RRS-021

Reusable Reentry Satellite (RRS) Summary Report

Launch Tradeoff Study

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FOREWORD

The Reusable Reentry Satellite (RRS) Launch Tradeoff Study described herein was performed during Part 1 of the RRS Phase B contract. This report is one of several that describes the results of various trade studies performed to arrive at a recommended design for the RRS satellite system. The overall RRS Phase B Study objective is to design a relatively inexpensive satellite to access space for extended periods of time, with eventual recovery of experiments on Earth. The RRS will be capable of: 1) being launched by a variety of expendable launch vehicles, 2) operating in low earth orbit as a free flying unmanned laboratory, and 3) executing an independent atmospheric reentry and soft landing. The RRS will be designed to be refurbished and reused up to three times a year for a period of 10 years. The expected principal use for such a system is research on the effects of variable gravity (0-1.5 g) and radiation on small animals, plants, lower life forms, tissue samples, and materials processes.

This Summary Report provides a description of the RRS Launch Tradeoff Study performed to identify available launch vehicles applicable for the RRS mission. This report discusses the various launch vehicle options, launch sites, shroud limitations, interfaces, launch environment, injection accuracy, availability, integration, and costs relevant to the overall RRS design.

The study was performed under the contract technical direction of Mr. Bob Curtis, SAIC Program Manager. The Launch Vehicle Tradeoff Study was performed by Eagle Engineering, via subcontract from SAIC, under the direction of Mr. William Davidson. Mr. Michael Richardson, JSC New Initiatives Office, provided the RRS objectives and policy guidance for the performance of these tasks under the NAS 9-18202 contract.

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LIST OF ABBREVIATIONS AND ACRONYMS

ACS	Attitude Control System
AI	Artificial Intelligence
ALS	Advanced Launch System
AmRoc	American Rocket Company
ARC	Ames Research Center
CAD	Computer Aided Design
CDR	Critical Design Review
CG	Center of Gravity
COTR	Contracting Officer's Technical Representative
COTS	Commercial Off-the-Shelf
CR	Change Request
DARPA	Defense Advanced Research Projects Agency
DDT&E	Design, Development, Test and Evaluation
DMS	Data Management System
DoD	Department of Defense
DoT	Department of Transportation
DRM	Design Reference Mission
ECLSS	Environmental Control and Life Support System
ELV	Expendable Launch Vehicle
EM	Experiment Module
EPAC	E Prime Aerospace Corporation
EPS	Electrical Power System
ETR	Eastern Test Range
GD	General Dynamics
GEO	Geosynchronous Earth Orbit
GSFC	Goddard Space Flight Center
GTO	Geosynchronous Transfer Orbit
ILV	Industrial Launch Vehicle
Incl	Including
IOC	Initial Operation Capability
JSC	Johnson Space Center
KSC	Kennedy Space Center
LO_2	Liquid Oxygen

LIST OF ABBREVIATIONS AND ACRONYMS (Cont.)

MDSSC	McDonnell Douglas Space Systems Company
MLV	Medium Launch Vehicle
MMC	Martin Marietta Corporation
MST	Mobile Service Tower
NASA	National Aeronautics and Space Administration
NSTS	National Space Transportation System
OSC	Orbital Sciences Corporation
OTS	Off-the-Shelf
PacAm	Pacific American Rocket Company
PAF	Payload Attach Fitting
PASARF	Payload Adapter Sleeve and Removable Fairing
PI	Principal Investigator
POCC	Payload Operations Control Center
RF	Radio Frequency
RMOAD	Reference Mission Operational Analysis Document
RM	Rodent Module
ROM	Rough Order Magnitude
ROSAT	Roentgen Satellite
RRS	Reusable Reentry Satellite
SLC	Space Launch Complex
SSLV	Standard Small Launch Vehicle
Std	Standard
Stg	Stage
TBD	To be determined
TDRSS	Tracking and Data Relay Satellite System
USAF	United States Air Force
VEB	Vehicle Equipment Bay
WSMR	White Sands Missile Range
WTR	Western Test Range

EXECUTIVE SUMMARY

A goal of the Phase B study is to define the launch system interfaces for the RRS program. The focus of the launch tradeoff study, documented in this report, is to determine which expendable launch vehicles (ELV's) are best suited for the RRS application by understanding the impact of all viable launch systems on RRS design and operation.

Initial study included an ELV technical merit scoring and design maturity analysis on 19 viable options. It was concluded that none of the 19 launch vehicles studied were optimum for RRS. Most ELV's demonstrated insufficient performance, excessive performance and cost, or a lack of design maturity. The Delta 6920 and the S-II were the candidates most worthy of further evaluation. The main discriminator was cost. It was also determined that since shared launches present severe operational complexities, dedicated launches are recommended.

A more detailed investigation of RRS/ELV interfaces was conducted for both the Delta and S-II vehicles. The analysis focused on Delta due to a lack of design detail for S-II. The payload attachment system, which features a thin, composite, cylindrical support structure, was configured to allow for late life specimen installation [i.e. Experiment Module (EM)] without scarring the heat shield. An alternative concept, which included a removable fairing and interstage-like adapter sleeve, presented performance penalties and payload access and cost advantages.

It was also concluded from this interface analysis that the Delta should employ the 10' fairing being modified to contain a large access panel for the Roentgen Satellite (ROSAT) program to meet the late EM installation requirement. Modifications to the pre-launch timeline are required for Delta to meet the RRS close-out requirement.

RRS design should proceed in Part II of the contract with a launch vehicle interface design compatible with Delta. The S-II lacks sufficient design maturity, eliminating it from further analysis. An effort should be made to address the interface issues presented in the Spacecraft Questionnaire used by McDonnell Douglas Space Systems Company (MDSSC). Significant issues with Delta will include pre-launch timeline adjustments and payload attachment/separation system design.

Recent developments indicate that the DoD may cut spending by reducing tactical missile inventories. Several companies could propose new ELV derivatives using these surplus components to offer low cost vehicles. NASA should consider a launch services contractor competition to reduce RRS program costs.

1.0 INTRODUCTION

1.1 Background

As currently conceived, the Reusable Reentry Satellite (RRS) will be designed to provide investigators, in several biological disciplines with a relatively inexpensive method of access to space for up to 60 days with eventual recovery on Earth. The RRS will be designed to permit totally intact, relatively soft recovery of the vehicle, system refurbishment, and reflight with new and varied payloads. The RRS system will be capable of 3 reflights per year over a 10-year program lifetime. The RRS vehicle will have a large and readily accessible volume near the vehicle center of gravity for the Payload Module (PM) containing the experiment hardware. The vehicle is configured to permit the experimenter late access to the PM prior to launch and rapid access following recovery.

The RRS will operate as a free-flying spacecraft in orbit and allowed to drift in attitude to provide an acceleration environment of less than 10-5 g's. The acceleration environment during orbital trim maneuvers will be less than 10-3 g's. The RRS is also configured to spin at controlled rates to provide an artificial gravity of up to 1.5 Earth g. The RRS system will be designed to be rugged, easily maintainable, and economically refurbishable for the next flight. Some systems may be designed to be replaced rather than refurbished if cost effective and capable of meeting the specified turnaround time. The minimum time between recovery and reflight will be approximately 60 days. The PM's will be designed to be relatively autonomous with experiments which require few commands and limited telemetry. Mass storage if needed will be accommodated in the PM. The start of the hardware development and implementation phase is expected in 1991 with a first launch in December 1994.

Numerous trade studies and RRS functional design descriptions are required to define a RRS concept which satisfies the requirements and is viable. NASA has contracted with Science Applications International Corporation (SAIC) to perform a Phase B study to provide the RRS concept definition. Eagle Engineering, Inc. is supporting SAIC in accomplishing the necessary studies. The Launch Tradeoff Study is one of the supporting study analyses performed by Eagle.

1.2 NASA JSC Statement of Work Task Definition

Conduct required study with depth of analysis as appropriate to clarify and document the viability of each approach. Give particular attention to effects of complexity, flexibility, or imposed constraints on the RRS design, RM design, or mission operations. Also, special consideration should be given to system reliability and operational safety as well as the reduction in program life cycle costs.

"Consider the launch vehicle options including: (1) Delta, (2) Titan II, (3) NASDA H-II, and (4) appropriate commercial vehicles (of U.S. or Foreign origin) which are likely to be available by the mid to late 1990s. Consideration shall be given to the likely launch sites, shroud limitations, interfaces, and launch environment and accuracy. It is desirable that the RRS be capable of being launched by any of the ELV's with no or minimum modification. Consideration shall also be given to the expected accuracy of orbital insertion, launch vehicle integration cost, and expected availability. For launch vehicles which are likely to be used during shared launches, consideration shall be given to the effect of late on-pad access to the payload on the RRS configuration and design. All launch vehicle data on performance, interface, environment, and available estimates shall be submitted for approval by the NASA COTR before use."

1.3 Scope

This NASA Phase B study is intended to provide the RRS concept definition. The study includes tradeoff studies with the depth of analysis as appropriate to clarify and document the viability of each approach. The RRS system and operations are developed to the degree necessary to provide a complete description of the conceptual designs and functional specifications. Detailed engineering designs are not produced during Phase B studies since the significant resources are allocated and reserved for the subsequent Phase C/D design and implementation activities. Therefore, many analyses and definitions in this study are based on engineering experience and judgement rather than detailed design calculations.

2.0 Study Approach

2.1 Organization

The study is organized to be accomplished in a series of related but separate tradeoff studies and system concept definitions. Therefore, the documentation has been formatted to accommodate a compendium of analyses which are published in one document for launch tradeoff. The document is produced in a series of report iterations in the form of interim reports which culminate in the publishing of the final report at the midterm of the RRS Phase B Study.

2.2 Document Format

Although the individual analyses and studies are not amenable to documentation in exactly the same topical arrangement, a general outline is used where reasonable. The guideline outline for preparing the individual study sections is provided below:

- Purpose
- Assumptions and Groundrules
- Tradeoff Options
- Analysis
- Conclusions
- Recommendations

2.3 Assumptions and Groundrules

In the process of performing the subject trade study, certain data or study definition was not available or specified. Assumptions and groundrules have been established to document, for the purposes of this trade study, the definition of important information which is not a definite fact or is not available in the study time period. Specific assumptions are listed in the section where appropriate. General assumptions and groundrules which affect all studies are listed in Table 2-1.

Table 2-1. General Launch Tradeoff Study Assumptions and Groundrules

Assumptions and Groundrules
1) Where project, hardware, and operations definition has been insufficient, detailed quantitative analysis has been supplemented with assessments based on experienced judgement of analysts with space flight experience from the Mercury Project through the current time.
2) The RRS missions to be supported are those baselined in the mission operations design definition study and referred to as RRS design reference missions (DRMs). The RRS design reference missions are identified in Table 2-2.

Table 2-2. Design Reference Mission Set Definition

Definition Parameter	Design Reference Mission Set				
	DRM-1	DRM-2	DRM-3	DRM-4	DRM-5
Character	Land Recovery	High Altitude	High Inclination	Integer Orbits	Water Recovery
Inclination	33.83°	33.83°	98°	35.65°	28.5°
Orbit Type	Circular	Circular	Circular, Near-Integer	Circular, Integer	Circular
Orbit Altitude	350 km (189 nm)	900 km (486 nm)	897 km (484 nm)	479 km (259 nm)	350 km (189 nm)
Launch Site	Eastern Test Range (ETR)	Eastern Test Range (ETR)	Western Test Range (WTR)	Eastern Test Range (ETR)	Eastern Test Range (ETR)
Recovery Site	White Sands Missile Range (WSMR)	White Sands Missile Range (WSMR)	White Sands Missile Range (WSMR)	White Sands Missile Range (WSMR)	Water (ETR, Gulf of Mexico, WTR)

3.0 PURPOSE

The purpose of this tradeoff study was to assess the expendable launch vehicle fleet to determine viable RRS launch options. Early in the study, bounds on RRS size and weight were established based on performance and fairings of ELV fleet candidates. The study involved understanding the impact of candidate ELV options on RRS design and operation. Based on this analysis, specific ELV options were recommended.

4.0 LAUNCH VEHICLE OPTIONS

A relatively large set of potential ELV candidates (listed below) was composed with respect to launching the initial RRS configuration (3,000 lbs). This list includes all viable ELV options. Viable options are defined as launch vehicles likely to be operational by the mid to late 1990s with a performance of at least 3,000 lbs to DRM-1 orbit (34° and 200 nm). This list excludes Soviet and Chinese launchers due to the political implications. Also, some medium and all heavy lift launchers (i.e., Titan IV, Ariane 5, ALS) were excluded as these would probably be more expensive than other vehicles. Consequently, the list is composed of all performance-adequate versions of Atlas, Delta, Titan, Ariane, Japan's H-series, and current commercial ELV programs. The Ariane 4, Atlas II, and Atlas JII can be modified for a range of payload performances (e.g., addition of strap-on boosters) but each can be discussed as one candidate.

Note, the candidate ELV's are separated into three categories indicating the present status of each vehicle. Some vehicles have limited private funding (i.e., proposed). The probability of these proposed vehicles being available and flight proven in the near future is not as high as for ELV's which are under development or operational. Note, shared launch possibilities were also evaluated.

<u>Operational</u>

- Ariane 4
- Atlas I
- Delta 6920
- H-I
- Titan II
- Titan III

<u>Under Development</u>

- Atlas II
- Atlas JII
- Delta 7920
- H-II
- OSC Taurus (SSLV)

<u>Proposed</u>

- AmRoc ILV-110
- SSI Conestoga 421-48
- EPAC S-II
- PacAm Liberty X
- Titan II/Biprop Kit

5.0 EVALUATION METHODOLOGY

The launch vehicles were evaluated from both a technical and maturity viewpoint. ELV maturity is a programmatic parameter to establish a confidence level in the proposed ELV availability and performance. Prior to any judgments regarding ELV maturity, the technical merit of each ELV was numerically evaluated. Each ELV was assessed with respect to RRS design and operation compatibility via a list of appropriate evaluation factors (listed and described below). Each candidate was evaluated relative to each individual factor using a scoring system described later and based on the manufacturer's description of his vehicle as it is currently designed. Modifications to improve compatibility with the RRS were not considered during this initial analysis. Each evaluation factor was weighted as each related to the current RRS design and operational objectives (e.g., RRS size, late access criteria). Another matrix was composed to illustrate the final comparison of the candidates. Based on these technical evaluations a set of candidate ELV's could be recommended for further investigation. Note, these evaluations were done repeatedly as the RRS design changed (i.e., size and weight).

It was later determined that some ELV characteristics were essentially constant among the candidates and could not serve in the selection of the ELV fleet. For instance, umbilical interfaces are available on all ELV's which provide sufficient air conditioning and electrical power on the pad. Also, all standard separation systems employed pyrotechnic devices. Other factors were found to be dependent on RRS design parameters yet to be determined. For example, vibrational characteristics depend on RRS structural dynamic responses. Note, insertion accuracy became a non-issue (see below) and was not scored.

5.1 Technical Merit Evaluation Factors

Availability Date	If the ELV is "operational" or "under development" and has full financial support, the date of Initial Operation Capability (IOC) is indicated. These vehicles have a high probability of being available. The "proposed developments" (or "conceptual designs") have a low probability of being available.
Performance/Inclination	A study assumption was established that 75% of the RRS missions would have no orbital objectives other than microgravity and are referred to as DRM-1 missions (200 nm, 34° incl.). NASA has specified that some missions may require high altitudes for radiation experiments, close to 500 nm (DRM-2). Other missions may also impose high inclinations of up to 98° (DRM-3). Consequently, performance bounds were developed for each ELV using the altitude range of 200 nm to 500 nm and an inclination range of 34° to 98°.
Insertion Accuracy	All ELV's evaluated had insertion accuracies which are compatible with the RRS objectives; therefore this factor was not a discriminator. ELV's can launch into a high enough altitude to ensure the 60 day mission duration despite insertion altitude error (i.e., 200 nm plus altitude dispersion).
Launch Site/Rate Capability	The RRS program is expected to require a maximum launch rate of 3 per year. Some ELV's cannot easily accommodate this flight rate (e.g., shared launch opportunities). It is desirable to be able to launch from the continental U.S. from KSC, VAFB, or Wallops to satisfy the DRMs.
Insertion Stg Stabilization	Due to the requirement for life science experiments aboard the RRS which require live animals (i.e., rodents) the ELV selected cannot use spin stabilization during orbital insertion.

Payload Accommodations	The RRS is to be flown in a "heat shield down" launch configuration to maintain a constant axial acceleration load direction during the entire mission. A payload fairing is required to protect the RRS during ascent. The most important constraint imposed by the fairing is on the size of the RRS (i.e., diameter and length). For some ELV's a large enough fairing cannot be developed. Payload attach fitting (PAF) constraints on RRS c.g. were not assessed in this part of the contract.
Payload Accessibility	RRS close-out is to occur up to T-4 hours before launch. Close-out is assumed to involve human access (e.g., final inspections, detachment of umbilicals, rodent cage replacement).
Flight Environment	Thermal loads during launch vary moderately among ELV's. Some candidate ELV's may require fairing insulation to reduce internal temperatures below RRS tolerance. Acceleration and shock loads were also determined. It is required that the ELV not impose greater than 10 g axial acceleration loads on the rodents.
Cost per Flight	Life cycle cost was a key issue regarding selection of the candidate ELV fleet.

5.2 Maturity Evaluation Factors

ELV maturity was considered a top-level programmatic concern which should be addressed after technical merit was established. Launch vehicle maturity factors were evaluated and included the following:

- Company Relevant Experience and Past Performance
- ELV Design Approach
- Development Status
- Recurring Cost Risk

These factors were introduced to establish a confidence level in the proposed ELV availability and performance. It is desirable to choose a flight proven, operational launch system as opposed to a proposed design involving potential development, cost, and schedule risk.

6.0 LAUNCH VEHICLE DATA

Data for each of the candidate ELV's was obtained with respect to the technical evaluation factors described earlier. Data sheets have been included in Tables 6-1 through 6-19 for each candidate launch system in alphabetical order. Extra Atlas configurations (i.e., Atlas IIA, Atlas IIAS, and Atlas IIIS) have also been included.

Table 6-1. Candidate Launch Vehicle Data Sheet, Ariane 4 (AR40)

Vehicle: Ariane 4 (AR40)
Sponsor: Arianespace/CNES

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	1987	2.5 yr ARO
Performance:		
200 nm due East @ 5°:	10,500 lbs	
500 nm due East @ 5°:	7,300 lbs	
200 nm @ 98°:	7,800 lbs	
500 nm @ 98°:	5,700 lbs	
Insertion Accuracy:		for 435 nm and 98.6°incl.
Altitude:	±0.38 nm	
Inclination:	±0.032°	
Launch Site Availability:	Kourou, French Guiana	ELA 3; ELA 2 is backup
Inclination Capability:	5.2°-102°	
Launch Rate Capability:	8-10/year	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	85%	23/27 launches
Cost per Flight:	$80M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	144" (12.0' int., 13.1' ext.)	15' Contraves fairing avail.
	Usable Fairing Length:	31.5' or 37.4'
Adapters and Interface Rings:	103.3" VEB, 75.6" std	3 other adapters available
Single/Multi Payload Capab.:	Dual Payload Capability	SPELDA (encloses lower payload)
Payload Accessibility:		
Fairing Timeline Installation:	TBD	
Flight Environment:		
Internal Fairing Skin Temp.:	59°F-77°F (prelnch); 77°F-94°F (in-flt)	
Accelerations:	4.5 g (long.), ±2 g (lat.)	
Shock:	1100 g (1050 Hz - 10000 Hz)	

Table 6-2. Candidate Launch Vehicle Data Sheet, Atlas I

Vehicle: Atlas I
Sponsor: General Dynamics/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	1990	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	11,600 lbs	
500 nm due East @ 28.5°:	5,100 lbs	
200 nm @ 98°:	n/a	
500 nm @ 98°:	n/a	
Insertion Accuracy:		
Altitude:	±3.5 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63.5°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	88% (demonstrated since 1970)	
Cost per Flight:	$65M-$70M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab.:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp.:	60°F-69°F (prelnch); 80°F-398°F (in-flt)	
Accelerations:	5.5 g (long.), ±2 g (lat.)	
Shock:	2000 g @ 2000 Hz	

Table 6-3. Candidate Launch Vehicle Data Sheet, Atlas II

Vehicle: Atlas II
Sponsor: General Dynamics/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	Early 1991	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	13,500 lbs	
500 nm due East @ 28.5°:	7,500 lbs	
200 nm @ 98°:	11,300 lbs	
500 nm @ 98°:	6,300 lbs	
Insertion Accuracy:		
Altitude:	±3.5 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	88% (demonstrated since 1970)	
Cost per Flight:	\$70 - \$80 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp:	69°F (prelnch); 80°F-398°F (in-flt)	
Accelerations:	5.5 g (long.); ±2.0 g (lat.)	
Shock:	8000 g @ 2000 Hz	

Table 6-4. Candidate Launch Vehicle Data Sheet, Atlas IIA

Vehicle: Atlas IIA
Sponsor: General Dynamics/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	Early 1991	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	13,900 lbs	
500 nm due East @ 28.5°:	7,800 lbs	
200 nm @ 98°:	11,700 lbs	
500 nm @ 98°:	5,450 lbs	
Insertion Accuracy:		
Altitude:	±3.5 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	88% (demonstrated since 1970)	
Cost per Flight:	$80 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp:	69°F (prelnch); 80°F-398°F (in-flt)	
Accelerations:	5.5 g (long.); ±2.0 g (lat.)	
Shock:	8000 g @ 2000 Hz	

Table 6-5. Candidate Launch Vehicle Data Sheet, Atlas IIAS

Vehicle: Atlas IIAS
Sponsor: General Dynamics

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	1992	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	14,750 lbs	
500 nm due East @ 28.5°:	7,000 lbs	
200 nm @ 98°:	unavail.	
500 nm @ 98°:	unavail.	
Insertion Accuracy:		
Altitude:	±3.5 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	88% (demonstrated since 1970)	
Cost per Flight:	$85 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp:	69°F (prelnch); 80°F-398°F (in-flt)	
Accelerations:	5.5 g (long.); ±2.0 g (lat.)	
Shock:	8000 g @ 2000 Hz	

Table 6-6. Candidate Launch Vehicle Data Sheet, Atlas JII

Vehicle: Atlas JII
Sponsor: General Dynamics/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	1989	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	8,700 lbs	
600 nm due East @ 28.5°:	6,200 lbs	
200 nm @ 98°:	n/a	
500 nm @ 98°:	n/a	
Insertion Accuracy:		
Altitude:	±3.5 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	Spin-stabilized	
Reliability:	90% (demonstrated by Atlas w/o Centaur)	
Cost per Flight:	$52 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp:	69°F (prelnch); 80°F-398°F (in-flt)	
Accelerations:	5.5 g (long.); ±2.0 g (lat.)	
Shock:	2000 g @ 2000 Hz	

Table 6-7. Candidate Launch Vehicle Data Sheet, Atlas JIIS

Vehicle: Atlas JIIS
Sponsor: General Dynamics

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date: Program	1992	Avail. via USAF MLV
Performance:		
200 nm due East @ 28.5°:	8,000 lbs	
700 nm due East @ 28.5°:	4,600 lbs	
200 nm @ 98°:	n/a	
500 nm @ 98°:	n/a	
Insertion Accuracy:		
Altitude:	±15 nm	
Inclination:	±0.011°	
Launch Site Availability:	ETR/WTR	36B/SLC-3
Inclination Capability:	28.5°-34°/63°-98°	
Launch Rate Capability:	1/yr commercial	4/yr USAF
Insertion Stage Stabilization:	Spin-stabilized	
Reliability:	90% (Atlas w/o Centaur)	
Cost per Flight:	$55 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	115" (9.6' int., 11' ext.); 147" (12.3' int., 14' ext.)	
Usable Fairing Length:	17.3'; 29.4'	
Adapters and Interface Rings:	Standard type A and B S/C adapters	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-8 to 13 days	
Flight Environment:		
Internal Fairing Skin Temp:	69°F (prelnch); 80°F-398°F (launch)	Insulation required
Accelerations:	5.5 g (long.); ±2.0 g (lat.)	
Shock:	2000 g @ 2000 Hz	

Table 6-8. Candidate Launch Vehicle Data Sheet, Conestoga 421-48

Vehicle: Conestoga 421-48
Sponsor: Space Services Inc.

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	Proposed development	
Performance:		
200 nm due East @ 37°:	2,800 lbs	
500 nm due East @ 37°:	1,900 lbs	
200 nm @ 98°:	2,030 lbs	
500 nm @ 98°:	1,350 lbs	
Insertion Accuracy:		
Altitude:	±17.6 nm	
Inclination:	±0.1°	
Launch Site Availability:	Wallops/WTR	
Inclination Capability:	38°-??/63°-98°	
Launch Rate Capability:	12/year	
Insertion Stage Stabilization:	3-axis stable with TVC	
Reliability:	95%	goal
Cost per Flight:	$15 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	57" (4.75')	76" could be developed
Usable Fairing Length:	11.3' or 12.3'	
Adapters and Interface Rings:	Std Delta PAF 3712C	
Single/Multi Payload Capab:	Single or Multiple (3 cannisters)	
Payload Accessibility:		
Fairing Timeline Installation:	Up to T-5 hours	
Flight Environment:		
Internal Fairing Skin Temp:	20°C-25°C (prelnch); 195°C (in-flt)	
Accelerations:	8.7 g (long.); ±0.4 g (lat.)	
Shock:	±1000 g @ 1000 Hz	

Table 6-9. Candidate Launch Vehicle Data Sheet, Delta 6920

Vehicle: Delta 6920
Sponsor: McDonnell Douglas/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	February 1989	
Performance:		
200 nm due East @ 28.7°:	8,400 lbs	
500 nm due East @ 28.7°:	7,200 lbs	
200 nm @ 98°:	5,600 lbs	
500 nm @ 98°:	5,000 lbs	
Insertion Accuracy:		
Altitude:	±10 nm	
Inclination:	±0.05°	
Launch Site Availability:	ETR (2)/WTR	17A, 17B/SLC-2W
Inclination Capability:	28.5°-42.5°/94°-145.3° (retro 34.7°)	
Launch Rate Capability:	10-12/yr	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	92% (demonstrated since 1970)	
Cost per Flight:	$43 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	100" (8.3' int., 9.5' ext.); 110" (9.2' int., 10' ext.)	
Usable Fairing Length:	21.7', 19.8' (143" cyl.)	
Adapters and Interface Rings:	6019 (5.0');6306 (5.25');6915 (5.7')	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-3 days	
Flight Environment:		
Internal Fairing Skin Temp:	62°F (prelnch); 62°F-110°F (launch) w/insulation	
Accelerations:	5.8 g (long.); ±2.5 g (lat.)	
Shock:	5500 g @ 4000 to 5000 Hz	

Note: MDSSC requires roll-back 5 hours, at the latest, before any Delta launch.

Table 6-10. Candidate Launch Vehicle Data Sheet, Delta 7920

Vehicle: Delta 7920
Sponsor: McDonnell Douglas/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	June 1990	Navstar launches
Performance:		
200 nm due East @ 28.7°:	10,500 lbs	
500 nm due East @ 28.7°:	9,200 lbs	
200 nm @ 98°:	7,300 lbs	
500 nm @ 98°:	6,700 lbs	
Insertion Accuracy:		
Altitude:	±10 nm	
Inclination:	±0.05°	
Launch Site Availability:	ETR (2)/WTR	17A, 17B/SLC-2W
Inclination Capability:	28.5°-42.5°/94°-145.3° (retro 34.7°)	
Launch Rate Capability:	10-12/yr	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	92% (demonstrated since 1970)	
Cost per Flight:	$45 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	100" (8.3' int., 9.5' ext.); 110" (9.2' int., 10' ext.)	
Usable Fairing Length:	21.7'; 19.8' (143" cyl.)	
Adapters and Interface Rings:	6019 (5.0');6306 (5.25');6915 (5.7')	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-3 days	
Flight Environment:		
Internal Fairing Skin Temp:	62°F (prelnch); 62°F-110°F (launch) w/insulation	
Accelerations:	10 g (long.); ±2.5 g (lat.)	
Shock:	5500 g @ 4000 Hz to 5000 Hz	

Note: MDSSC requires roll-back 5 hours, at the latest, before any Delta launch.

Table 6-11. Candidate Launch Vehicle Data Sheet, H-I

Vehicle: H-I
Sponsor: Mitsubishi/NASDA

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	1986 - 1991	
Performance:		Two-stage
200 nm due East @ 30°:	5,500 lbs	
500 nm due East @ 30°:	5,100 lbs	
200 nm @ 98°:	3,250 lbs	
500 nm @ 98°:	2,900 lbs	
Insertion Accuracy:	3-sigma	Restartable LE-5 engine
Altitude:	TBD	
Inclination:	TBD	
Launch Site Availability:	Tanegashima Island, Japan	Osaki Pad
Inclination Capability:	30°-96°+	
Launch Rate Capability: launches	2-4/year	4 months/yr avail. for
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	89% (demonstrated)	"N" & "H" vehicles
Cost per Flight:	$70 M	NASA estimate
Payload Accommodations: ·		
Usable Fairing Diameter:	86" (7.2' int., 8' ext.)	
Usable Fairing Length:	20 ' (22'ext.)	
Adapters and Interface Rings:	60.2"	
Single/Multi Payload Capab.:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-1 to 4 days	
Flight Environment:		
Internal Fairing Skin Temp.:	25°C (prelnch); 66°C (in-flt)	w/acoustic blankets
Accelerations:	8 g (long.)	
Shock:	2000 g @ 600 Hz to 4000 Hz	

Table 6-12. Candidate Launch Vehicle Data Sheet, H-II

Vehicle: H-II
Sponsor: Mitsubishi/NASDA

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	1992	
Performance:		
200 nm due East @ 30°:	22,000 lbs	
500 nm due East @ 30°:	19,900 lbs	
200 nm @ 98°:	14,100 lbs	
500 nm @ 98°:	11,200 lbs	
Insertion Accuracy:	3-sigma	Restartable LE-5 engine
Altitude:	TBD	
Inclination:	TBD	
Launch Site Availability:	Tanegashima Island, Japan	Yoshinobuzaki Pad
Inclination Capability:	30°-100°+	
Launch Rate Capability:	2-4/year	4 months/yr avail. for launches
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	91% (predicted)	
Cost per Flight:	$110 M	NASA estimate
Payload Accommodations:		
Usable Fairing Diameter: available	145" (12.1' int., 13.4' ext.)	15' Contraves fairing
	Usable Fairing Length:	39.4'
Adapters and Interface Rings:	TBD	
Single/Multi Payload Capab.:	Dual payload capability	
Payload Accessibility:		
Fairing Timeline Installation:	TBD	
Flight Environment:		
Internal Fairing Skin Temp.:	15°C-25°C (prelnch)	
Accelerations:	4 g (long.)	
Shock:	2000 g @ 750 Hz to 5000 Hz	

Table 6-13. Candidate Launch Vehicle Data Sheet, ILV-110

Vehicle: ILV-110
Sponsor: American Rocket Company (AmRoc)

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	Proposed development for 1991	Static-test fires complete
Performance:		
200 nm due East @ 28.5°:	3,600 lbs	
500 nm due East @ 28.5°:	2,800 lbs	
200 nm @ 98°:	2,500 lbs	
500 nm @ 98°:	1,850 lbs	
Insertion Accuracy:		
Altitude:	TBD	
Inclination:	TBD	
Launch Site Availability:	WTR	Proposed
Inclination Capability:	63°-98°	Estimated from Titan II
Launch Rate Capability:	3/yr	Predicted
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	TBD	
Cost per Flight:	$10 M	ROM estimate
Payload Accommodations:		
Usable Fairing Diameter:	90" (7.5')	Similar to Delta fairing
Usable Fairing Length:	15' (10' cyl.)	Includes 6' conical section
Adapters and Interface Rings:	TBD	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	TBD	
Flight Environment:		
Internal Fairing Skin Temp:	TBD	
Accelerations:	5.2 g (long.)	
Shock:	TBD	

Table 6-14. Candidate Launch Vehicle Data Sheet, Liberty X

Vehicle: Liberty X
Sponsor: Pacific American

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	Conceptual Design	
Performance:		
150 nm due East @ 28.5°:	2000 lbs	Predicted
500 nm due East @ 28.5°:	TBD	
200 nm @ 98°:	TBD	
500 nm @ 98°:	TBD	
Insertion Accuracy:		
Altitude:	TBD	
Inclination:	TBD	
Launch Site Availability:	Hawaii/ETR/WTR	$5M development/pad
Inclination Capability:	TBD	
Launch Rate Capability:	TBD	
Insertion Stage Stabilization:	TVC	
Reliability:	TBD	
Cost per Flight:	$5 M	ROM estimate
Payload Accommodations:		
Usable Fairing Diameter:	TBD	
Usable Fairing Length:	TBD	
Adapters and Interface Rings:	24" Scout-type and 37" Delta-type	
Single/Multi Payload Capab.:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	Day of launch	
Flight Environment:		
Internal Fairing Skin Temp.:	TBD	
Accelerations:	10 g	
Shock:	TBD	

Table 6-15. Candidate Launch Vehicle Data Sheet, S-II

Date: 8-15-89

Vehicle: S-II
Sponsor: E Prime Aerospace (EPAC)

Factor	Evaluation	Remarks
Availability Date:	Proposed Development	
Performance:		
200 nm due East @ 28.5°:	7,800 lbs	
500 nm due East @ 28.5°:	6,700 lbs	
200 nm @ 98°:	5,200 lbs	
500 nm @ 98°:	4,400 lbs	
Insertion Accuracy:		
Altitude:	±9 nm	
Inclination:	±0.05°	
Launch Site Availability:	ETR/WTR	Preliminary discussions w/KSC
Inclination Capability:	28.5°-57°/63°-90°+	
Launch Rate Capability:	1/wk	Predicted
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	TBD	Derived from Peacekeeper
Cost per Flight:	$30 M	ROM estimate
Payload Accommodations:		
Usable Fairing Diameter:	88" (7.4'); 100" (8.3')	100" hammerhead being studied
Usable Fairing Length:	10.3' + TBD extension ; 13.6' (cyl.)	
Adapters and Interface Rings:	92"	
Single/Multi Payload Capab.:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	TBD	
Flight Environment:		
Internal Fairing Skin Temp.:	70°F-75°F	Cork insulation
Accelerations:	6.5 g (long.); ±2.5 g (lat.)	
Shock:	9,000 g @ 1050 Hz to 5000 Hz	

Table 6-16. Candidate Launch Vehicle Data Sheet, Taurus (SSLV)

Vehicle: Taurus (SSLV)
Sponsor: Orbital Sciences/DARPA

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	March 1991	18 month ARO
Performance:		
225 nm due East @ 28.5°:	3,350 lbs	
500 nm due East @ 28.5°:	2,750 lbs	
225 nm @ 98°:	2,600 lbs	
500 nm @ 98°:	2,100 lbs	
Insertion Accuracy:		
Altitude:	±25 nm	
Inclination:	±0.2°	
Launch Site Availability:	ETR/WTR	Mobile launch system
Inclination Capability:	28.5°-42.5°/63°-98°	
Launch Rate Capability:	TBD	Greater than 3/yr
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	95%	Goal
Cost per Flight:	$15 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	50" (4.2'); 75" (6.3')	75" hammerhead being studied
Usable Fairing Length:	8'; 15' (10' cyl.)	
Adapters and Interface Rings:	TBD	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	Day of launch	
Flight Environment:		
Internal Fairing Skin Temp:	TBD	
Accelerations:	< 7 g (long.)	Less than 10 g
Shock:	TBD	

Table 6-17. Candidate Launch Vehicle Data Sheet, Titan II

Vehicle: Titan II
Sponsor: Martin Marietta/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	September 1988	Not commercially avail.
Performance:		
200 nm due East @ 63.5°:	700 lbs	
500 nm due East @ 63.5°:	n/a	
200 nm @ 99°:	500 lbs	
500 nm @ 99°:	n/a	
Insertion Accuracy:		
Altitude:	±12-25 ft/sec	
Inclination:	±0.15°	
Launch Site Availability:	WTR	
Inclination Capability:	63.5° - 100°	
Launch Rate Capability:	3/yr	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	98.8% (Gemini and 34B)	
Cost per Flight:	$30-35 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	112" (9.36' int., 10' ext.)	
Usable Fairing Length:	20', 25', 30'	
Adapters and Interface Rings:	36" and 56"	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-12 to 14 days	
Flight Environment:		
Internal Fairing Skin Temp:	40°F-110°F (prelnch); 75°F-200°F (launch)	
Accelerations:	10 g (long.); ±2.5 g (lat.)	
Shock:	200 g @ 400 Hz	

Table 6-18. Candidate Launch Vehicle Data Sheet, Titan II/Biprop

Vehicle: Titan II/Biprop Extended Mission Kit — Date: 8-15-89

Sponsor: Martin Marietta

Factor	Evaluation	Remarks
Availability Date:	Proposed Development	
Performance:		
200 nm due East @ 63.5°:	4,400 lbs	
500 nm due East @ 63.5°:	3,200 lbs	
200 nm @ 99°:	3,300 lbs	
500 nm @ 99°:	2,200 lbs	
Insertion Accuracy:		
Altitude:	±12-25 ft/sec	
Inclination:	±0.15°	
Launch Site Availability:	WTR	
Inclination Capability:	63.5° - 100°	
Launch Rate Capability:	3/yr	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	98.8% (Gemini and 34B)	
Cost per Flight:	$45 M (w/biprops kit)	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	112" (9.36' int., 10' ext.)	
Usable Fairing Length:	20', 25', 30'	
Adapters and Interface Rings:	36" and 56"	
Single/Multi Payload Capab:	Single	
Payload Accessibility:		
Fairing Timeline Installation:	T-12 to 14 days	
Flight Environment:		
Internal Fairing Skin Temp:	40°F-110°F (prelnch); 75°F-200°F (launch)	
Accelerations:	10 g (long.); ±2.5 g (lat.)	
Shock:	200 g @ 400 Hz	

Martin has found significant structural problems in using the Delta SSPS as an Upper Stage on Titan II.

Table 6-19. Candidate Launch Vehicle Data Sheet, Titan III

Vehicle: Titan III
Sponsor: Martin Marietta/USAF

Date: 8-15-89

Factor	Evaluation	Remarks
Availability Date:	July 1989	
Performance:		
200 nm due East @ 28.5°:	22,000 lbs	
500 nm due East @ 28.5°:		
200 nm @ 98°:	n/a	
500 nm @ 98°:	n/a	
Insertion Accuracy:	3-sigma	
Altitude:		
Inclination:		
Launch Site Availability:	ETR	Pad 40
Inclination Capability:	28.6°-35.4°	
Launch Rate Capability:	2-4/yr	
Insertion Stage Stabilization:	3-axis ACS	
Reliability:	94% (demonstrated since 1970)	
Cost per Flight:	$130 - $140 M	Fixed price estimate
Payload Accommodations:		
Usable Fairing Diameter:	112" (9.36' int., 10' ext.)	Larger fairings avail.
Usable Fairing Length:	15.0' + 5' increments	
Adapters and Interface Rings:	120"	
Single/Multi Payload Capab.:	Dual launch capability	
Payload Accessibility:		
Fairing Timeline Installation:	TBD	
Flight Environment:		
Internal Fairing Skin Temp:	50°F-100°F(prelaunch);500°F(in-flt)	125°F w/insulation
Accelerations:	7.0 g (long.); ±1.0 g (lat.)	
Shock:	1,000 g @ 2000 Hz	

Note: Titan III requires roll-back 2 hours before launch.7.0 ELV Candidate Analysis

7.0 ELV CANDIDATE ANALYSIS

7.1 Dedicated versus Shared Launches

Analysis regarding shared-launch potential was included in the technical merit evaluation. Several medium-lift launchers can provide shared launch opportunities (i.e., Delta, Atlas, Ariane, Titan III, H-II). Delta and Atlas have not developed a shared launch capability; however, they could if required. The Japanese are currently developing a dual launch system for H-II. Ariane and the commercial Titan III already employ operational dual launch systems.

The majority of the Delta launches support GEO payloads and are launched due East from ETR into a 28.5° inclination. On these flights normally no excess payload capability is available. On Delta low Earth orbit missions there could be opportunities for shared missions; however, none could be assured. Titan III has sufficient performance to launch two relatively heavy payloads to LEO (e.g., two GEO satellites). Titan III will typically insert into a 90 or 100 nm circular orbit. Titan III cannot reach the 200 nm circular orbit required by RRS due its continuous burn trajectory (no coast). The propulsive requirements for a maneuver to raise perigee from 100 nm to 200 nm is large enough (ΔV = 350 ft/sec) to significantly change the baseline RRS design (150 lbs. of propellant estimated), which allows for only minor orbit corrections. Development of restart capability on the Titan 2nd stage is an alternative to this additional RRS propulsive requirement. General Dynamics (Atlas) estimates that possibly two ETR opportunities per year could arise by shifting primary payloads to Atlas versions with more performance to allow for an RRS piggyback (e.g., from Atlas II to Atlas IIA).

A shared launch on Ariane 4 may be difficult to schedule in 1992 or 1993 due to the large backlog of payloads. The launch site latitude of Kourou (5°) would make shared launches with GEO costumers extremely difficult. The Ariane vehicle must first deliver the RRS to 200 nm and 34° inclination. It would not be possible for the Ariane 4 to make the substantial inclination change back to 5° for delivery of a GEO payload. The software and propulsion systems onboard the GEO satellite would have to be modified significantly to accommodate this scenario. One piggyback opportunity per year may exist on the H-II once it becomes operational. However, the relatively high cost per lb (i.e., $5,000/lb) would drive the cost of a shared launch close to that of a dedicated launch on a smaller ELV. Note, a shared launch opportunity could be offered to the RRS program by NASDA for political reasons at no expense.

Shared launches are operationally complex. Launch scheduling can be extremely delicate. The RRS must be ready to fly when the primary payload is scheduled to launch. Even if the RRS is prepared to fly the primary payload may create a delay or cancellation causing RRS to be remanifested. Pad accessibility and launch timelines are also restricted when another payload is involved since crews for both payloads may want late access. It may also be difficult to install the Experiment Module (EM) via access panels due to the EM size and cumbersome weight without modifications to the fairing (access issues discussed in more detail later). As mentioned earlier, the RRS will impose modifications on the ELV trajectory and the primary payload's orbital operations due to RRS unique orbit requirements (i.e., 34° inclination). These maneuvers will significantly affect GEO payload software and propulsion requirements from any launch site below 34° latitude (e.g., KSC). The potential cost savings for a shared launch on Atlas would not be desirable due to these operational impacts. Even if a shared launch is offered for political reasons (i.e., on H-II or Ariane 4) it appears that such a mission could not be accomplished without major inconveniences. Dedicated ELV's appear to be more desirable for RRS launches. ELV shared launches could be assessed in depth during the second half of the Phase B contract.

7.2 Technical Evaluation Factor Scoring System

A discussion follows pertaining to the technical merit scoring for each evaluation factor. Figure 7-3 has been inserted after the discussion to summarize the ELV candidate comparison results based on current baseline RRS design (3,800 lbs. including attach hardware, 90" diameter). This chart includes shaded circles to represent points (see legend). Note, these points are adjusted by appropriate weighing factors in a final scoring matrix (discussed later). For purposes of final evaluation the H-II and Titan III were assumed to allow one shared launch opportunity per year at half the dedicated launch cost.

7.2.1 Availability Date

The likelihood of a specific ELV being operational for the RRS program (early to mid 1990s) was assessed. Any vehicle funded by the government is assumed to have a high probability of becoming operational (e.g., OSC's Taurus is being funded by DARPA) and was given a full point. It is difficult to project that a specific proposed vehicle will be fully-funded and be available for RRS. Vehicles which require technology developments were considered less likely to become operational than proposed vehicles which incorporate existing, flight proven hardware. For instance, AmRoc's ILV is to use the first large hybrid motor. The first launch was a failure due to a LO_2 valve malfunction. It is difficult to determine whether the vehicle will be fully operational in the near term. SSI's Conestoga and EPAC's S-II are based on existing, flight proven components; however, are not currently fully-funded for development. The same is true for the Titan II with a new bi-propellant upper stage. All of these vehicles were given half points.

Materials obtained from Pacific American demonstrated that the Liberty X design was a technology development not mature enough to seriously consider (no points). Note, the H-I will be out of production in 1991 and could not be considered available (no points).

7.2.2 Performance/Inclination Capability

It is appropriate to consider only those ELV configurations which have the performance to satisfy DRM-1 (200 nm, 34°). Note, the RRS weight has grown from the proposal configuration of 3,000 lbs to the current baseline of 3,400 lbs. Including minimum attach hardware this weight is expected to increase to approximately 3,800 lbs. This weight growth eliminates the Taurus as currently designed for DRM-1. Note, Taurus could be modified to accommodate the weight growth. Performance-inadequate vehicles were included in the final score matrix (Figure 7-4) but were considered "not applicable" for RRS launches. A summary chart of performance capabilities for all candidates is provided in Figures 7-1.

Any ELV which cannot accomplish all orbit requirements (DRM-3) but can handle the minimum orbit requirement (DRM-1) should be given 3/4 of a point since their performance would satisfy a predicted 75% of RRS missions. If an ELV can accomplish the entire range of required orbits a full point is given. Note, it will be necessary to have at least one ELV capable of all reference missions.

7.2.3 Launch Site/Launch Rate Capability

The capability of ELV candidates to handle the mission model was assessed. The RRS program may require up to three launches per year. The mission model requires high inclination launch capability. The ability to launch from both ETR and WTR allows for the entire range of inclinations. If an ELV has access to the entire range of inclinations from its pad(s) and can accommodate three launches per year a full point was given. If an ELV cannot launch into high inclinations (i.e., 98°) only half a point was allocated. If three launch opportunities per year are not available no points were given.

7.2.4 Insertion Stage Stabilization

Any ELV configuration that uses spin stabilization during insertion provides an unacceptable environment for RRS life science payloads. However, ELV manufacturers could

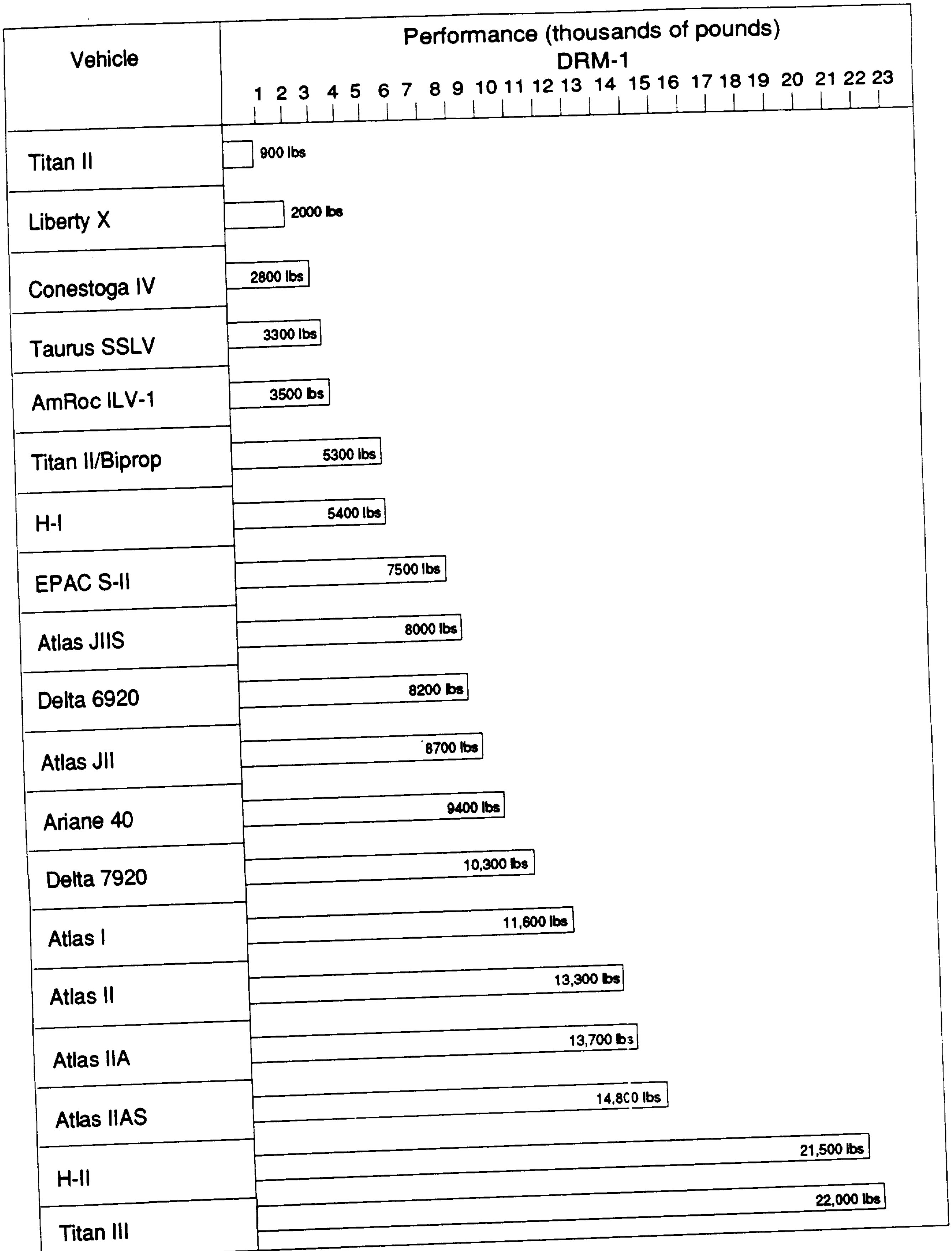


Figure 7-1. Launch Vehicle Evaluation – Performance Assessment

replace the spin-stabilized upper stage with a 3-axis ACS at some development cost. All ELV's which employ spin-stabilization were not given a point.

7.2.5 Payload Accommodations

Provided the candidate ELV can supply a standard fairing to enclose the RRS dimensions, the ELV candidate will not seriously constrain the RRS shape and a full point is given. If a new fairing must and can be developed to enclose the RRS only half a point was given. ELV's which cannot feasibly accommodate the size of the RRS despite a fairing development received no points. Here again, the dimensions of the RRS changed during the tradeoff study. The diameter increased from about 80" to the current baseline of 90".

7.2.6 Payload Accessibility

The late installation requirement is for EM installment up to T-12 hours before launch. The late access requirement is for hands-on (human) interaction with the payload up to T-4 hours before launch (e.g., visual inspection). The EM must be installed vertically. Provided the candidate ELV can satisfy both late installation and close-out requirements, the ELV is given a full point.

Typical size access panels cannot be used to install the baseline EM of 34" diameters, 30" long, and approximately 300 lbs. Fairing modifications, discussed in section 8.3.1, appear necessary. It is assumed that standard fairing access panels provide adequate means to perform close-out activities. However, some ELV's do not allow for late human access to the payload due to presence of hazardous propellants, gantry removal, etc. It may be possible to alter pre-launch timelines to accommodate the close-out requirement for most ELV's (also discussed in section 8.3.1).

Although solid and pre-packaged storable propellant ELV's have less restrictions on late human access (i.e., no propellant loading), these installation and access issues are still not easily resolved. None of the ELV's evaluated complied with the requirements in their entirety and were given half points; therefore, this is a major issue to be worked in the second half of the contract.

7.2.7 Flight Environment

All the ELV candidates can offer reasonable thermal environments. Some candidates will require fairing insulation while others will not. The addition of insulation is not a significant issue. The ELV candidates impose axial acceleration loads varying from 4 g to 10 g. It was assumed that any ELV imposing 10 g loads is approaching the threshold of rodent tolerance. Also, the structural implications may be significant since re-entry g loads are not expected to reach 10 g. Consequently, the ELV's which impose 10 g axial acceleration loads during ascent were only given half points.

7.2.8 Cost per Flight

A major driver in RRS design is life cycle cost. The cost per flight indicated for each candidate launch vehicle is given in Figure 7-2. Since even small variations in cost are important, launch costs were normalized by dividing each cost into the lowest launch cost of $30 M. Note, Liberty X, Conestoga IV, and ILV-110 launch costs were not considered applicable due to performance inadequacy. For instance, the Delta 6920 is scored as 0.70 points (30/43) and the S-II as 1.00 points (30/30). The launch costs were obtained from NASA, the DoT, and ELV representatives, and include basic launch services (i.e., propellants, mission analysis, etc.) and hardware costs. These normalized ratios have been placed in the summary chart as points in Figure 7-3. Due to the proposed status of the S-II there is risk associated with assessing the cost.

7.3 Technical Evaluation Factor Weighting

The following relative weights were used to assess the technical merit of the candidate ELV's. However, the resulting scores were not intended to produce clear winners. This scoring exercise was used to determine the optimum or most desirable candidate ELV's for the RRS program.

Technical Evaluation Factors	Weights
Availability Date	10
Performance/Inclination Capability	20
Launch Site/Rate Capability	5
Insertion Stage Stabilization	10
Payload Accommodations	10
Payload Accessibility	15
Flight Environment	5
Cost per Flight	25
TOTAL	100

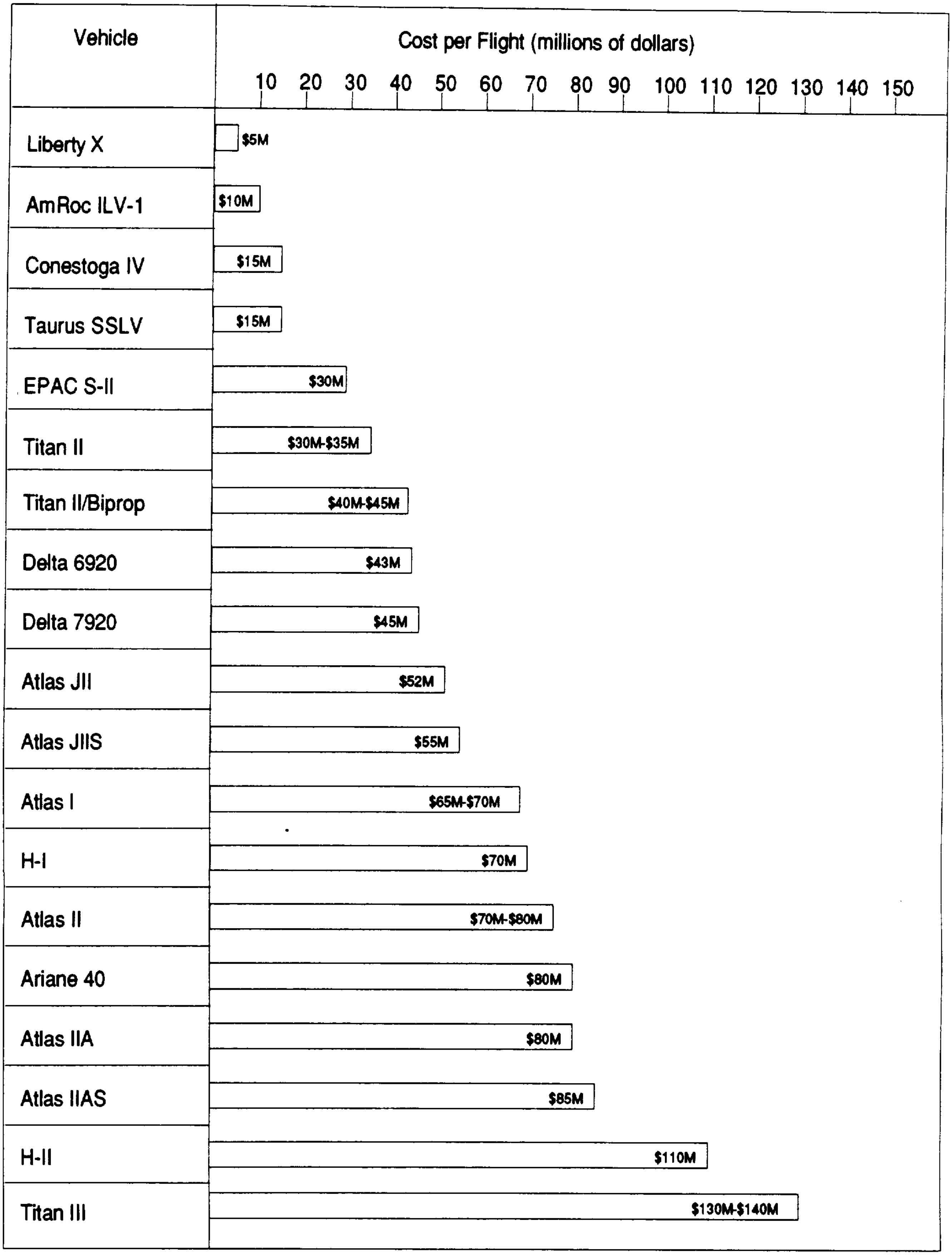


Figure 7-2. Launch Vehicle Evaluation – Cost Assessment

LAUNCH VEHICLE EVALUATION

LAUNCH VEHICLE CATEGORIES	Availability Date	Performance/Inclination Capability	Launch Site/Rate Capability	Insertion Stage Stabilization	Payload Accommodations	Payload Accessibility	Flight Environment	Cost per Flight [3]
● OPERATIONAL								
ARIANE 4	●	●	●	●	●	◒	●	0.38
ATLAS I	●	●	●	●	●	◒	●	0.46
DELTA 6920	●	●	●	●	●	◒	●	0.70
NASDA H-1 [1]	○	◕	◒	●	◒	◒	●	0.43
TITAN II	●	○	◒	●	●	◒	◒	N/A
TITAN III [2]	●	●	○	●	●	◒	●	0.23
● UNDER DEVELOPMENT								
ATLAS II	●	●	●	●	●	◒	●	0.43
ATLAS JII	●	●	●	○	●	◒	●	0.58
DELTA 7920	●	●	●	●	●	◒	●	0.67
NASDA H-II [2]	●	●	○	●	●	◒	●	0.27
TAURUS	●	○	●	●	○	◒	●	N/A
● PROPOSED								
AMROC ILV	◒	○	◒	●	●	◒	●	N/A
CONESTOGA IV	◒	○	◒	●	○	◒	◒	N/A
EPAC S-II	◒	●	◒	●	◒	◒	●	1.00 [5]
LIBERTY X	○	○	○	●	○	◒	◒	N/A
TITAN II [4]	◒	◕	◒	●	●	◒	◒	0.67

1. Out of production in 1991
2. Dual launch
3. Normalized ratio of $10M/cost per launch
4. Includes biprop extended mission kit
5. Early estimate; may not materialize

● 1 pt. ◒ 1/2 pt. ○ 0 pt.
◕ 3/4 pt. ◔ 1/4 pt.

Figure 7-3. ELV Candidate Evaluation Summary

7.4 Technical Evaluation Scoring Results

Initial ELV scoring was completed based on the early RRS baseline configuration of 3,000 lbs (3,400 lbs with attach hardware) and approximately 76" diameter. The top scorers were Taurus, ILV-110, S-II, and Delta 6920. After further analysis Taurus became clearly the best option for DRM-1. Delta or S-II vehicles would still be required for the high inclination missions.

By dropping the RRS payload weight to 2,500 lbs., no ELV's would be added to the list of performance qualifiers from the 3,000 lb case. The smaller ELV's would still not be able to perform the entire range of orbit requirements (i.e., DRM-3).

Science requirements pushed the weight to the current baseline of 3,800 (with attach hardware), out of the performance range of Taurus and ILV-110. The diameter has also increased to 90". OSC calculations indicate that a maximum usable diameter of 75" could be achieved on Taurus via a hammerhead shroud. Another set of scores were generated (Figure 7-4). Note, the launch costs for Liberty X, Conestoga IV, ILV-110, and Taurus are based on performance-inadequate designs and could not be scored and compared to the other launchers (i.e., n/a).

Since all of the scored ELV's could perform DRM-1 the "performance/inclination capability" category did not significantly affect the relative scores. It was difficult to determine which ELV's posed less serious late accessibility issues (i.e., fairing modifications, timeline adjustments, etc.) and all ELV's were given the same score. The most significant factor, as expected, was cost per flight. Many of the ELV's can do the job provided certain modifications are made and minor additional costs are accepted; however, the basic cost of the dedicated launch vehicle service makes Delta 6920 and S-II (the two most inexpensive vehicles with sufficient performance) the best options from a purely technical viewpoint.

Top Technical Scores	Cost Per Flight	Performance (DRM-1)
Delta 6920 (85 pts.)	$43 M (Fixed price estimate)	8,150 lbs
S-II (80 pts)	$30 M (ROM estimate)	7,500 lbs

LAUNCH VEHICLE EVALUATION

LAUNCH VEHICLE CATEGORIES	Availability Date	Performance/Inclination Capability	Launch Site/Rate Capability	Insertion Stage Stabilization	Payload Accommodations	Payload Accessibility	Flight Environment	Cost per Flight	Total (%)
● OPERATIONAL									
ARIANE 4	10	20	5	10	10	7.5	5	9.50	77
ATLAS I	10	20	5	10	10	7.5	5	11.50	79
DELTA 6920	10	20	5	10	10	7.5	5	17.50	84
NASDA H-1	0	15	2.5	10	5	7.5	5	10.75	56
TITAN II	10	0	2.5	10	10	7.5	2.5	N/A	N/A
TITAN III	10	20	0	10	10	7.5	5	5.75	68
● UNDER DEVELOPMENT									
ATLAS II	10	20	5	10	10	7.5	5	10.75	79
ATLAS JII	10	20	2.5	0	10	7.5	5	14.50	N/A
DELTA 7920	10	20	5	10	10	7.5	2.5	16.75	84
NASDA H-II	10	20	0	10	10	7.5	5	6.75	69
TAURUS	10	0	5	10	0	7.5	5	N/A	N/A
● PROPOSED									
AMROC ILV	5	0	2.5	10	10	7.5	5	N/A	N/A
CONESTOGA IV	5	0	2.5	10	0	7.5	2.5	N/A	N/A
EPAC S-II	5	20	2.5	10	5	7.5	5	25.00	80
LIBERTY X	0	0	0	10	0	7.5	2.5	N/A	N/A
TITAN II/Biprop	5	15	2.5	10	10	7.5	2.5	16.75	69

Figure 7-4. ELV Candidate Score Matrix

7.5 Maturity Considerations

The S-II is a conceptual design based on existing, flight proven peacekeeper components and the Star-92 motors under study by Morton Thiokol. Currently, EPAC is not actively developing the S-II due to a lack of funding. EPAC has no contractual agreements to launch the S-II. Although EPAC was involved in a sub-orbital launch, EPAC has not conducted an any orbital launches to date. Consequently, due to the lack of flight experience of EPAC and current development status of the S-II, it would be premature to choose the S-II as a leading candidate.

In contrast, the Delta 6920 is based on a mature launch system. Delta versions have demonstrated the highest success rate (98%) of any medium launcher over the past 20 years and have flown about 140 missions since 1967. The only major modifications involved with the Delta II series over its predecessor is the stretching of the first stage (i.e., propellant addition), first stage motor nozzle extension, and replacement of the Castor IV solid booster strap-ons with the Castor IVA. The Delta 6920 does not include the Star-48 upper stage. There is little reason to question the reliability of the Delta II 6920 launch vehicle. MDSSC has had nearly 30 years experience, including over 180 flights, with the earlier Delta vehicles.

7.6 Evaluation Results

Utilizing the technical evaluation factor weighting the Delta 6920 and S-II vehicles are the best options among the 19 ELV options investigated. The S-II cannot be considered a serious contender to the Delta 6920 due to a lack of maturity. Altering the weight and size of the RRS slightly (10%) will not result in a different set of best ELV candidate options. ELV/RRS interface analysis was conducted for both vehicles and has been summarized in Section 8.0.

8.0 ELV/RRS INTERFACE ANALYSIS

8.1 Purpose

The purpose of this analysis is to define the interfaces between the RRS and the candidate ELV's (i.e., Delta 6920 and S-II) and determine the constraints imposed by these interfaces on RRS design and operation. A more thorough interface analysis will be conducted during the second half of the Phase B contract and summarized in another report.

8.2 Interface Definition

The process of defining the spacecraft/ELV interfaces typically requires that the user agency and ELV representatives follow an extensive documentation process. The Delta program has a well-defined documentation process. Typically MDSSC conducts a broad feasibility study to determine if general needs (e.g., performance, payload volume, launch window, etc.) can be accommodated. The user's first responsibility to MDSSC is the Spacecraft Questionnaire (see Figure 8-1). This document is not kept current, but evolves into the Payload/Launch System Interface Specification. The focus of the questionnaire is on specific spacecraft requirements which might interfere with ELV operation. For instance, Section 2.4 requires a description of the spacecraft mass properties and a dynamic model. This data is used to determine how the spacecraft will respond to various ascent and separation loads.

Several aspects of the RRS/ELV interface were investigated and are discussed in this section including structural (attach fittings, fairing), electrical, thermal, data, separation, and flight environment interfaces. Potential interface issues are discussed.

8.3 Interface Options and Analysis

8.3.1 Payload Attachment System

The Payload Attach Fitting (PAF) is typically the structural interface between the RRS and the ELV. These mechanisms are typically supplied by the ELV manufacturer and do not count as user payload. These devices normally employ pyrotechnics to separate the payload. PAF's constrain the payload within the allowable dynamic fairing envelope tolerances during ascent. Standard PAF configurations have been studied and are summarized in Table 8-1. MDSSC offers three different PAF's for the Delta 6920, each capable of handling a CG at least 80" from the attachment plane. EPAC (S-II) plans to employ the 3712 PAF (see Figure 8-2) currently used on 3-stage Delta vehicles.

The RRS will be flown "heat shield down" to maintain the axial acceleration loads in the same direction throughout the mission, to keep the rodents oriented in the same direction. A groundrule has been established that no scarring of the heat shield is permitted; therefore, structural attachment of the RRS to the ELV will require an extra support structure between the PAF and the flat side of the RRS (see Figure 8-3). The separation mechanism on the PAF will not be activated.

1 Mission Requirements and Restraints
1.1 Number of launches
1.2 Frequency of launches
1.3 Desired transfer orbit
  1.3.1 Apogee (integrated)
  1.3.2 Perigee (integrated)
  1.3.3 Inclination
  1.3.4 Argument of perigee
  1.3.5 Other
1.4 Launch window restraints and duration
  1.4.1 Sun angle
  1.4.2 Eclipse
  1.4.3 Ascending node
  1.4.4 Inclination
  1.4.5 Window durations (over a year's span)
  1.4.6 Other
1.5 Separation requirements (including tolerances)
  1.5.1 Position
  1.5.2 Attitude
  1.5.3 Sequence and timing
  1.5.4 Tipoff and coning
  1.5.5 Spin rate at separation
  1.5.6 Other
1.6 Special trajectory requirements
  1.6.1 Thermal maneuvers
  1.6.2 T/M maneuvers
  1.6.3 Free molecular heating restraints

2 Spacecraft Characteristics
2.1 Size and space envelope (ground configuration)
  2.1.1 Dimensioned drawings
2.2 Size and space envelope (launch configuration)
  2.2.1 Dimensioned drawings
  2.2.2 Protuberances beyond allowable fairing envelope
2.3 Orbit configuration
  2.3.1 Assembly drawings and dimensioned sketches
2.4 Mass properties and dynamnic data (launch configuration/separation configuration)
  2.4.1 Weight
  2.4.2 CG and principal axis misalignment
  2.4.3 I_x
  2.4.4 I_y
  2.4.5 I_z
  2.4.6 Coordinate system (spacecraft versus launch vehicle)
  2.4.7 Fundamental frequencies (thrust axis/lateral axis)
  2.4.8 Are all significant vibration modes above 35 Hz in thrust axis and 15 Hz in lateral axes?
  2.4.9 Description of spacecraft dynamic model
    2.4.9.1 Mass matrix
    2.4.9.2 Stiffness matrix
    2.4.9.3 Response recovery matrix
  2.4.10 Combined spacecraft-third stage nutation time constant for ignition and burnout conditions
  2.4.11 Time constant and description of spacecraft energy dissipation sources and locations (i.e., hydrazine fill factor, passive nutation dampers, flexible antennae, etc.)
2.5 Antennas (ground and launch)
  2.5.1 Location
  2.5.2 Direction
  2.5.3 Frequency
  2.5.4 Who provides (ground only)?
2.6 Ordnance items (EEDs)
  2.6.1 Category A EEDs
  2.6.2 Category B EEDs
  2.6.3 Location
  2.6.4 Function
  2.6.5 Type
  2.6.6 Manufacturing part number
  2.6.7 No fire current levels
  2.6.8 All fire current levels
  2.6.9 When installed
  2.6.10 Where installed
  2.6.11 When connected
  2.6.12 Where connected
  2.6.13 Do shielding caps comply with safety requirements?
  2.6.14 Is RF susceptibility data available? List references
  2.6.15 Are electrostatic sensitivity data available on Category A EEDs? List references
2.7 Ordnance items
  2.7.1 Type and manufacturer
  2.7.2 Characteristics
  2.7.3 S&A description
  2.7.4 Ignition system description
  2.7.5 Will AKM be shipped with pyrogen installed?
  2.7.6 Has a hazard classification been assigned to the AKM (see ESMCR-127-1, Paragraph 3.13.2)
  2.7.7 Qualification and test documentation references
  2.7.8 Will a flight termination waiver be requested? If yes, provide data required in Section 4
  2.7.9 Other
2.8 Special safety items
  2.8.1 Does spacecraft contain:
    2.8.1.1 Hydrazine? Quantity, spec, etc.
    2.8.1.2 Other hazardous fluids? Quantity, spec, etc.
    2.8.1.3 High pressure gas? Quantity, spec, etc.
    2.8.1.4 Radioactive devices?
  2.8.2 Can spacecraft produce nonionizing radiation at hazardous levels
  2.8.3 Do pressure vessles conform to special safety requirements of Section 4?
2.9 Fairing environmental control
  2.9.1 In-flight requirements of spacecraft
  2.9.2 Spacecraft ground requirements (fairing installed)
  2.9.3 Critical surfaces (i.e., type, size, location)
  2.9.4 Surface sensitivity (e.g., susceptibility to propellants, gases and exhaust products and other contaminants)

3 Flight Spacecraft Requirements
3.1 Mechanical attachment description (include separation switches, if required)
  3.1.1 PAF desired
3.2 Fairing requirements
  3.2.1 Maximum internal temperature
  3.2.2 RF transparent window requirements
    3.2.2.1 Number
    3.2.2.2 Size
    3.2.2.3 Location
  3.2.3 Special access door requirements
    3.2.3.1 Number
    3.2.3.2 Size
    3.2.3.3 Location, orientation, and station referenced to booster system
3.3 Spacecraft RF radiation
  3.3.1 Power levels
  3.3.2 Frequency
  3.3.3 Open-loop checkout requirements
  3.3.4 In flight prior to separation
  3.3.5 Closed loop checkout requirements
  3.3.6 Location of spacecraft when radiating
  3.3.7 Duration of radiation
  3.3.8 To what location
3.4 Electrical bonding
  3.4.1 Does the spacecraft comply with the electrical bonding requirements of Section 3?
3.5 Interface connector and umbilicals
  3.5.1 Type and part number
  3.5.2 Location
  3.5.3 Orientation
3.6 Spacecraft to blockhouse wiring requirements
  3.6.1 Number of wires required
  3.6.2 Pin assignments in the spacecraft umbilical connector(s)
  3.6.3 Purpose and nomenclature of each wire including voltage, current, polarity requirements, and maximum resistance
  3.6.4 Shielding requirements
  3.6.5 Voltage of the spacecraft battery and polarity of the battery ground

4 Facility Requirements
4.1 Facility preference and priority. Does the spacecraft project intend to contract with the government or Astrotech directly?
4.2 List the payload processing facilities the spacecraft project desires to use in priority.
4.3 List the hazardous processing facilities the spacecraft project desires to use in priority.
4.4 What are the expected dwell times the spacecraft project would spend in the payload processing facilities?
4.5 Spacecraft environmental requirements.
4.6 Do spacecraft contamination requirements conform with capabilities of existing facilities?
4.7 What are the spacecraft and ground equipment space requirements?
4.8 What are the facility crane requirements?
4.9 What are the facility electrical requirements?
4.10 What are the security requirements?
4.11 Is a multishift operation planned?
4.12 List the support items the spacecraft project needs from NASA, USAF, or Astrotech to support the processing of spacecraft. Are there any unique support items?

5 Spacecraft Development and Test Programs
5.1 Test schedule at launch site
  5.1.1 Operations flow chart (flow chart should be a detailed sequence of operations referencing days and shifts and location)
5.2 Spacecraft development and test schedules
  5.2.1 Flow chart
  5.2.2 Is a test PAF required? When?
  5.2.3 Is clamp band ordnance required? When?
5.3 Special test requirements
  5.3.1 PAF fit check
  5.3.2 Spacecraft spin balancing
  5.3.3 Other

6 Identify any additional spacecraft or mission requirements that exceed the capabilities or violate the constraints defined in the User Manual

Figure 8-1. Questionnaire

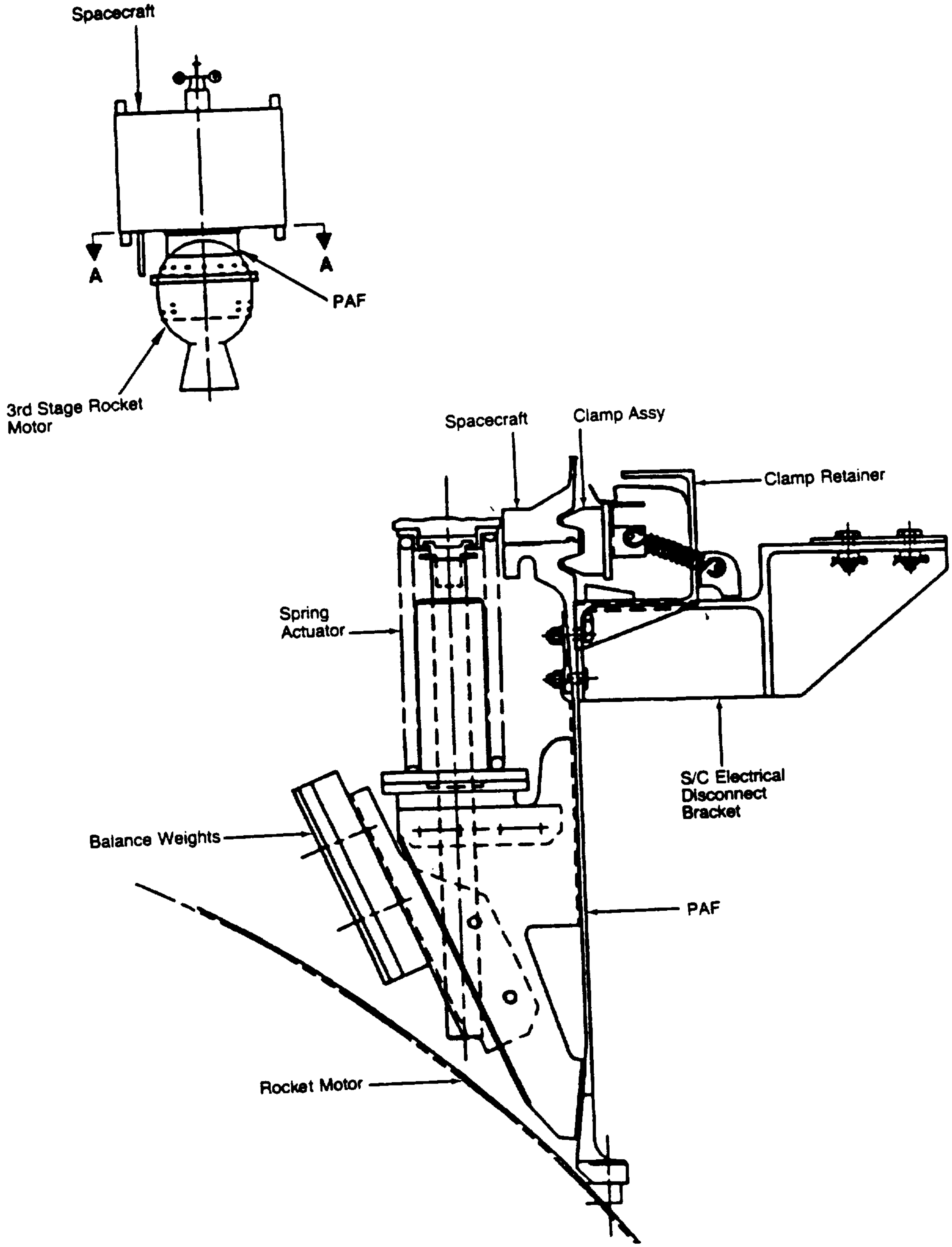


Figure 8-2. Delta PAF 3712

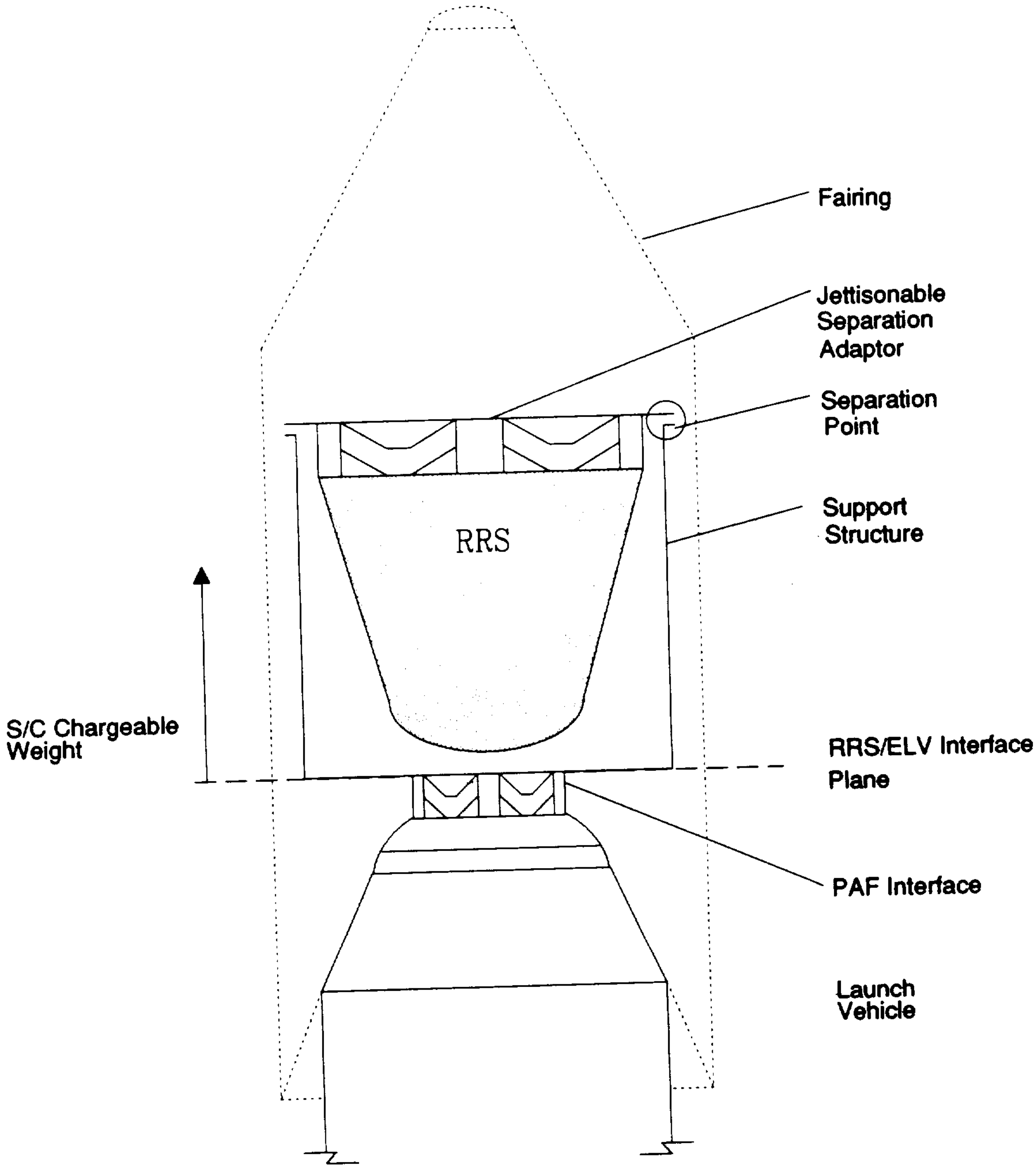


Figure 8-3. RRS Attachment System

A separation system which detaches and deploys the RRS from the support structure must be designed.

Table 8-1. Payload Attach Fitting Configurations

ELV	PAF	Weight[3]	Release Type	Separation System	Mating Dia.	CG Limit[1]	Comments
Delta	6019	125 lbs	Bolt/Latch[2]	Retro	60"	82"	3 bolts
Delta	6915	210 lbs	Bolt	Spring/Retro[2]	69"	82"	4 bolts
Delta	6306	110 lbs	Clamp/Latch[2]	Retro	63"	101"	
S-II	3712[4]	27 lbs	Clamp	Spring	37"	40"	Same as Delta 3712

[1] Distance from separation plane to payload CG (based on payload weight of 3800 lbs)
[2] Optional for reducing attitude dispersions
[3] Not chargeable to payload
[4] MDSSC designation

The major consideration involved in PAF selection is the constraint on payload CG position. The baseline RRS length of 80" would indicate that the standard Delta PAF's will be compatible. The PAF 3712, supplied by MDSSC for Delta missions using the Star-48 upper stage, and baselined for the S-II would not provide sufficient support. EPAC would have to investigate alternative PAF designs.

The extra support structure between the PAF and RRS could be a composite, cylinder-shaped, support structure analogous to the SYLDA, which is used to separate/support payloads for dual launches on Ariane 4. The weight for such a structure would be driven by the payload size and weight, and ascent loads. By properly scaling the SYLDA to support the RRS, it appears that this structure would weigh between 200 and 400 lbs, or roughly 5% to 10% of the RRS weight. This weight is chargeable to the RRS spacecraft.

It would be advantageous to be able to attach this support structure to any candidate PAF (i.e., to a wide range of PAF mating diameters). A flat, disk-shaped interface (called baseplate herein) could accommodate the range of PAF mating diameters, acting as an adapter between the PAF and support structure.

If a clamp type attachment system is employed (as on PAF 3712), an RRS spacecraft chargeable ring must be attached to the baseplate and then the ring is connected to the PAF via clamp retainers. PAF's with a bolt type attachment do not require this extra ring and can be bolted to the baseplate directly. Bolt type options are somewhat easier to use than clamp systems. Holes must be drilled via a template into the baseplate. Since these systems will not be activated (i.e., no separation) the choice between PAF's should be based on structural requirements. Consequently, RRS structural attachment to various ELV's can be accomplished via traditional PAF attachment mechanism along with a special adapter and support cylinder structure at a moderate weight penalty (10% of RRS weight estimated). Note, the separation interface is discussed later in Section 8.3.2.

8.3.1 Fairing Volume/Access

Significant characteristics for fairings offered or under study by MDSSC (Delta) and EPAC (S-II) are displayed in Table 8-2 below. The RRS must be attached in a "heat shield down" configuration. Since the maximum diameter of the RRS occurs 80" from the nose of the shield, it is necessary to have a usable cylindrical volume within the fairing of 90" diameter and 80" in length. Only the 10' standard Delta fairing currently in development will accommodate this volume. A hammerhead fairing concept being studied by EPAC could also provide sufficient volume.

Table 8-2. Candidate Fairing Characteristics

Fairing	RRS Dia. Constraint	RRS Length Constraint[4]	Sectors	Install. Time (nominal)
Delta 9.5' (std)	86"	144"	2	T-3 days
Delta 10'(std)[1]	110"	147"	3	T-3 days
S-II (standard)	88"	TBD[3]	None	TBD
S-II (hammerhead)[2]	100"	163"	2	TBD

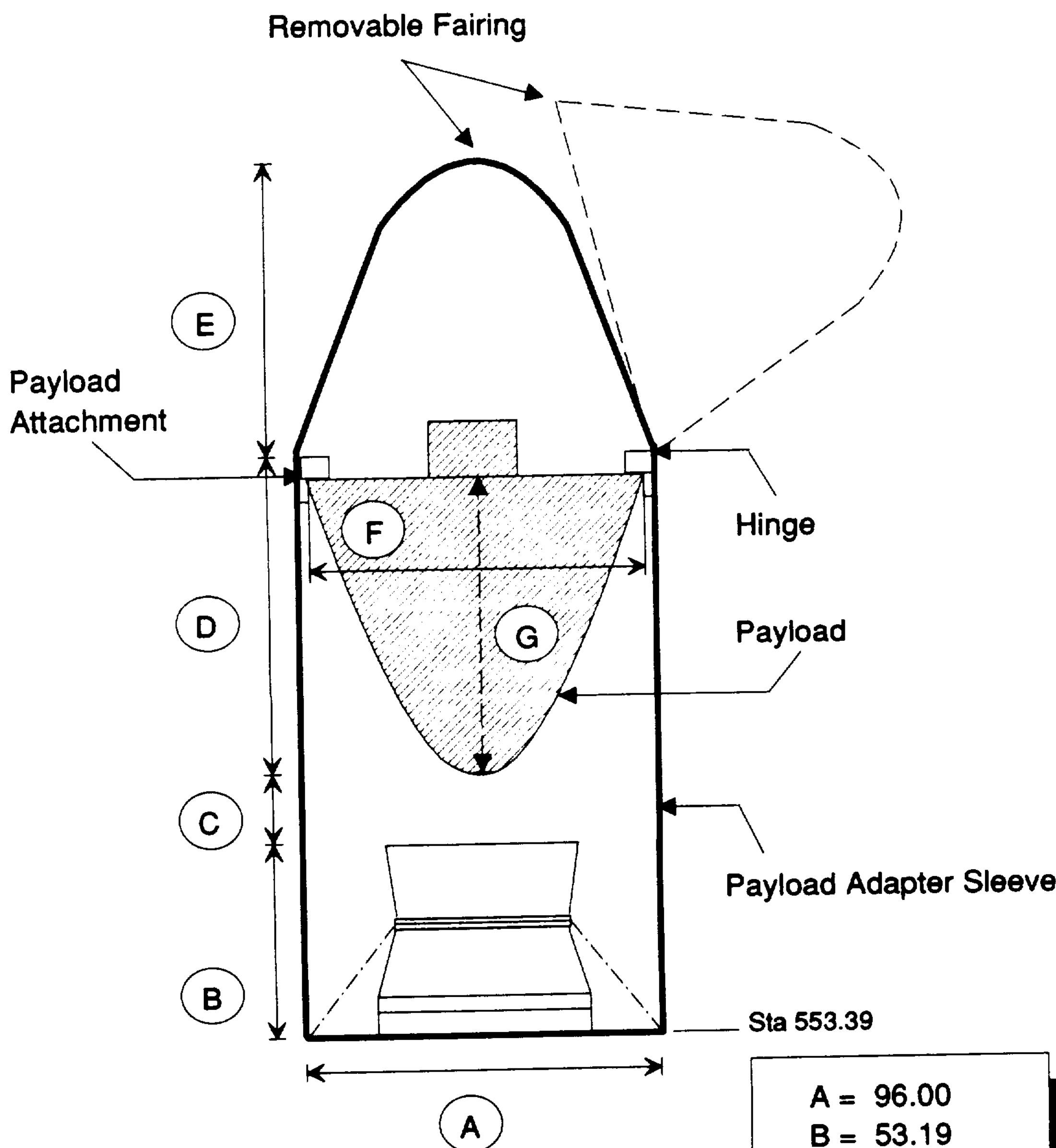
[1] Currently under final development
[2] Growth approximate (under study)
[3] Requires extension module
[4] Fairing cylindrical length

The requirement for late installation (T-12 hours) will be difficult to satisfy with these standard fairings. On Delta the problem of late installation is compounded by liquid propellant loading, which requires fairing installation at T-3 days to minimize human presence around the payload during subsequent hazardous activities (e.g., 2nd stage propellant loading at T-2 days). A generalized pre-launch activity timeline for Delta vehicles during the final week has been inserted as Appendix A.

Since the Delta requires fairing installation at T-3 days, access panels must be used to install the EM at T-12 hours before launch to avoid costly modifications to the timeline, fairing, etc. The standard access panel size for Delta is 24" by 21". Larger panels cause structural problems for the fairing which can be compensated for via extra stiffeners. The maximum panel size allowable for the Delta fairings, without significant re-design, is a 36" by 36" panel.

The EM, which contains the rodent cages, some consumables, and the rodents themselves, is currently 34" wide and 30" long, and weighs almost 300 lbs. Installation of the EM via a panel adjacent to the RRS periphery would not be possible as there is not sufficient clearance to maneuver the EM between the fairing and RRS. A portable gantry crane could be used to translate the EM over the RRS and lower it into position. Note, the EM must remain vertical to keep the life specimens oriented properly. The portable gantry crane boom (I-beam) and hook mechanism would require some clearance (12" estimated). The 36" X 36" panels already available on Delta fairings would not provide sufficient clearance for maneuvering the EM into place. Note, the recovery package must be installed after the EM and will affect the pre-launch timeline. This package is smaller and lighter (150 lbs) than the EM and should not constrain the access infrastructure.

A special fairing/interstage development, illustrated in Figure 8-4, was conceptualized to improve access conditions and provide a system more easily adapted for use on other ELV's. The concept includes a removable fairing nose and an adapter sleeve for structural support. The removable fairing could be installed after close-out, or be connected via a hinge and then rotated into place and installed after close-out. The nose piece would be pyrotechnically separated after leaving the atmosphere. The adapter sleeve would act as an interstage and support the payload. A structural adapter could be added to the base of this sleeve to allow for smooth aerodynamic transition to ELV's with different mating diameters at a economical cost and weight penalty.



A =	96.00
B =	53.19
C =	18.00
D =	87.00
E =	80.70
F =	90.00
G =	81.00

all lengths in inches

Reference:
Commercial Delta II User Manual, Raeburn, L.C., McDonnell Douglas Commercial Delta, Inc., p. 3-6, MDC H3224A, July, 1987.

Figure 8-4. Payload Adapter Sleeve and Removable Fairing (PASARF)

This concept is similar to the standard S-II fairing (derived from the MX) which is tri-conic and installed in longitudinal segments. This MX fairing would require an extension module, similar to the adapter sleeve, to enclose the RRS. This approach would require the white room ceiling to be disassembled just prior to lowering the EM into place.

The consensus from MDSSC, EPAC, and Hercules was that this special fairing unit cost would be slightly cheaper than standard fairings. The development cost would be offset by the unit cost savings over 30 flights according to MDSSC. Some cost benefit may arise, according to Hercules and EPAC.

This interstage would be approximately 160" long and 96" wide to accommodate the RRS and the separation device on the Delta; however, this sleeve would be carried to orbit and a portion of the weight would be payload chargeable. The performance penalty would be significantly more than if the fairing was jettisoned sub-orbital, which is the standard procedure (1,000 lbs penalty estimated based on Delta interstage scaling calculations). Consequently, the removable fairing/ adapter sleeve concept does appear to have some performance penalties; but the excessive capability of the Delta for RRS missions does not rule out this option.

MDSSC has recently announced plans to modify the 10' fairing for the ROSAT program to include a larger access panel under the Goddard contract. This panel could be installed any place on the cylindrical portion of the fairing without significant cost penalty, and will be 36" wide and 60" tall. Further investigation of this new development will be conducted during the second half of the contract as information becomes available (e.g., design drawings).

Installing the 36" X 60" access panel near the upper rim of the cylindrical section of the fairing, above the RRS, would accommodate the portable gantry crane clearance requirements (i.e., 12"). Since the standard Delta 10' fairing with the large access panel will exist and appears to present no serious late access issues, it was baselined for the Delta vehicle. Figure 8-5 illustrates this Delta EM installation scenario.

The late access requirement of T-4 hours requires human access to the RRS (e.g., visual inspection, rodent cage change-out). Access panels are typically employed for late access requirements. However, late access on the Delta is a significant issue. The current pre-launch timelines indicate that MST roll-back must commence at T-5 to T-7 hours prior to launch. Preparations for roll-back typically take approximately two hours to complete; therefore, the actual close-out of the RRS must occur sometime before the roll-back. It may be possible to work out a

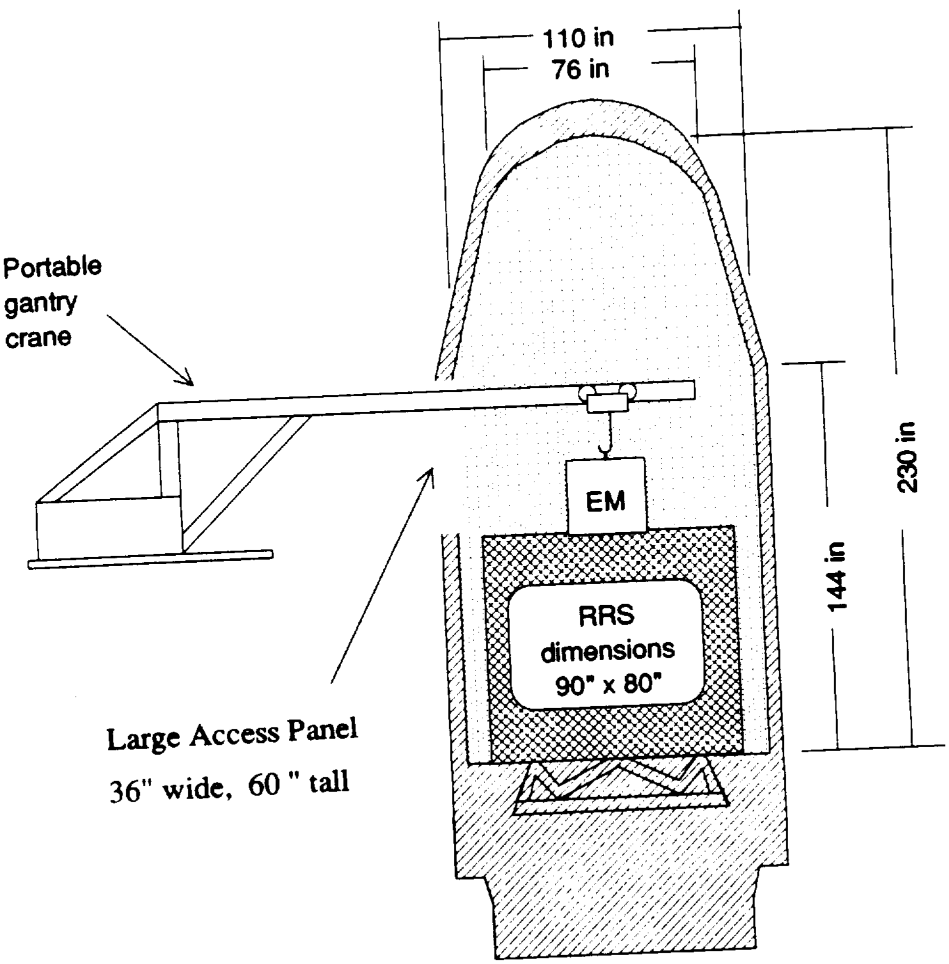


Figure 8-5. Delta Late EM Installation Scenario (ROSAT Fairing)

new pre-launch timeline (e.g., reduce built-in holds, reduce activity durations, etc.) at some cost. Preliminary analysis indicates only minor time reductions are possible. MDSSC could determine if meeting the T-4 hour constraint is feasible. This issue must be studied in depth in Part II of the contract.

If it is later determined that the large panel does not provide adequate access conditions (e.g., not compatible with pre-launch timeline adjustments for late access), new fairing designs will be explored. For instance, modifications to a standard fairing to allow for a removable nose which is pyrotechnically separated prior to separation of the fairing sectors may improve access without significant cost and performance penalties.

The S-II is configured for horizontal integration (including payload) with a portable clean room on the ground. The S-II uses no gantry; rather, this vehicle is launched from a canister much like the MX missile from which it was derived. This canister is to be rotated into vertical position, followed by solid motor arming, platform stabilization, etc. and launched in two hours or less; however, as mentioned earlier, horizontal integration of the EM is unacceptable. Consequently, a portable white room (on work platform) and vertical installation procedure will probably be required if S-II was utilized.

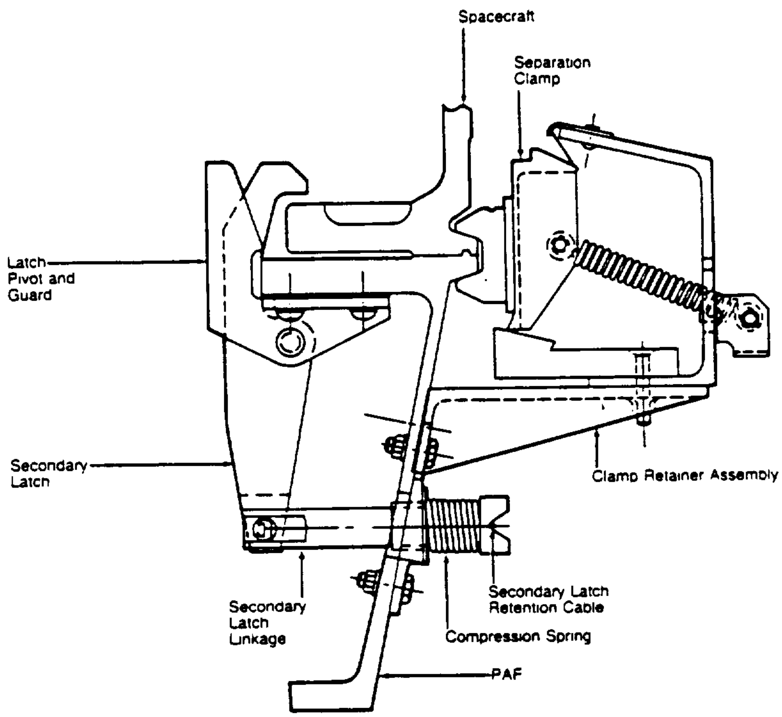
A solid and pre-packaged storable propellant-using a vehicle like the S-II does not require hazardous propellant loading and can accommodate late fairing installation somewhat easier. However, due to the typically employed "crocodile" separation technique it takes a considerable amount of time to install one fairing half and perform all post-installation checks. It takes approximately 12 hours to prepare for, install, and check the Delta 9.5' fairing. Note, the Delta 10' fairing separates into three sectors. The S-II hammerhead shroud would require similar installation efforts. It is questionable whether one half could be installed, followed by final solid motor arming, terminal countdown, etc., without exceeding the 12 hour late installation requirement. Note, an autonomous crane could lower the EM into place since no gantry crane normally exists with the current S-II baseline. The proposed S-II hammerhead design could include a large enough access panel (e.g., similar to the ROSAT fairing) to satisfy the installation requirement; however, it may be more difficult to use an access door on the S-II without a gantry crane. A removable nose fairing design may be a better solution to the late installation requirement for S-II since the EM could be lowered into position via an autonomous crane.

8.3.2 Separation

As explained earlier, no attachments (i.e., scars) to the heat shield are acceptable. The cylindrical support structure (see Figure 8-3) must attach to the top side of the RRS. The separation plane must then be above the heat shield in the launch configuration. Typical separation mechanisms, discussed herein, will require attach points which protrude beyond the periphery of the RRS; however, no protrusions over the periphery of the RRS are allowable due to the need for a smooth aerodynamic shape. Consequently, these attach points must be retracted (i.e., four individual attach points) or jettisoned (i.e., ring structure). An alternative would be to have the support structure separate much like a fairing followed by a 2nd stage retro fire to produce the relative separation velocity; however, such an option would be too complex and expensive to develop.

MDSSC offers four standard PAF's with the Delta which use two basic release mechanisms including the clamp retainer and the exploding nut mechanisms. Secondary latch systems are offered to reduce attitude dispersions upon release. The RRS separation system could employ these mechanisms, but not with the PAF. For missions where the separation tip-off angular rate must be less than 0.2° per second, a two-step separation system is recommended by MDSSC. After bolt or clamp release and a sufficient time for angular-rate dissipation (e.g., 15 sec), secondary latches are released and the 2nd stage retro fires to provide the relative separation velocity. However, this recommendation is based on separation from PAF's. Springs can also be used to provide separation velocity. It is desirable that the RRS separate from cylinder support structure without contacting the inner wall. The payload tip-off rate must be relatively small compared to that required during a typical payload release from a PAF due to the presence of the cylinder. Attitude dynamics modelling will be necessary to determine which separation systems should be employed. Tentatively, it would appear that a secondary latch system could be desirable to reduce attitude dispersions. If it is determined that the inner wall will be contacted additional design modifications should be investigated (e.g., guides, teflon inner wall, etc.).

Since the Delta-type separation systems must remain outside the maximum RRS diameter, attached to protruding attach points, separation clearance becomes an issue. This tolerance would be at least 4" on each side if the Delta clamp release ring (see Figure 8-6) is attached just above and on the periphery of the RRS due to the clamp retainers. The Delta exploding nut release mechanism (see Figure 8-7) also will require significant additional clearances since the bolt catcher must remain outside the RRS diameter (e.g., 2" on each side). The presence of secondary latches (as devised by MDSSC) will not increase the clearance requirements if used as a secondary



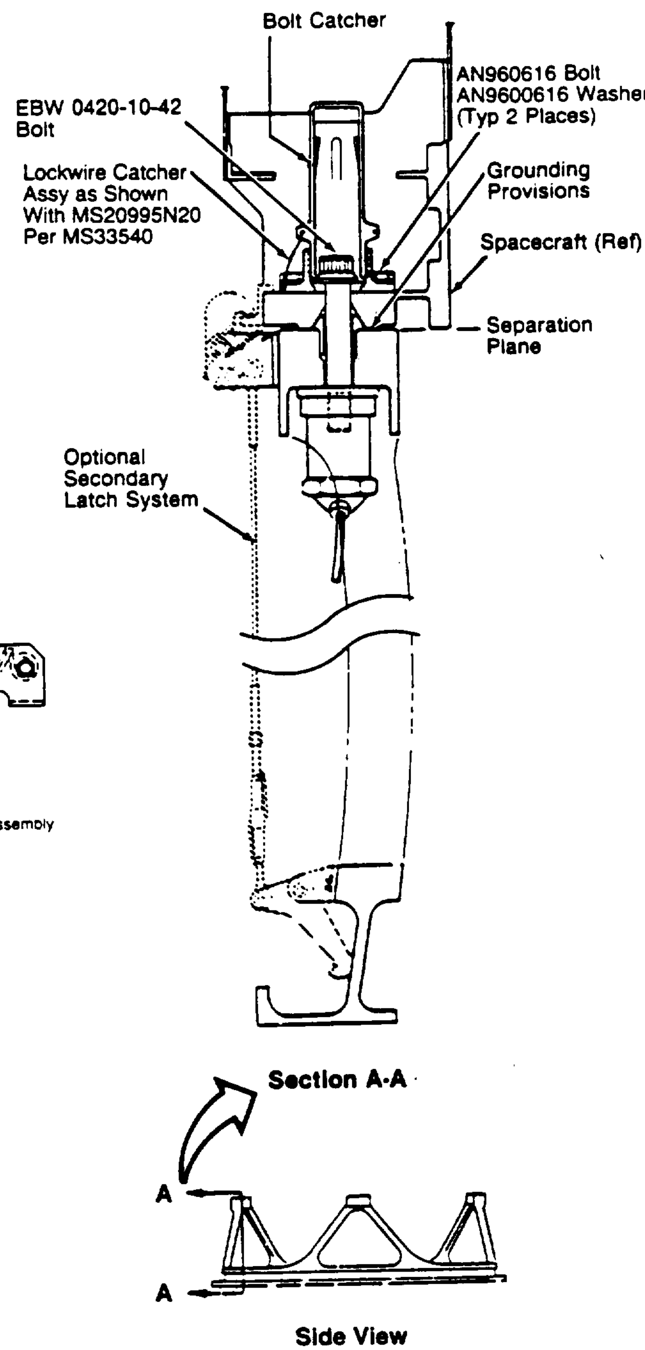


Figure 8-6. Delta PAF Clamp Release

Figure 8-7. Delta PAF Exploding Nut Release separation mechanism. It appears that the 10' standard Delta fairing and the hammerhead S-II fairing could accommodate this extra clearance requirement.

Design of the separation system will be a design challenge. The impact of scarring the heat shield to attach the separation device will be investigated but presents potential safety problems by tripping the boundary layer during reentry causing asymmetric ablation. Attitude dynamics models should be produced to verify the accuracy of typical separation techniques and innovative approaches. This effort will be accomplished during the Part II exercises.

8.3.3 Electrical/Data

Umbilical connectors are used to allow for transmission of power, telemetry, and commands between the payload and the blockhouse payload console. Typical Delta umbilicals link the payload console in the blockhouse to the RRS via quick-disconnect connectors. An umbilical connection is made between the blockhouse and 2nd stage and then routed to the RRS via a wire harness and finally connected to the payload using another umbilical connector (see Figure 8-8). Two such connections are allowed (i.e., Quad I and Quad III). These connectors detach just prior to liftoff (between blockhouse and 2nd stage) and during fairing separation (between 2nd stage and fairing) via disconnect lanyards (see Figure 8-9). The umbilicals are connected prior to fairing installation using the same configuration described earlier with additional, temporary, extensions. These extra cables are removed during fairing installation (T-3 days).

The Delta umbilical interface (see Figure 8-10) includes a 32-pin umbilical plug, a battery flight plug, and an ordnance arming plug. The 32-pins allow for 32 hard-lined spacecraft command, telemetry, and/or power connections. The battery plug is used to maintain spacecraft battery charge. The ordnance arming plug enables the explosive nut mechanism for umbilical separation via lanyards upon command from the payload console.

The RRS will require monitoring during pre-launch operations. Since the RRS is to be equipped with its own radio-frequency (RF) telemetry system it may be desirable to have significant data transmitted to the payload console directly. Fairings typically allow for installment of transparent fiberglass RF windows or panels. The Delta 10' fairing is made of three sections, of which one is fiberglass, allowing for RF telemetry capability. Critical parameters, pertaining to the health of the rodents, for example, should also be monitored via the umbilicals to guarantee accurate data on the launch pad (i.e., redundancy to RF telemetry).

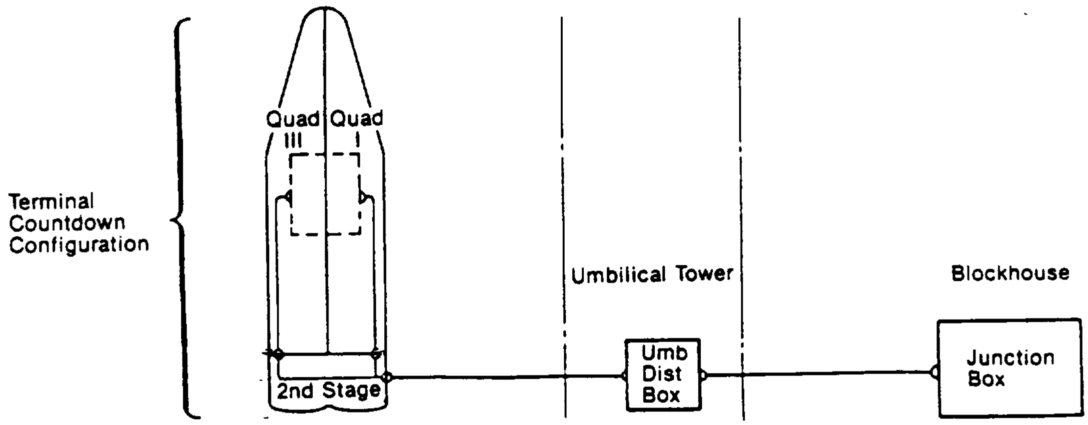


Figure 8-8. Umbilical Wiring

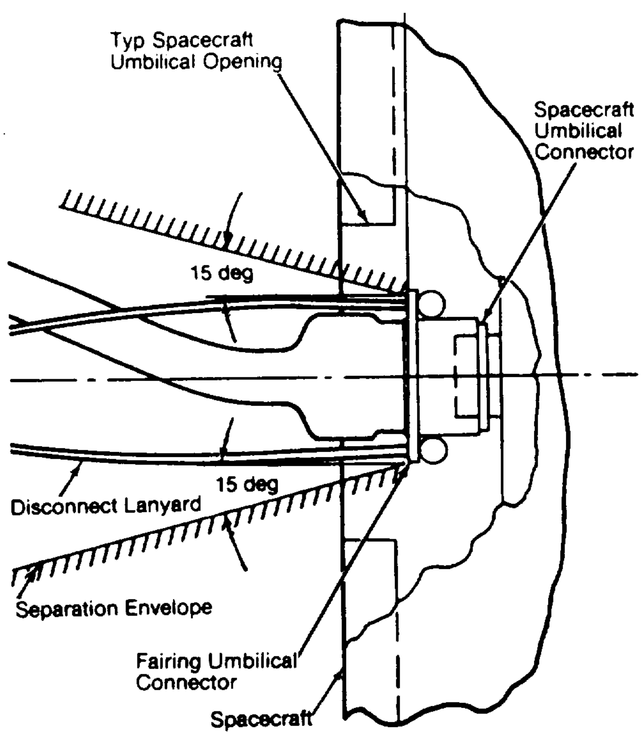


Figure 8-9. Umbilical Disconnection

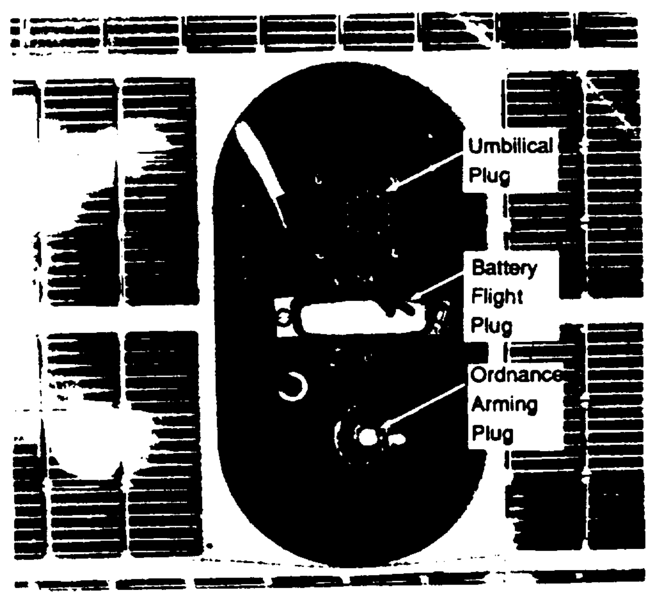


Figure 8-10. Umbilical Plug Interface

The RRS will require power to measure and transmit data during pre-launch. The power could be obtained via the 32-pin umbilical plug (each wire has a voltage rating of 600 volts DC) or indirectly from the battery plug. If the RRS uses a battery as a major source of power, the battery plug could be used to maintain charge during pre-launch.

The lanyard disconnects operate properly as long as they are oriented within a specific envelope. Lanyards disconnect as the fairing splits and must be oriented close to normal to the fairing wall (see Figure 8-9). Since, an umbilical connection cannot be made to the heat shield (i.e., no scars allowed), attachment could be made to the adapter or an attach point already required to employ the separation mechanism. Note, if the adapter sleeve plus removable nose fairing concept is employed the lanyard mechanisms could not be employed and manual detachment would be necessary (i.e., no fairing split). Consequently, umbilical disconnection presents a minor issue.

The spacecraft designer will have to supply the information pertaining to spacecraft-to-blockhouse wiring including:

- Number of wires required
- Pin assignments in the RRS umbilical connector
- Function of each wire (e.g., voltage, current, frequency requirements)
- Shield requirements for RF protection
- Voltage of RRS battery and polarity of battery ground
- Part and item numbers for RRS umbilical connectors
- Physical location of RRS umbilical
- Periods during which hard-line systems will be operated

The RRS wiring must be designed not to exceed the standards provided by each ELV in the fleet. The requirements for the Delta 6920 have been roughly defined herein. However, standard wiring for the S-II has not yet been determined.

8.3.4 Thermal/Cleanliness

RRS and fairing mating operations for Delta 6920 take place in the whiteroom of the mobile service tower (MST) where temperature and humidity is controlled. Once the fairing is installed a payload air distribution system (air conditioning) is typically installed via an umbilical which connects to a fairing access door to maintain temperature and humidity control. However, late access requirements will require the white room to provide thermal control until the white room is dissembled and the gantry rolled-back. The air-conditioning umbilical is ejected at liftoff via a lanyard disconnect and the access door automatically closes. It is expected that all of the candidate ELV's will provide air conditioning in a similar manner.

If the removable fairing and adapter sleeve concept is developed the RRS environmental control requirements may be more difficult to satisfy since the payload is exposed to the external atmosphere during EM installation. However, this is an extremely short period of time and should not pose significant cleanliness or humidity control problems provided the RRS is not subjected to excessive moisture during these minutes (i.e., rain). facility to protect the RRS.

During ascent on a Delta 6920 the internal fairing skin temperature will reach a maximum of about 110°F assuming standard acoustic blankets have been installed. The clearance required for the acoustic blankets has been included in the fairing unusable volume.

8.3.5 Flight Environment

Flight environment statistics for maximum acceleration and shock loads are presented in the candidate ELV data sheets. The maximum steady-state axial acceleration imposed by Delta ascent is approximately 6.7 g for a dedicated RRS launch without ballast. This load occurs at 2nd stage burn-out. The S-II should impose similar loads, crudely estimated at 6.5 g. This maximum loading occurs during the third stage burn. The Star-92 motor currently under study by Morton Thiokol is the S-II third stage baseline and is to incorporate a decreasing thrust pattern to reduce peak axial accelerations. The axial loads during ballistic re-entry are expected to exceed this range. Consequently, ascent accelerations will probably not drive RRS structural design criteria. However, all payload attach mechanisms must be designed for the ascent environment.

8.4 RRS/ELV Interface Analysis Summary

The focus of interface work was on the Delta 6920 since the S-II lacks detailed design maturity. MDSSC uses the Spacecraft Questionnaire to screen potential Delta costumers to identify significant interface issues. Only some of the major interfaces were analyzed herein.

The payload attach fittings available on Delta should easily accommodate RRS support requirements. A composite, cylindrical structure (analogous to SYLDA used on Ariane) could be used to attach the RRS to the PAF without scarring the heat shield. Use of standard separation mechanisms would require incorporating jettisonable or retractable protrusions (attach points). This system could weigh approximately 400 lbs (spacecraft chargeable).

The late installation requirement (T-12 hours) will be difficult to meet without use of a relatively large access panel. MDSSC is currently modifying the standard 10' fairing under the Delta II GSFC contract to incorporate access panels of ample size (36" X 60") to accommodate ROSAT. A portable gantry crane can be used to translate and lower the EM into place. However, the close-out requirement (T-4 hours) will require significant timeline adjustments on the Delta vehicle since gantry roll-back normally occurs between 5 and 7 hours before launch. This will be further assessed in the last half of the Phase B contract. Use of the large access panel was baselined for the Delta scenario since it will exist and does not appear to impose difficult access issues.

A removable fairing nose and interstage-like adapter sleeve could be developed as an alternative to the system just described; however, there will be a performance penalty as previously discussed (1,000 lbs estimated). Such a system may present some cost savings (i.e., more economical than standard fairing).

Scarring the heat shield was assumed unacceptable. However, scars to implement the separation system should be investigated. Also, detailed attitude dynamics analysis is required to determine if use of Delta-type separation mechanisms (i.e., clamp retainer and exploding nut) are viable.

Umbilical connectors can be used to hard-line critical telemetry and commands during pre-launch and provide power as needed. A RRS battery charge could be maintained via a battery flight plug if the battery is used as a power source on the pad. RF capability should also be incorporated during pre-launch operations to monitor the RRS. White room facilities and air conditioning umbilicals are available to maintain payload temperature and humidity. Since vertical integration is required, a portable white room facility is required on the gantry or work platform. It may be difficult to implement electrical umbilicals due to the lanyard disconnection envelope required unless manually disconnected.

9.0 FINAL ELV CANDIDATE ANALYSIS

After careful consideration of all viable ELV options and a more in-depth look at the Delta 6920 and S-II candidates, key launch vehicle discriminating factors were identified (see below). The ELV's typically demonstrated either a lack of performance, excessive cost and performance, or a lack of design maturity. All of the ELV's also presented moderate to serious late installation and access complications. None of the 19 ELV's analyzed were considered optimum for RRS launches.

- Cost
- Performance
- Development Maturity
- Payload Accessibility

9.1 ELV Candidate Issues

The best candidates, the Delta 6920 and S-II, present significant issues with respect to these discriminating factors. The Delta 6920 cost is estimated at $43 M. This value may escalate. Assuming the current requirement for late access up to T-4 hours is maintained, significant timeline adjustments must be produced by MDSSC. The new countdown procedures must also be coordinated with range safety. The standard 10' fairing offered by MDSSC will not provide sufficient access for late installation of the EM. However, modifications for the ROSAT program under contract from Goddard will enable installation of large enough access panels for EM installation without additional cost to the RRS program.

An alternative fairing concept for improving access conditions could be the development of a removable fairing and an interstage-like adapter sleeve. This system would provide convenient access via the removable fairing, nose piece. MDSSC has estimated that the cost to the RRS program of this system will be negligible over 30 flights; the development cost will be offset by the savings in unit cost. However, a significant performance penalty (1,000 lbs) is incurred by carrying the adapter sleeve to orbit, which makes this option less attractive.

One way to reduce the Delta unit cost would be to remove several Castor IVA solids (i.e., derate 6920) or GEMs (i.e., derate 7920). The basic cost of the 6920 or 7920 may be reduced by $3 M (Delta 6920) or $5 M (Delta 7920) in this manner; however, this savings is not as much as desired. McDonnell Douglas is currently evaluating this option (7300 series).

The S-II concept is based on Peacekeeper components and appears to present a considerably less expensive alternative to the Delta. However, the S-II severely lacks maturity. The Star-92 baselined for the S-II is currently only under study. EPAC has no experience with orbital launches. Furthermore, EPAC has no firm contracts to launch S series vehicles. The performance values have not been verified by an SAIC or Eagle ascent simulation. The pre-launch timeline has only been roughly conceptualized. Interface analysis on the S-II was limited due to a lack of design detail. Consequently, the S-II is an interesting concept for the RRS launch application; however, the S-II should not be considered an alternative to Delta due to a lack of design maturity.

9.2 Recent Launch Vehicle Developments

The launch vehicle environment has been significantly affected by the recent warming of relations between the United States and the Soviet Union. The DoD is considering substantial reductions in tactical missile inventories. Between 250 and 400 Minuteman II's are being considered for removal from silos to cut DoD costs. The Trident is replacing the Poseidon in nuclear submarines. The MX has been a political controversy, and its future is not clear. Peacekeeper solid components have already been made available for purchase. Use of existing missile components reduces development and unit costs significantly. Several companies could propose new ELV derivatives using Minuteman, Poseidon, and/or Peacekeeper stages to offer low cost launch vehicles. OSC is employing the Peacekeeper 1st stage in the Taurus vehicle selected as the SSLV for DARPA. EPAC has proposed using Peacekeeper stages in various S series vehicle configurations.

Consequently, a competitive procurement for RRS launches via a launch services contract could result in significant cost savings. It would be reasonable to expect ELV launch costs of $20 M. For example, the Taurus launch cost is presently $15 M. Adding another Peacekeeper stage or strap-ons could increase performance to an adequate level while increasing cost to approximately $20 M. This cost per flight would result in significant ELV life cycle cost savings as compared to a program using only Delta launch vehicles (i.e., $0.7 B over 30 flights, or a 50% cost reduction). Another benefit with issuing a launch services contract would be that these new vehicle concepts could be developed to meet RRS unique requirements (e.g., access requirements) at no expense to the RRS program.

This RRS-specific launch vehicle could be based entirely on existing and flight proven stages. The maturity or reliability of the launch vehicle may be in serious question if companies without previous ELV design experience are proposing a concept which includes new hardware elements. Even if no hardware development is proposed the reliability of a design established by an inexperienced company may be considered lower than for a design proposed by a company with a long history of launch vehicle design and launch experience. There are many companies with significant experience that could enter the competitive procurement. Several potential launch service contractors are listed below.

Boeing	Minuteman Integration Contractor
GD	Atlas Prime Contractor
Lockheed	Poseidon, Polaris, Trident Integration Contractor
MDSSC	Delta 6920 Contractor
MMC	Peacekeeper Integration Contractor, Titan Prime Contractor
OSC	Pegasus and Taurus Contractor
TRW	BMD Systems Contractor
Others	E-Prime, SSIA, AmRoc, PacAm, etc.

10.0 CONCLUSIONS

There is no optimum launch vehicle among the candidates studied. The most appropriate, operational launch vehicle is the Delta 6920. From a technical viewpoint the S-II also appeared desirable; however, this concept lacks design maturity. Recent launch vehicle developments indicate that a competitive procurement via a launch services contract may result in the development of a "tailor-made" launch vehicle, possibly derived from flight proven weapon system components at significant cost savings (50% of Delta cost estimated).

Shared launches present serious operational challenges (e.g., scheduling, access, orbital operations) and minor cost savings over small dedicated launchers. Shared launches may be desirable only if offered for political reasons at no cost to the RRS program and if the primary payload(s) can be delivered to the same orbital inclination as RRS (i.e., 34°).

The Delta 6920 could employ the 10' standard fairing with the large access panel, being developed for the ROSAT program, to handle the late EM installation requirement (T-12 hours). Significant modifications to the pre-launch timeline, if possible, are necessary to meet the close-out requirement (T-4 hours) which will result in some cost penalty. Other fairing modifications (e.g., removable nose cap) may be necessary to meet the close-out requirement (i.e., if the access panel is not compatible with the necessary timeline adjustments).

The RRS should be supported within the fairing using a composite structure analogous to the SYLDA used on Ariane to separate and support payloads in a dual launch configuration. A PAF will connect the support structure to the ELV. The separation system must be attached above the heat shield and will require a significant design effort despite use of existing separation mechanisms. Scarring the heat shield to connect the separation mechanisms would eliminate the need for protruding attach points which must be retracted or jettisoned after separation but may create other problems. The total chargeable weight to RRS is estimated to be a maximum of 400 lbs.

The alternative concept for improved access analyzed herein included a removable fairing for convenient access and an interstage-like adapter sleeve to support the payload and handle aerodynamic loads. Despite some access advantages (i.e., crane installs EM without need for gantry crane; fairing nose may be installed after close-out) and negligible cost to the RRS program, this concept presents a significant performance penalty (1,000 lbs estimated) since the adapter sleeve is carried to orbit as previously discussed.

Umbilicals are typically available for electrical, data, thermal control interfaces. No significant issues are expected with respect to these interfaces or the flight environment.

11.0 RECOMMENDATIONS

RRS design should proceed in Part II of the Phase B contract with a launch vehicle interface design compatible with the Delta. Due to the lack of detail available for the S-II concept, the analysis of specificatiốns should focus on the Delta vehicle. An effort should be made to address as many of the interface issues presented in the Spacecraft Questionnaire used by MDSSC as possible.

A significant issue for Delta which will require further input from MDSSC will be pre-launch timeline adjustments. Separation system design will also be a significant issue. Scarring the heat shield to attach the separation system should be studied. A more detailed structural analysis should be conducted to define the support structure size and mass. Specifications for other interfaces (i.e, electrical, data, thermal, etc.) should be determined as spacecraft design detail is available.

NASA should consider a launch services contractor competition to reduce cost per flight and life cycle costs.

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APPENDIX A

DELTA 6920 PRE-LAUNCH TIMELINE

The prelaunch activities for Delta 6920 during the final days before launch (starting at T-6 days) are highlighted in the schedules on the following pages. These schedules have been included to demonstrate the types of activities involved during the final days of prelaunch. hazardous access periods are indicated throughout the schedules.

Significant launch site payload processing events prior to T-6 days are listed below. Payload checkout activities after T-6 days are conducted in the MST white room.

T-16 days	S/C transported to explosive-safe facility for hazardous payload systems preparation
T-10 days	S/C ready to mount to Delta launcher
T-8 days	S/C flight weight measured
T-7 days	S/C handling canister assembled

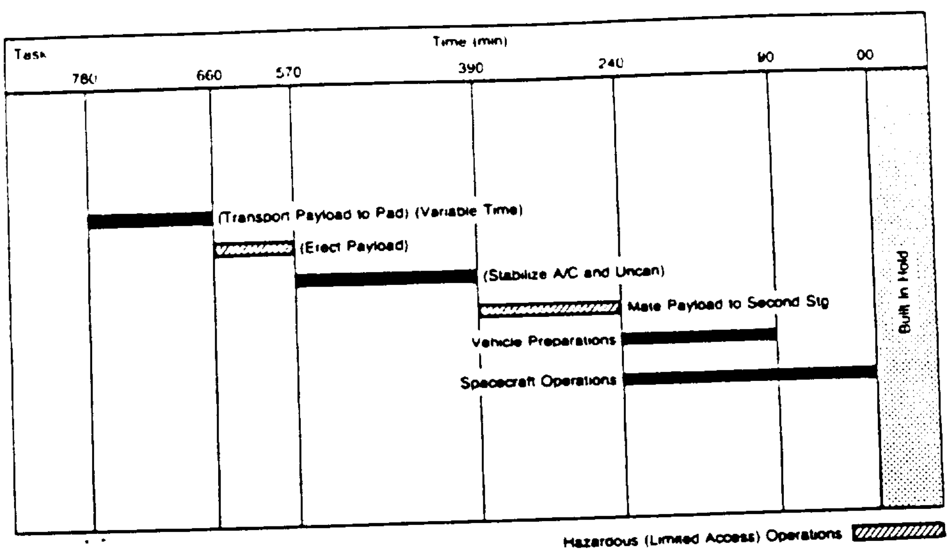

Figure A-1. Typical Schedule of Activities F-6 Day

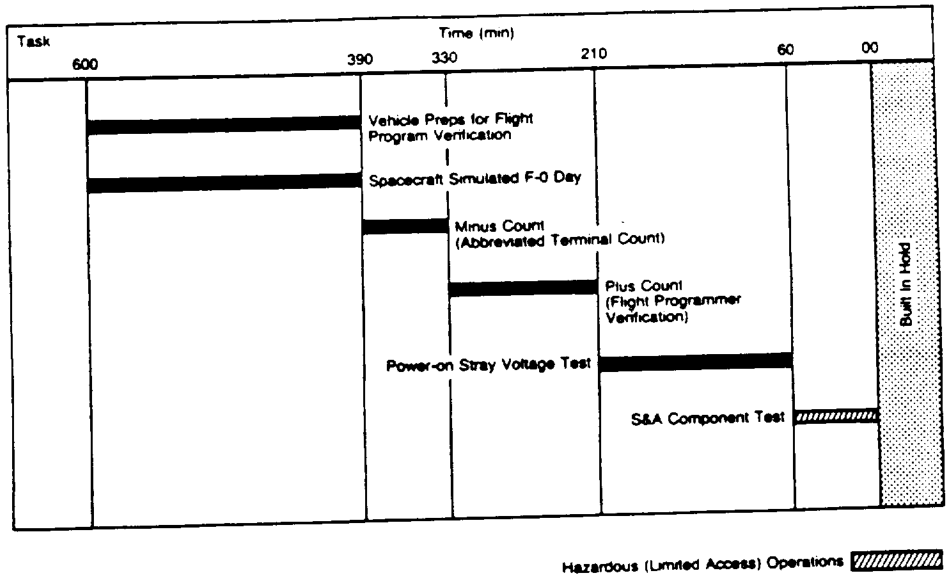

Figure A-2. Typical Schedule of Activities F-5 Day

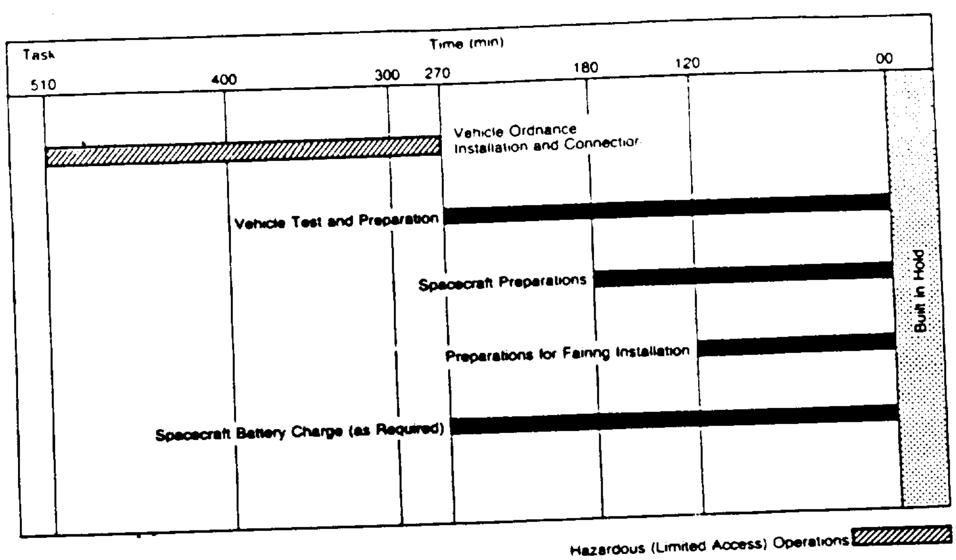

Figure A-3. Typical Schedule of Activities F-4 Day

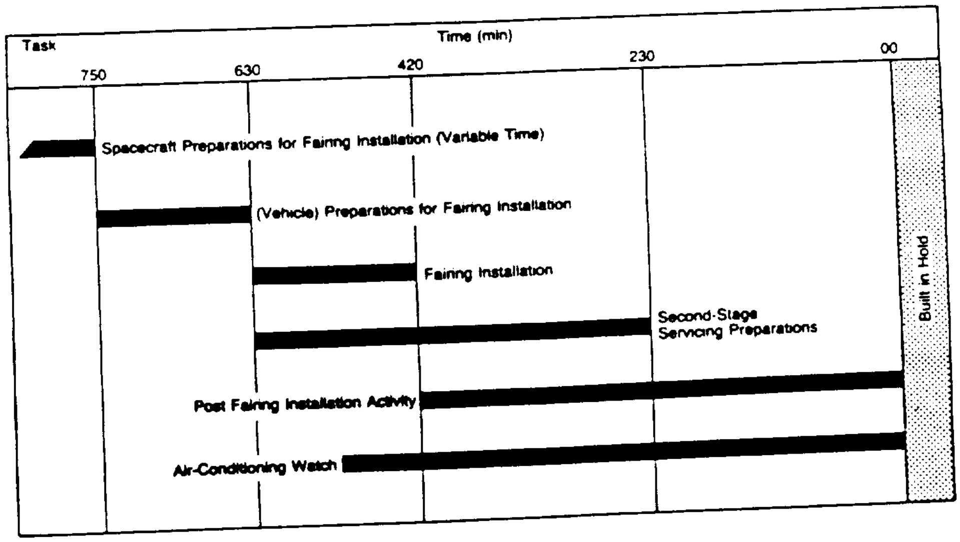

Figure A-4. Typical Schedule of Activities F-3 Day

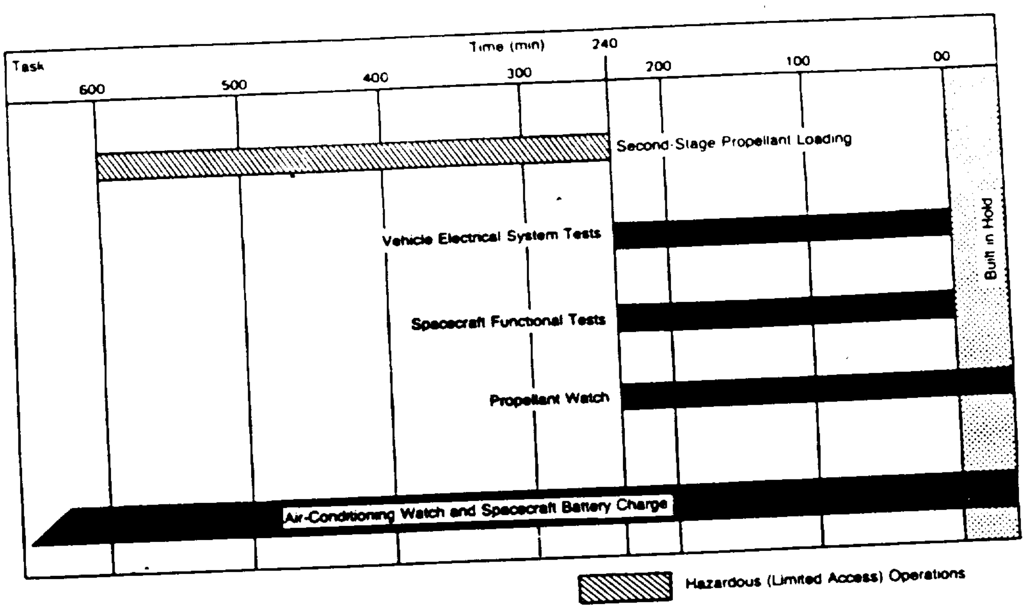

Figure A-5. Typical Schedule of Activities F-2 Day

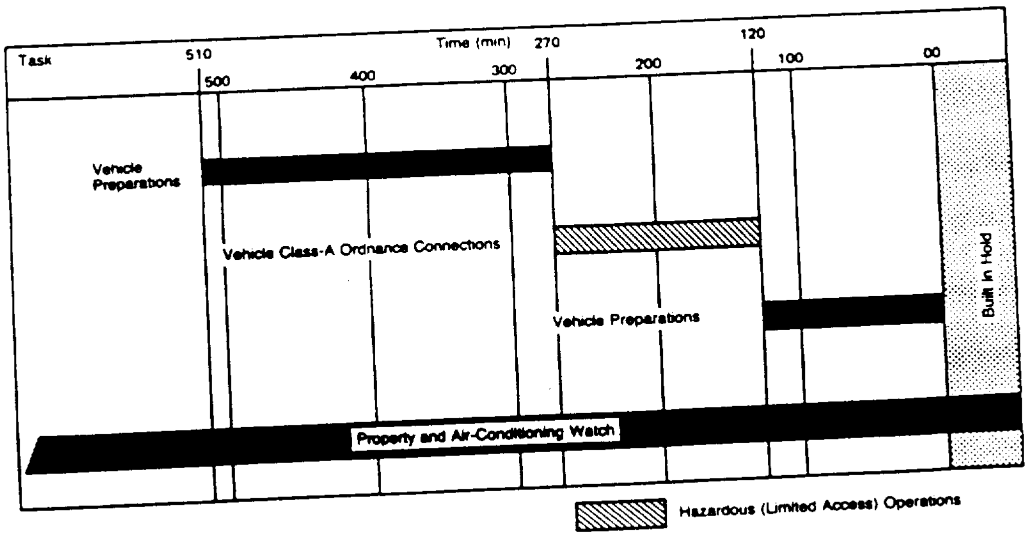

Figure A-6. Typical Schedule of Activities F-1 Day

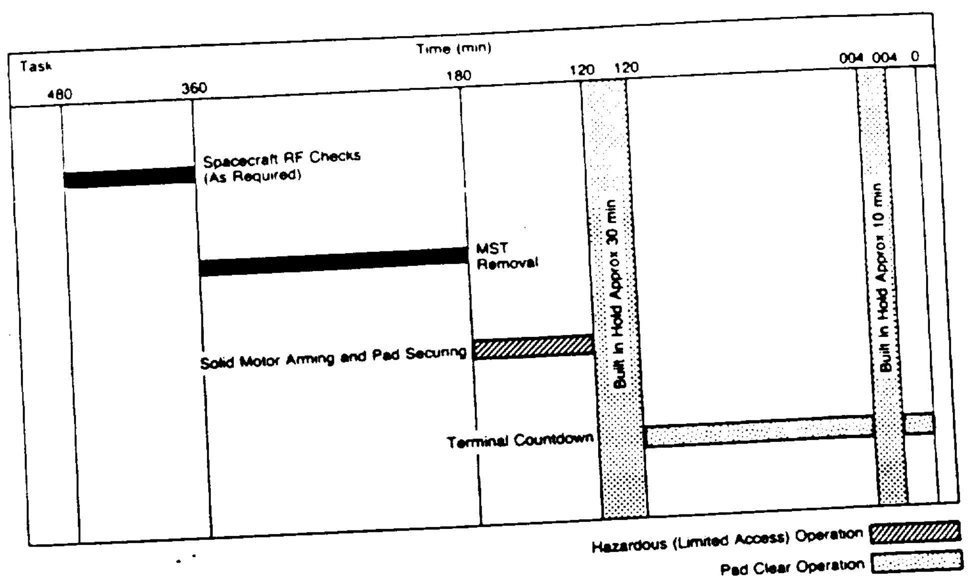

Figure A-7. Typical Schedule of Activities F-0 Day