transporting - Potential hazards during handling and transporting are a result of LES mass and momentum. The hazards are minimized by using only equipment that is rated for the LES and LES payload weight by using approved safety procedures, and by periodically proof loading the LES and LES payload handling and transporting equipment.

- Environmental - Proposed materials for the LES have been examined for toxicity and unusual off-gasing and combustion products. The information will be provided to personnel and in a test plan concerning these materials.

- Radiation - Potential radiation hazards at levels that cause injury do not exist.

- Prepackaged Propellants - The recommended safety instructions for hydrazine and monomethylhydrazine from AFM 161-30 will be followed.

- Battery - The safe wet stand time for the battery is estimated to be eight hours. If the mission should be aborted after battery activation discharging provisions are provided to prevent overheating and the possibility of explosion. The electrical interfaces to perform this discharge function are provided via the umbilical and ASE.

6.3 PROPULSION CONCEPT ASSESSMENT

During the conceptual design certain features were identified that provide a more efficient and reliable operations processing concept. These features include:

- Access from rear of the LES for installing system components and initiators.
- Modularized LES structure to simplify installation of preserviced propellant tanks.

- Provisions for quick disconnect to plumbing for pressurizing and leak surveillance check.
- Provisions for single umbilical interface.
- Provisions for mounting the LES flat (laying down) on the Mobile Flat Bed Assembly turntable to facilitate installation of preserviced propellant tanks.
- Provisions for a positive ground system on the insulating blanket and a common single point ground to LES.
- Provide a dummy load for discharging battery.
- No problems anticipated due to the electromagnetic environment at this study level.
- Control operating temperatures by providing insulation on components and/or stage. Specifically, the auto destruct temperature of initiators.