~~6-6812-73~~
7-2976-73

BIF003W/2-069331-73
31 August 1973
Copy No. 6
Sheet Count: 117

FINAL REPORT
FOR

DESIGN STUDY FOR HEXAGON RECONNAISSANCE SYSTEM USING STS (TS)

31 AUGUST 1973

X2509

HANDLE VIA BYEMAN
CONTROL SYSTEM ONLY

FOREWORD

This document is the final report on the Hexagon Satellite Vehicle/Space Transportation System study undertaken by Lockheed Missiles and Space Company, Inc. (LMSC). The work was performed under Contract F18600-73-C-0074, to the requirements of Statement of Work BYE 73-14.

CONTENTS

Section Page

GLOSSARY

ACS	attitude control system
AGE	Aerospace Ground Equipment
BCS	backup command system
CCC	charge current controller
DBS	doppler beacon system
DIU	data interface unit
ECS	extended command system
EDAP	electrical distribution and power
EMC	electromagnetic compatibility
EVA	extra-vehicular activity
FOSR	flexible optical solar reflector
GFE	Government Furnished Equipment
GPC	general purpose computer
GRD	ground resolved distance
GSE	ground support equipment
HYD	hydrazine
IRA	inertial reference assembly
LB	Lifeboat (system)
LMSC	Lockheed Missiles & Space Company, Inc.
MCM	Mapping Camera Module
MCS	minimal command system
MSI	medium scale integration
MST	Missile Service Tower
NC	normally closed
nm	nautical mile
NO	normally open
NRS	non-resupply
NSPC	normally stored program command

OA	orbit adjust (engine)
OAS	orbit adjust system
OMS	orbital maneuvering subsystem
PCM	pulse code modulation
P-E	Perkin-Elmer, Inc.
P/L	payload
PMD	propellant management device
PRAT	Product Reliability Assessment Test
PSPC	protected stored program command
RCS	reaction control system
REA	reaction engine assembly
RF	radio frequency
RMS	Remote Manipulator System (also root mean square)
RS	resupply
RTS	realtime command
RV	recovery vehicle
SBA	Satellite Basic Assembly
SCF	Satellite Control Facility
SCS	Satellite Control Section
SGLS	Space-Ground Link System
SI	stellar index
SLV	Standard Launch Vehicle
SPC	stored program command
SPSPC	secure protected stored program command
SRB	solid rocket booster
SRTC	secure realtime command
SRU	Space Replaceable Unit
SRV	Satellite Recovery Vehicle
STC	Satellite Test Center
STS	Space Transportation System
SU	supply unit
SV	Satellite Vehicle

TCEA	temperature control electronics assembly
TT&C	tracking, telemetry, and command
TU	takeup
TVC	thrust vector control
VIS	Vertical Integration Stand
VSWR	voltage standing wave ratio
WTR	Western Test Range

Section 1
INTRODUCTION

1.1 STUDY OBJECTIVES AND BACKGROUND

This final report presents the results of a 17-week study to establish operational and design concepts and associated system cost for a future Hexagon Satellite Vehicle (SV) system designed specifically for use with the Space Transportation System (STS). Three operational concepts were considered singularly and in combination:

- Resupply: on-orbit replacement of expendables Anderson see?
- Maintenance: on-orbit replacement of failed or life-limited items
- Refurbishment: return to earth and restoration to flight configuration

The purpose of this study is to assist the Customer in examination of potential operational and economic benefits of STS usage within the framework of requirements postulated for the 1980s.

A prior study on compatibility of the Hexagon SV with the STS was completed on 28 January 1972. The objectives of that study were to develop and describe the minimum modifications required to the SV and its supporting AGE and facilities to make them compatible with the STS, and to estimate incremental costs associated with these modifications. Two primary modes of SV/STS operation were considered: (1) Booster Substitution, where the STS is used only as a booster; (2) Boost/Retrieval, where the STS is used as an SV booster and a retrieval vehicle, with refurbishment and reuse of the SV after retrieval. This study is therefore a follow-on to further examine the extent of SV design change which would more fully utilize the capability of the STS.

1.2 STUDY ORGANIZATIONS

The study was conducted as a team effort by Perkin-Elmer, Inc. (P-E) (camera system contractor) and LMSC. The latter served as the Study Integrating Contractor. Each contractor established a designated Study Leader within his Hexagon activity, and made necessary security arrangements to segregate this 10116 study from the on-going Hexagon program work. Other contractors who provide equipment for the program, but who did not participate in this study, were represented by LMSC. Typical of the contractors so represented are:

General Electric, Aerospace Electronics Systems Department, contractor for the Extended Command System (ECS)

McDonnell/Douglas Astronautics Co., contractor for the Hexagon Recovery Vehicle (RV).

Both contractors (LMSC and P-E) were responsible for defining the design changes to their equipment in support of the study objective. LMSC was additionally responsible for developing the interface between the SV and the STS. Both contractors identified the one-time tests required for all development and qualification, as well as those operations and tests required during the refurbishment activities. The current Hexagon Block III SV (13-18) was used as the point of design departure for the study.

The final reporting on the study is being accomplished separately by P-E and LMSC. However, a brief description of P-E's camera system design and program cost is included within Section 2 of this LMSC report.

1.3 STUDY APPROACH AND SCHEDULE

The first half of the study period was devoted to determining the operational concept (on-orbit resupply/maintenance versus refurbishment-only) and selecting a Satellite Vehicle configuration. The results were presented at the mid term briefing. Figure 1-1 shows the work flow that generated the operational concept and SV configuration. The second portion of the study covered the refinement of the selected SV and subsystems design, SV/STS compatibility, and definition of supporting AGE, facilities, tests, schedules, and costs. A decision-dependent area of work, the

~~TOP SECRET/H/10116~~

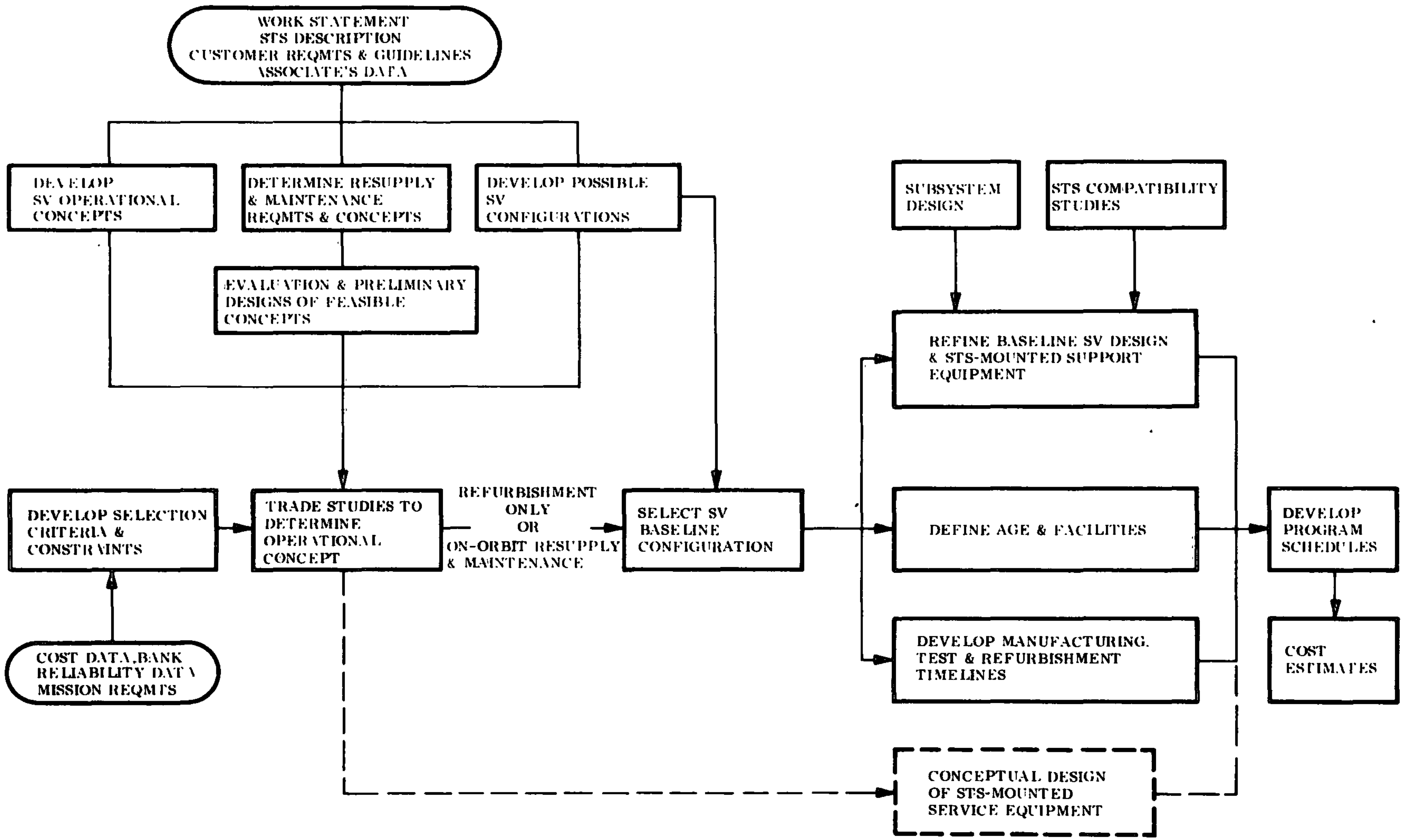


Fig. 1-1 Hexagon/STS Study Approach

~~TOP SECRET/H~~7 10116

BIF003W/2-069331-73

conceptual design of STS-mounted service equipment, is shown in dashed lines on the work flow depicted in Fig. 1-1. If trade studies had shown that resupply/ maintenance was the preferred operational concept, then the work on STS-mounted service equipment would have been accomplished.

P-E and LMSC developed estimates of non-recurring and recurring costs for the total SV/STS program. Costs were based upon a phase-in from the existing Titan IIID-launched Hexagon SV program. No costs were developed for STS launch, Western Test Range (WTR) facilities, RVs, or tracking network support; these were assumed to be GFE.

The major milestones for the study were as follows:

Study initiation:	28 March 1973
Mid term briefing:	5 June 1973
Final briefing:	26 July 1973
Final report	31 August 1973

1.4 GENERAL CRITERIA

The Customer set forth the general criteria against which the study was conducted. Additional criteria were formulated by LMSC; they were derived from the Work Statement and through discussions with the Customer and Aerospace Corporation.

assumption

? Two missions per year will be conducted, each mission a minimum of 120 days duration, with the desire that continuous operational capability exist. Each mission will result in the stereo exposure of the same amount of film as currently flown on a Hexagon mission (104,000 ft per camera). The quarterly, semi-annual, and annual coverage requirements of the intelligence community are assumed satisfied by recovering each quarter's coverage separately. Best photographic resolution of targets will be equal to that currently being accomplished at 82 nm (2.27 ft GRD at nadir).

over taken in study by * on page 2-7

The first Hexagon SV/STS launch will take place in 1982 from WTR. SV/STS return from orbit was also assumed to take place at WTR. A ten-year operational program will be costed, with provision made for a continuing program beyond the 1992 cutoff for pricing.

The size, type, and quantity of RVs to be employed were not specified, nor was the frequency of data return specified except in the case of quarterly coverage requirements. However, all film must be returned by RVs, except that the last portion of the mission take could be retained on board the SV and returned by the STS during an SV retrieval or resupply mission.

The SV will maintain the capability to carry survivability aids, and auxiliary and piggyback payloads. Accordingly, approximately 5 percent of the SV fixed weight was allocated for survivability aids (750 lb) and 800 lb was allotted for piggyback payloads, i.e., subsatellites. The Mapping Camera Module currently being flown is not part of the SV/STS configuration.

The Satellite Control Facility (SCF) network will be used to control and monitor the STS and SV on-orbit operations. No additional stations beyond those currently existing are contemplated, nor are data relay satellites considered.

The reliability goal for the Block III Hexagon SV is 0.85 for 60 days (excluding camera and RV separation systems). The SV for STS operations will also have a reliability goal of 0.85 for the orbital life of the SV. The SV deployment/retrieval operations will have a higher reliability goal.

All SVs will have a deboost capability similar to the present system. In the event an SV cannot be retrieved by the STS it would then be deboosted from orbit into a deep ocean area. The study did not consider backup or pipeline vehicles which might be required to replace a failed or lost SV.

All vehicles were assumed to be launched into the same basic sun-synchronous orbit currently employed by the Hexagon program: 96.4 deg inclination with the argument of perigee being located at 45° North latitude.

? interesting

An alternate operational concept, the "non-overflight option," was to be investigated to determine its effect on selected SV design and operations. No design or costing activities were required. The option involves launch of an SV into, or retrieval from, a 104 deg inclination orbit with a one-orbit flight of the STS. The selected orbit inclination eliminates STS overflight of the Sino-Soviet area.

Section 2
SUMMARY

This section provides an overview of the hardware and functional elements of the Hexagon SV, optimized for STS operations. It describes the STS configuration and its capabilities, as well as the Hexagon Block III SV. A summary of the selection of an operational concept (on-orbit resupply/maintenance versus refurbishment-only), and the SV factory, launch, on-orbit, retrieval, and post-retrieval operations are presented. Overall program schedules and cost estimates are also shown, plus suggested areas of work for subsequent studies.

2.1 SV/STS SYSTEM SUMMARY

This section summarizes the STS and the Hexagon Block III SV configurations employed in the study.

2.1.1 STS Configuration

The STS is a two-stage space launch vehicle, composed of a reusable, manned Orbiter and an unmanned, recoverable booster. The STS will operate between the surface of the earth and low earth orbit. As shown in Fig. 2-1, the Orbiter with its crew and payload, is mounted "piggyback" to the single, expendable tank which contains all of the hydrogen and oxygen propellants used by the Orbiter rocket engines during the ascent phase of flight. The two solid rockets that comprise the booster are located under the wings of the Orbiter and are attached directly to the propellant tank.

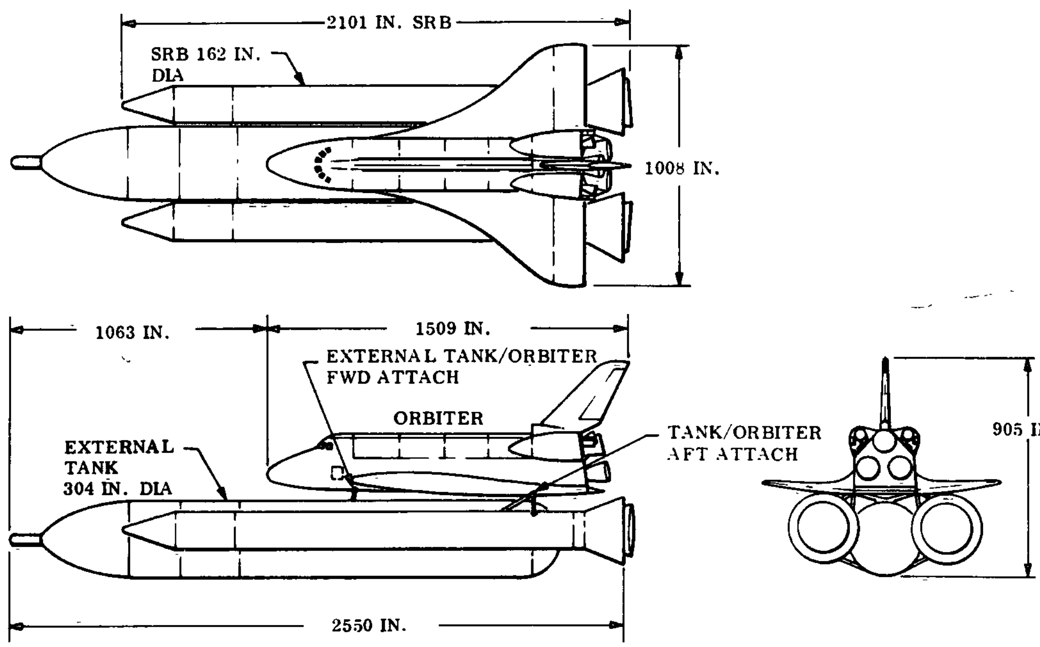


Fig. 2-1 Baseline STS Configuration

Liftoff thrust is provided by parallel burning of the solid rocket boosters (SRB) and Orbiter main engines. Guidance and control through the boost phase is provided by Orbiter main engine thrust vector control (TVC), SRB TVC, and elevon deflection. At SRB staging (approximately 162,000 ft), auxiliary rockets are fired to accelerate the expended SRB cases from the vehicle. The cases follow a ballistic trajectory after separation, are decelerated by parachute, and are recovered after water landing. The three Orbiter main engines continue firing to orbit injection at 50 miles altitude. The external tank is separated after injection, and its de-orbit motor is fired to place it into a trajectory with impact in an unpopulated designated ocean area.

The capability for intact Orbiter recovery in the event of premature mission termination is provided throughout the entire mission sequence. Intact abort capability is provided from lift-off to SRB burnout by the SRBs with TVC. The main engine will be shut down at abort initiation and the SRBs will provide sufficient control and thrust to continue ascent until SRB burnout, after which an abort glide-back is achieved. In the regime from SRB separation to orbital injection, return to launch site following an abort is accomplished by Orbiter separation and glide-back, or by continued flight using the orbital maneuvering subsystem (OMS) and the main engines into a direct return or once-around abort trajectory.

The orbit characteristics of the Orbiter vehicle are adjusted as required by using the OMS. On completing mission operations, the OMS is fired to initiate de-orbit and establish an entry trajectory. The Orbiter achieves required cross range by energy management, and returns to base where the vehicle is landed in a manner similar to that of high-performance aircraft.

Significant characteristics and capabilities of the Customer-provided STS model are:

- Payload bay 15 ft in diameter by 60 ft long
- 36,400-lb payload capability into a 50 x 100 nm, 96.4 deg sun-synchronous orbit
- Launch and recovery at Western Test Range.

The STS standard equipment includes a Remote Manipulator System (RMS) which is available for payload deployment, retrieval, and on-orbit resupply/maintenance.

2.1.2 Hexagon Block III Configuration

The Block III Hexagon SV includes a camera system, recovery vehicles (RVs), and the Mapping Camera Module (MCM). The Titan IIID is employed to launch the SV from WTR launch complex SLC-4E. The first Block III vehiches will be launched in 1976.

The SV (Fig. 2-2) is divided into three major sections: Aft or Satellite Control Section (SCS), Mid Section, and Forward Section. The SCS is devoted primarily

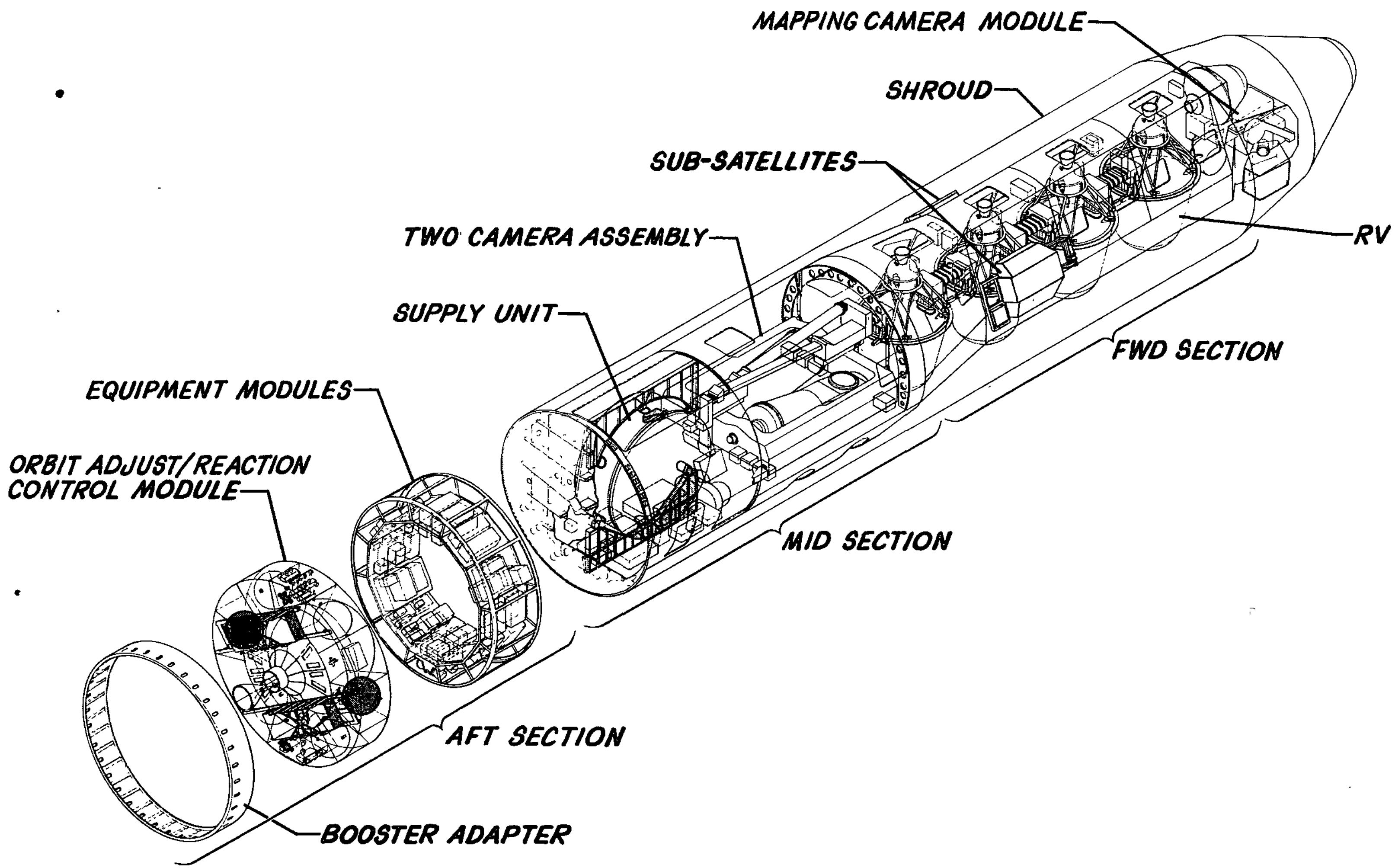


Fig. 2-2 Hexagon Satellite Vehicle

~~TOP SECRET/H~~/10116

BIF003W/2-069331-73

to spacecraft support and is divided into modules that house the support subsystem equipment. The SCS subsystems include: electrical power and distribution; propulsion; reaction control; orbit adjust; Lifeboat; attitude control; and tracking, telemetry, and command. The Mid Section houses the camera system and its support equipment. The Forward Section carries the four main RVs and the MCM which mounts on the front bulkhead. In the current Hexagon SV application, a shroud protects the Mid and Forward sections against launch and ascent environments; the shroud is ejected after Titan IIID Stage II ignition.

2.2 OPERATIONAL CONCEPTS

LMSC's approach to selecting an operational concept (on-orbit resupply/maintenance, refurbishment-only, or combination thereof) was accomplished by employing three parallel areas of work which led to a trade study (ref. Fig. 1-1). Specifically, (1) on-orbit resupply/maintenance concepts were analyzed and preliminary designs of feasible concepts accomplished; (2) conceptual vehicle configurations were generated in resupply/maintenance and non-resupply versions; and (3) operational characteristics for candidate vehicles were established, i.e., orbital life and operating orbits. This data was subjected to trade studies. Cost was determined to be the most significant variable among configuration/operating concepts, and was thus employed as the selection parameter.

2.2.1 On-Orbit Resupply/Maintenance

The resupply and maintenance study was approached by assuming that an SV consists of three sizes of Space Replaceable Units (SRU): modules, subsections, and sections. A module was estimated to be very similar to current SCS modules. A subsection represented a large removable item of equipment (e.g., an RV or camera system supply unit). A section was defined to be an entire slice of the vehicle similar to the SCS or the Forward Section of the Block III SV. Fluid and pressurant transfer were also considered. It is important to note that approximately half of the Hexagon vehicle weight is in expendables (fuel, film, RVs, etc.). This factor was found to be important in the operational concept study. Most resupply/maintenance studies to date have concentrated on vehicles that operate at higher altitudes and have a much smaller portion of their total weight allocated to expendables.

Three modes of SV/STS operation were established for on-orbit resupply/maintenance:

(1) Formation Flying: no physical coupling between the SV and STS, which requires maintaining constant separation distances and relative attitudes with both SV and STS reaction control systems active.

(2) Soft Dock: SV/STS spatial orientation provided by minimum of one RMS arm.

(3) Hard Dock: SV rigidly attached to the STS by a docking collar of NASA neutered type.

The methods of accomplishing the resupply and/or maintenance, were: (1) extra vehicular activity (EVA) by an astronaut, (2) non-EVA with use of the RMS, and (3) non-EVA with use of special equipment mounted in the STS payload bay and provided by the using program. It was concluded that if on-orbit maintenance and/or resupply is to be accomplished the SV should be hard docked to the STS. EVA was eliminated as a direct means of resupply and/or maintenance, but additional studies are suggested to determine the possible benefits of an astronaut in a monitoring or control function. Exchange of SRUs would be accomplished by program-provided special equipment.

Figure 2-3 shows a concept for a Hexagon SV configured for on-orbit resupply and maintenance. The SCS replacements are stored in a rotating storage unit with direct transfer mechanisms at each storage location. The SV also rotates for access. The Forward Section is reconfigured to include the Supply Unit and is replaced as a complete section.

2.2.2 Candidate SV Configurations

Candidate Hexagon SV configurations were created to conduct trade studies on operational concepts of resupply vs. non-resupply. Each configuration was

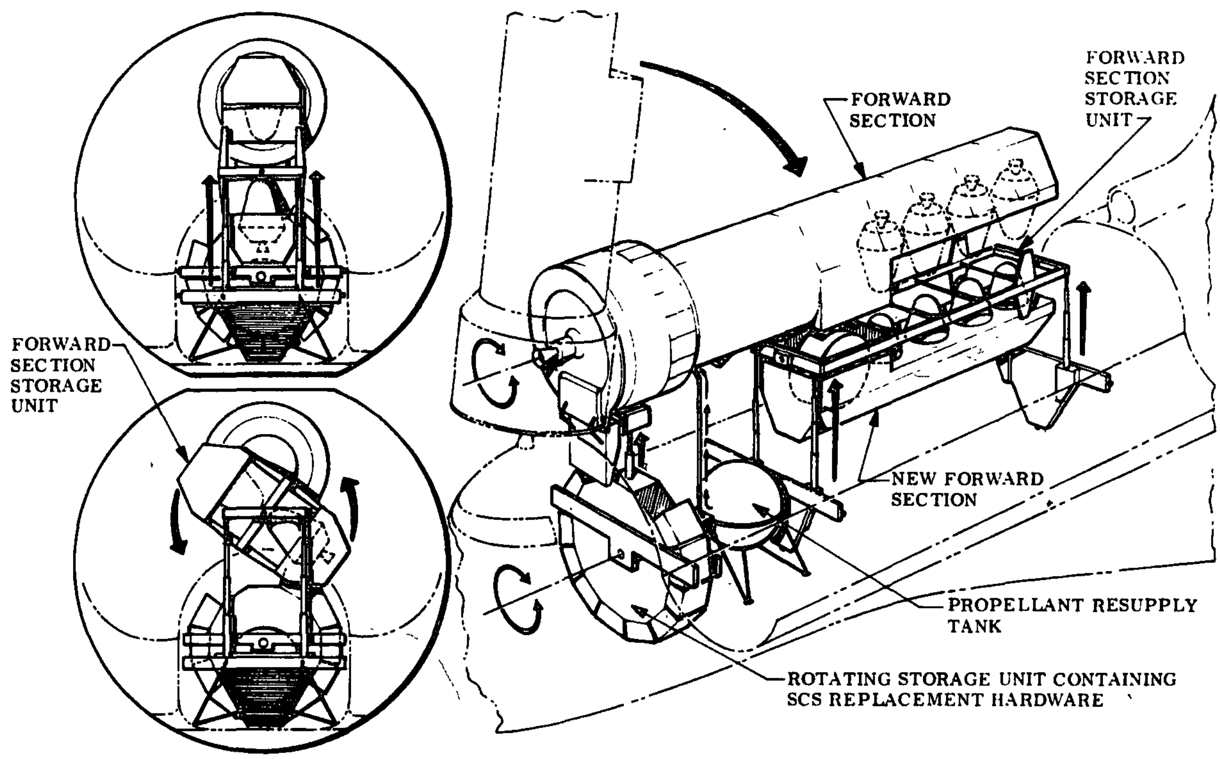


Fig. 2-3 Resupply and Maintenance Concept

designed to fully utilize the STS payload weight capability. Propellant was off-loaded as # of RVs were added, necessitating a corresponding increase in the operational orbit altitude. Orbit durations were established by the fact that four RV loads of film are to be exposed and returned every six months. Perkin-Elmer, working in conjunction with LMSC, studied camera system design changes required to provide current ground resolution when the SV is operated in orbits higher than the current 82 x 144 nm.

2.2.3 Concept Selection

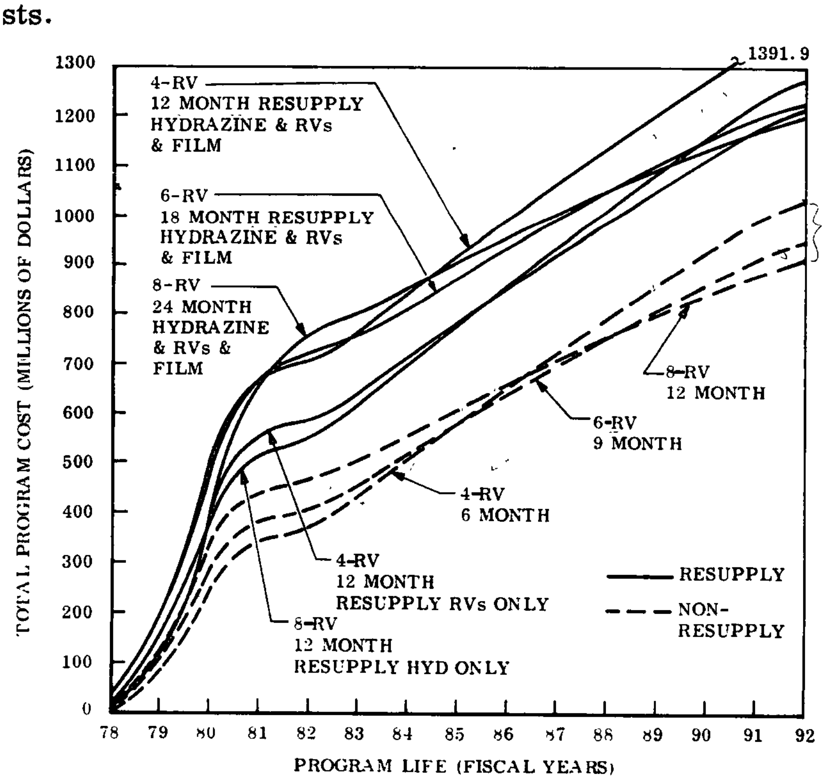
Since all configurations met the same mission requirements (coverage, resolution, etc.), other variables were identified for tradeoff to facilitate operational concept selection. Conceptually, all identified configurations seemed technically feasible.

How does this work? especially on non-recurring costs?

Cost was the most significant variable among configurations. Since a detailed design is required for bottom-up costing, a top-down method was used at this time. Historically, weight has been shown to be a relatively constant factor in cost estimation of aerospace systems, and was therefore employed in this preliminary cost analysis. The method used may not yield a precise total cost, but it is valid for a configuration-to-configuration comparison.

Figure 2-4 shows the cumulative cost curves for all configurations studied. In every case the resupply/maintenance concept is more costly than refurbishment only. The primary cost driver is the non-recurring cost for development of resupply kits, special STS-mounted equipment, and configuring the SV for resupply/maintenance. Annual recurring costs for either operational concept are approximately the same. Therefore, a non-resupply operational concept was selected.

The total costs of the non-resupply configurations are so close that a choice based on cost alone is not clear, but it is evident that longer-life configurations have lower recurring costs.



slope is the same ∴ recurring about same. non-recurring of resupply is high.

Fig. 2-4 Cost Summary

2.3 SELECTED SV CONFIGURATION

Selection of the non-resupply SV configuration was made by a series of trade studies which considered SV frontal area, propellant usage, orbit characteristics (including repeat cycle), STS effective payload weight capability, and ease of refurbishment. This selection process yielded the 6-RV configuration because it: L(1) has an acceptable range of orbits that yield good repeat cycles for photographic access, (2) has the longest practical life, and (3) requires only a reasonable scaling of the current optical system design. ? okay as of now.

Figure 2-5 is a preliminary design of the selected 6-RV Satellite Vehicle optimized for STS operations and refurbishment. Salient characteristics and performance parameters are identified. The significant changes from the current SLV-launched SV are: (1) new structure compatible with STS mounting, (2) addition of two RVs and related film capacity, (3) improved access, and (4) larger camera system, including additional redundancy. The SV subsystems in many cases are the same as Block III Hexagon. no MFT

P-E examined the application of SO-446 film for this configuration and found that it would not provide 2.27 ft GRD at 120 nm altitude without optical bar modification. Since SO-1414 is the film currently used and SO-446 is still in development, P-E elected to employ SO-1414 as a basis for design. However, LMSC believes that further detailed studies should be accomplished on the application of SO-446 to the Hexagon SV/STS program and on the degree to which the optical bars should be modified for a 120 nm orbit. A disagree: 446 will be available ∴ no devel costs

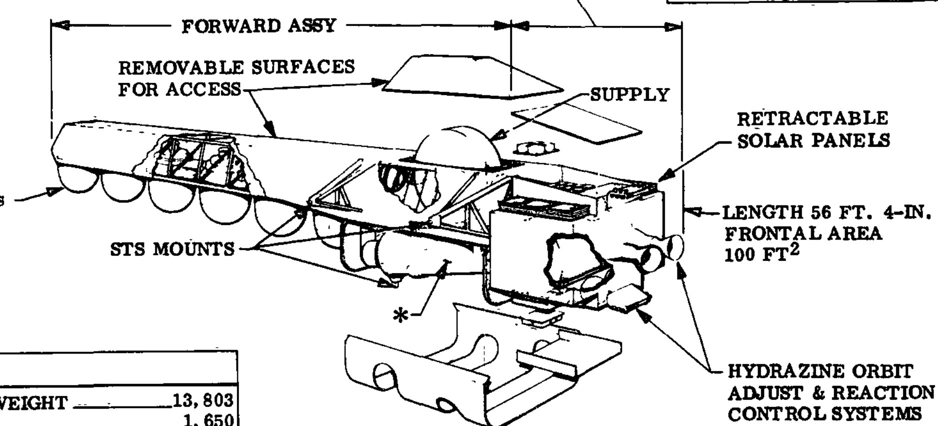
The orbit parameters attainable with the selected 6-RV Satellite Vehicle are shown by Fig. 2-6. Any combination of perigee-apogee and sync cycle below the operating range line may be utilized. Daily propellant usage includes that required for maintenance of perigee location.

PERFORMANCE

ORBITAL LIFE 9 MOS.
COVERAGE 36.0 M SQ N. M.
RESOLUTION 2.27 FT (NADIR)
PERIGEE RANGE 110-120 N. M.
LMSC EQUIPMENT REL. .85(9 MOS)
TOTAL SV RELIABILITY .56(9 MOS)

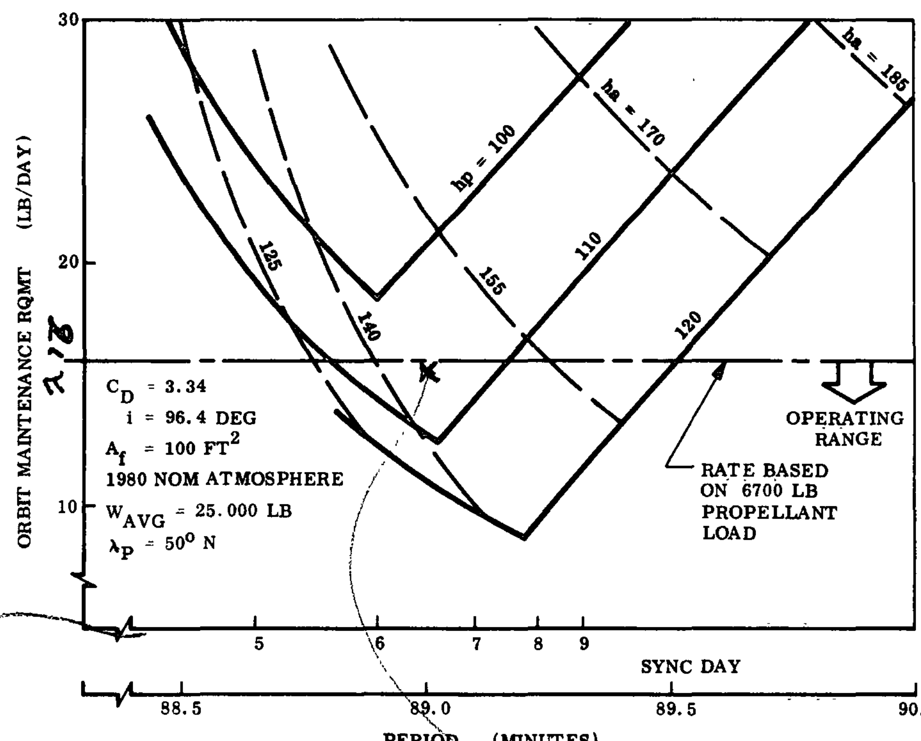
* OPTICAL BARS

	SV/STS	BLK III
FOCAL L.	72 IN.	60 IN.
F NO.	2.6	3.0
FILM	SO 1414	SO 1414
LINES/MM	176	139



WEIGHTS (LB)	
SV & STS FIXED WEIGHT	13,803
CONTINGENCY	1,650
SUBSATS & SURVIVABILITY AIDS	1,550
RVs & TAKEUPS	6,240
FILM	2,700
PROPELLANTS & GASES	7,407
TOTAL	33,350

Fig. 2-5 Satellite Vehicle



what is this

Fig. 2-6 SV Operational Orbits

9 months at 105 n mi x 147 n m.

2.4 SV/STS INTERFACES

The selected SV has been designed to be compatible with the STS as described above. All SV c.g. conditions (launch, abort, and retrieval) are within the STS payload c.g. range. Existing STS payload mounts can be employed and the loads reacted into these mounts are within STS design limits. Electrical interfaces are compatible with the STS design. The singular incompatibility is in the area of propellant and pressurant dump. For STS and crew safety in an abort or SV retrieval mode, LMSC recommends that all fuels and high pressure pneumatics be dumped and the SV has been so designed. LMSC understands that the current STS design does not have the provision to accommodate this dump.

SV has ability, STS does not.

2.5 SV/STS OPERATIONS

Sections 2.3 and 2.4 presented a brief description of the selected SV and its interface compatibility with the STS. This section describes the operations associated with the SV from manufacturing through launch, retrieval, and refurbishment.

2.5.1 Manufacturing, Assembly, and Test

Adaptation of the Hexagon SV to the STS mode of operation does not significantly change the current assembly and test concept nor does it require the construction of new facilities. The SVs for the STS flight program (starting in 1982) will be assembled in LMSC Bldg 156, the same as present. Perkin-Elmer will assemble and test the camera system at their Danbury, Conn. facility, then ship it to LMSC for SV installation. Since the camera system will be modularized there is no requirement to ship a portion of the flight SV to Danbury for camera system installation, which is the procedure on the current program. Environmental and optical tests, including acoustic temperature/vacuum and vacuum collimation, will be conducted, and a flight-ready SV will be shipped to WTR. Testing will utilize the SV RF link for computer commanding and data analysis.

Lange got their work back

A-2 attach points.

The required facility modifications are due to the larger cross-section of the SV and new attach points for STS mounting. The STS mounts also serve as SV handling points. The Vertical Integration Stand requires platform revision, the A-1 thermal/vacuum chamber, and the A2 collimation chamber monorails must be extended to accommodate the new handling points, and the A-2 collimation chamber SV support structure must be modified.

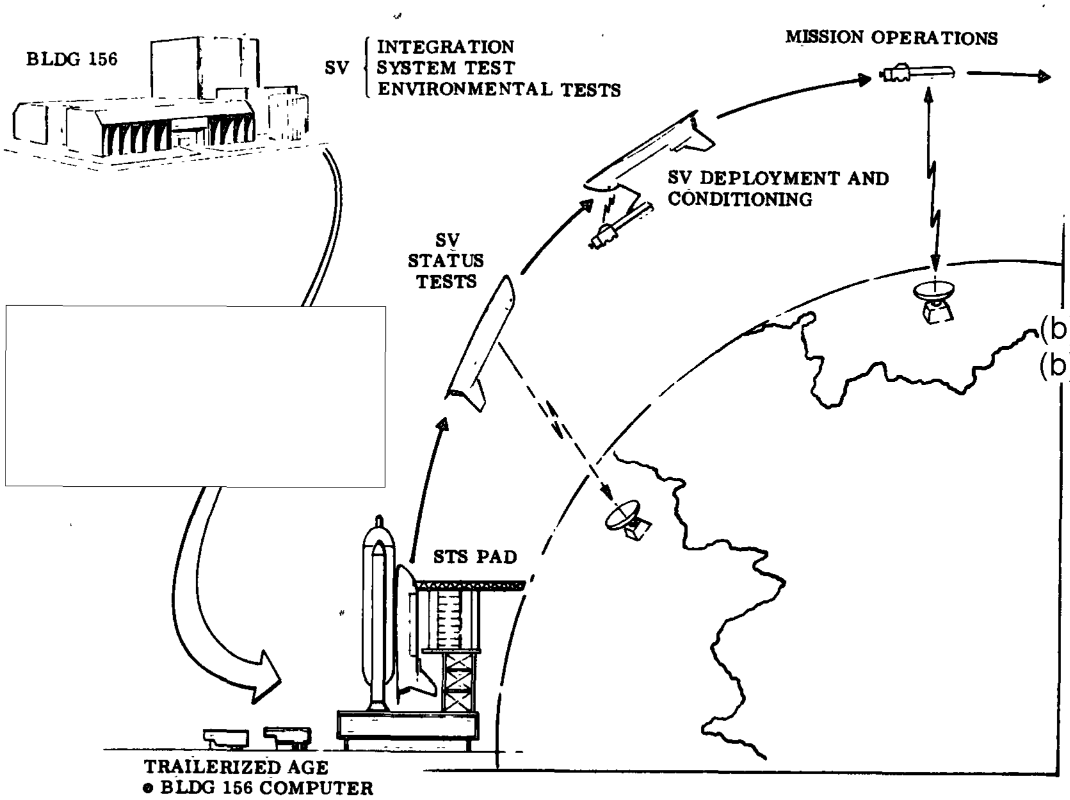
2.5.2 Pre-Launch Operations

The SVs will be transported to WTR by a [redacted] The STS (b)(1) (b)(3) 10 USC 424 gram launch complex is assumed to include a service tower/gantry with provisions for testing and pre-flight preparations of the SV in a clean room prior to arrival of the STS at the pad. SV/STS mating and final checks can be accomplished within the allocated STS on-pad time span. All SV testing and servicing will be by Hexagon program AGE of existing design, trailerized so that it may be removed from the pad area and stored between launches. Control over all pre-launch testing and monitoring of the SV ~~will be by computers located at LMSC Bldg. 156 and connected by a~~ data link to WTR. Once the SV is in the closed STS payload bay, monitoring of the SV will be through the data relay links located in the STS and through the STS skin umbilical. The STS has ultimate control of the SV at all times when the SV is mated to the STS. Safety only SV status will be monitored before launch by the LMSC computer and STS, and by the STS and the SCF network after launch.

2.5.3 Launch Operations

During a normal launch the STS will not be required to exercise any command control over the SV until orbit injection. In the event of an abort the STS will initiate a preloaded SV abort sequence, automatically commanding SV safing functions such as propellant dumping and pneumatic depressurization. Status of these abort activities will be displayed to the STS crew.

Good



(b)(1)
(b)(3) 10 USC 424

Fig. 2-7 SV Prelaunch and Launch Operations

2.5.4 Orbital Operations

The orbital operation concepts developed by LMSC require that, once the Hexagon SV/STS program is operational, each STS launch will accomplish two objectives: (1) deployment of a refurbished SV and (2) retrieval of an expended SV. The purpose of this is to reduce program launch cost. The actual operational sequence employed to achieve these objectives was derived by considering a key driver, the STS method of payload weight accounting.

The STS payload weight capability (36,400 lb at 96.4 deg inclination) is based upon injection into a 50 x 100 nm orbit (not a very useful orbit). All propellants used by the STS to adjust to a higher orbit are charged to payload weight. The payload weight penalties associated with this are rather severe. The STS with payload requires 23 lb we lose propellant (life time) by boosting ourselves into a higher orbit (operational orbit)

what?

of propellant to achieve 1.0 ft/sec change in orbital velocity. Because of its much smaller mass, the SV can achieve the same velocity change for about one-fifth the propellant weight. Accordingly, LMSC has adopted an operational concept which minimizes STS use of OMS propellant. As shown in Fig. 2-8, the SV will accomplish all orbit adjust maneuvers except the critical orbit adjust from 50 x 100 nm to 80 x 100 nm. The latter is deemed the minimum acceptable altitude for reliable deployment or retrieval of an SV. The SV life in this orbit is 8 revs tumbling and 32 revs normal operation without orbit adjust.

~~Bad!~~ ~~BS!~~ Bird!

~~The STS is a transport lets use it as such.~~

I understand this now

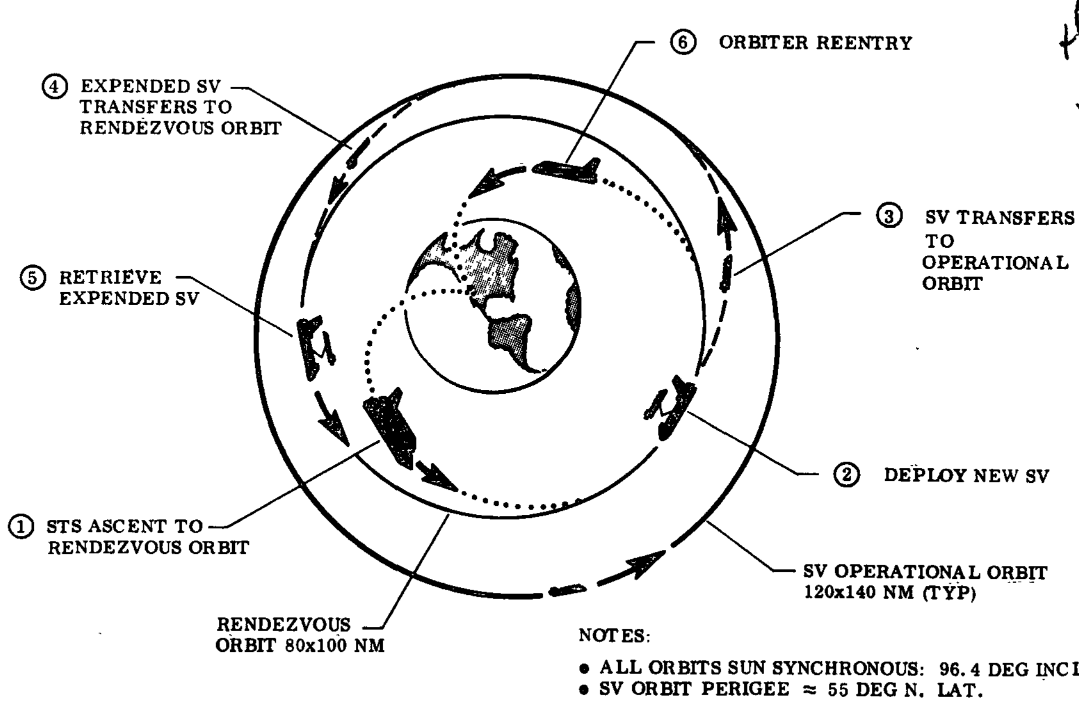


Fig. 2-8 SV/STS Flight Operation

The 16.7 lb/day of propellant available for orbit maintenance, as shown in Fig. 2-6 SV operational orbits, was derived by using the above described operational concept. Total propellant and gas allocations are as shown in Table 2-1.

Prior to SV deployment an SV health and status test will be accomplished to verify readiness for deployment. During these pre-deployment tests the SV/STS will operate autonomously from the SCF, in that all SV functions will be under direct control of the STS crew. This approach was taken to maximize operational flexibility and crew safety due to the relatively infrequent and brief tracking station contacts.

During normal operations, when the SV is in or near the STS, the SCF will operate primarily in a backup mode. If SV problems occur, the SCF can analyze SV real-time or recorder playback data and recommend corrective action.

Table 2-1

SV PROPELLANT AND GAS ALLOCATIONS

Total weight available for propellant and gases	7,407 lb
Less propellant/pneumatic weight required for:	
(1) SV transfer from STS to SV operational orbit	800
(2) SV return to STS orbit for retrieval	365
(3) SV orbit phasing corrections for rendezvous	125
(4) RCS, Lifeboat, and pressurants	828
(5) Camera system pneumatics	175
(6) Deboost	600
Weight available for orbit maintenance propellants	4,514
Max propellant usage per day (270 days)	16.7 lb/day

$\boxed{87 \times 150}$

270
27 lb/day
1890
540
7290

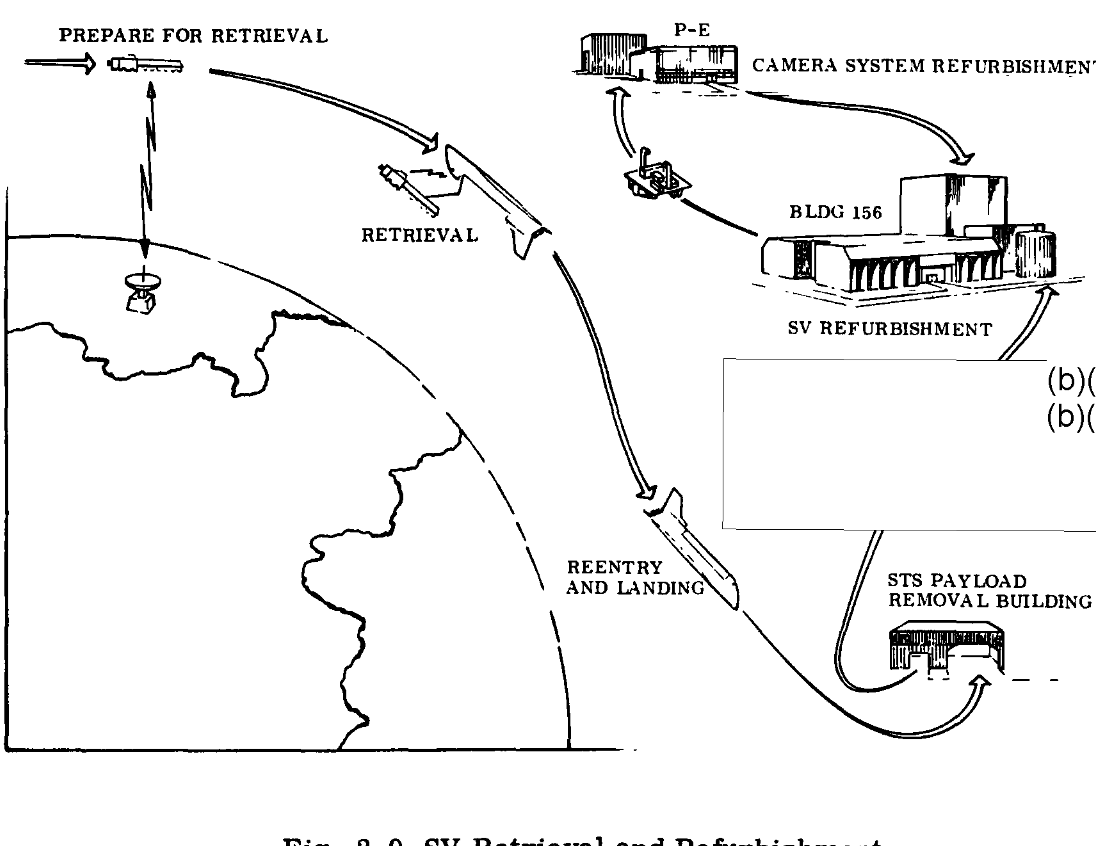
16.7
270
11690
334
4.50.90

ok

has to be in high orbit.

2.5.5 SV Retrieval

For retrieval the SV will be transferred to the STS rendezvous orbit under SCF control and commanded into retrieval status by the SCF (Fig. 2-9). Final rendezvous will be effected by the STS, using its reaction control system (RCS) for required velocity and positional changes. All payloads and subsystems not required for the retrieval operation will be shut down. The STS will control the SV during retrieval and will monitor the SV status. Propellants and pneumatics in excess of those required for SV deboost will be dumped. If retrieval is not accomplished the SV could then be deboosted into a deep ocean area. After retrieval and storage in the STS payload bay, electrical and fuel umbilicals will be mated. Remaining propellants will then be dumped through the STS system prior to STS reentry.



(b)(1)
(b)(3) 10 USC 424

Fig. 2-9 SV Retrieval and Refurbishment

2.5.6 Post-Landing Operations

The ground air conditioning must be connected as soon as possible after landing to prevent SV equipment temperatures from exceeding specification values. After Orbiter deservicing, the Orbiter will be towed to the maintenance and checkout facility where the SV will be removed and prepared for shipment to the factory for refurbishment. These preparations include draining, flushing, and purging of the SV propellant system as a first stage in the refurbishment cycle.

2.5.7 SV Refurbishment

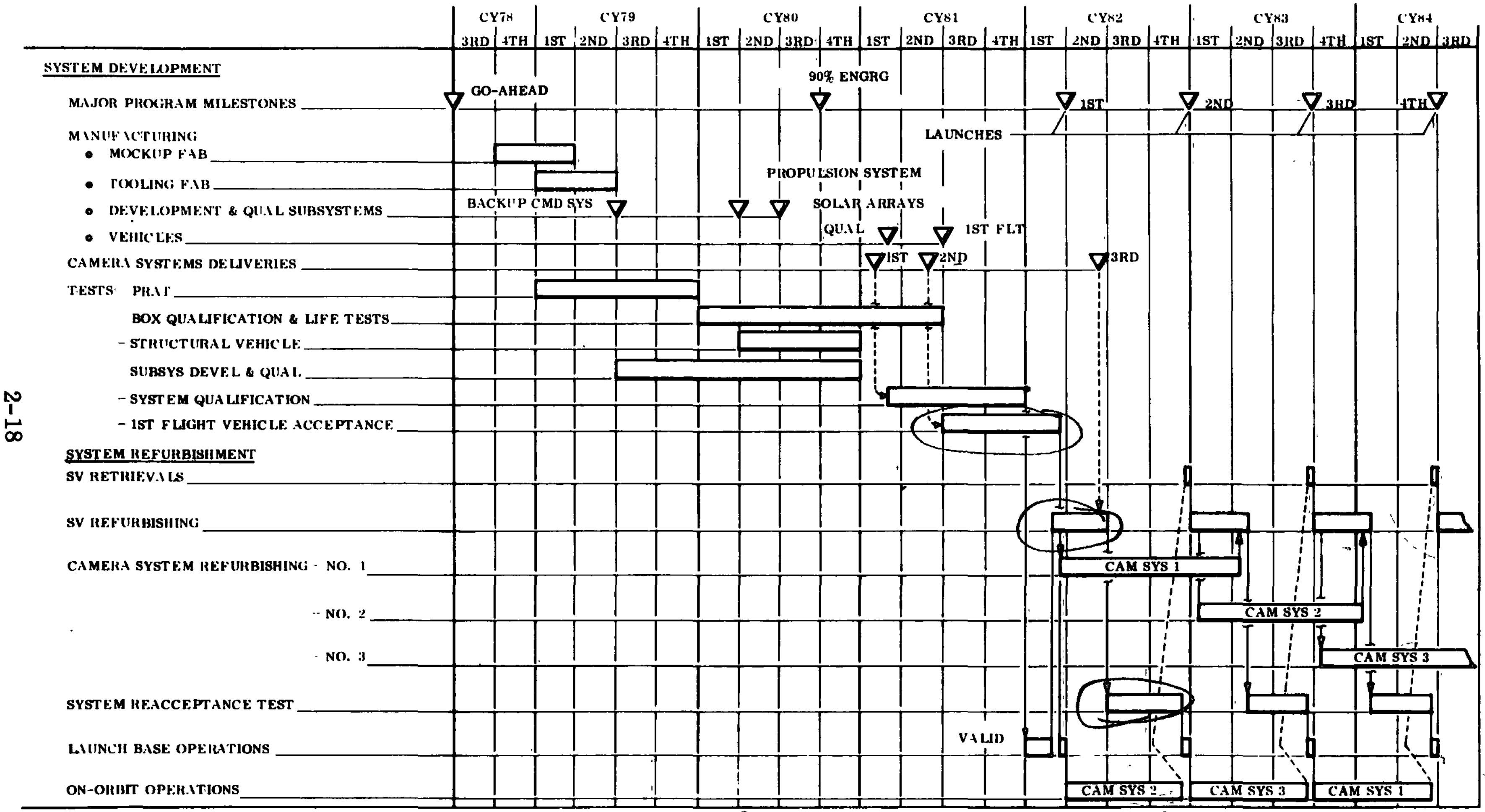
Refurbishment operations at the factory will commence with the camera system removal and shipment to the P-E facility for refurbishment and retest. A test on the SV will then be performed to determine the status of all primary and redundant systems. All malfunctioned equipment on the SV will be replaced and new RVs installed. Time spans dictated that the camera system cannot be refurbished and returned to LMSC in time for reinstallation into the same SV from which it was removed. Three camera systems will therefore be required to support a two-SV squadron. Once the camera system is installed and the supply unit reloaded, the SV will go through a complete functional testing program and recycled to WTR for reflight. In the event that certain key SV systems require replacing, LMSC may elect to accomplish a thermal/vacuum reverification test on the vehicle before shipment to WTR.

why

2.6 SCHEDULE

LMSC has developed a program plan for designing, fabricating and qualifying the 6-RV configuration for an STS launch in the first quarter of 1982. Figure 2-10 shows a 45-month span between LMSC Fiscal Year 1979 go-ahead and launch. P-E requires an earlier authority to proceed – Fiscal Year 1977. based on opt. release

Two flight-type SVs will be fabricated. The first SV will receive qualification testing and then be refurbished and flown as the second flight vehicle. nice



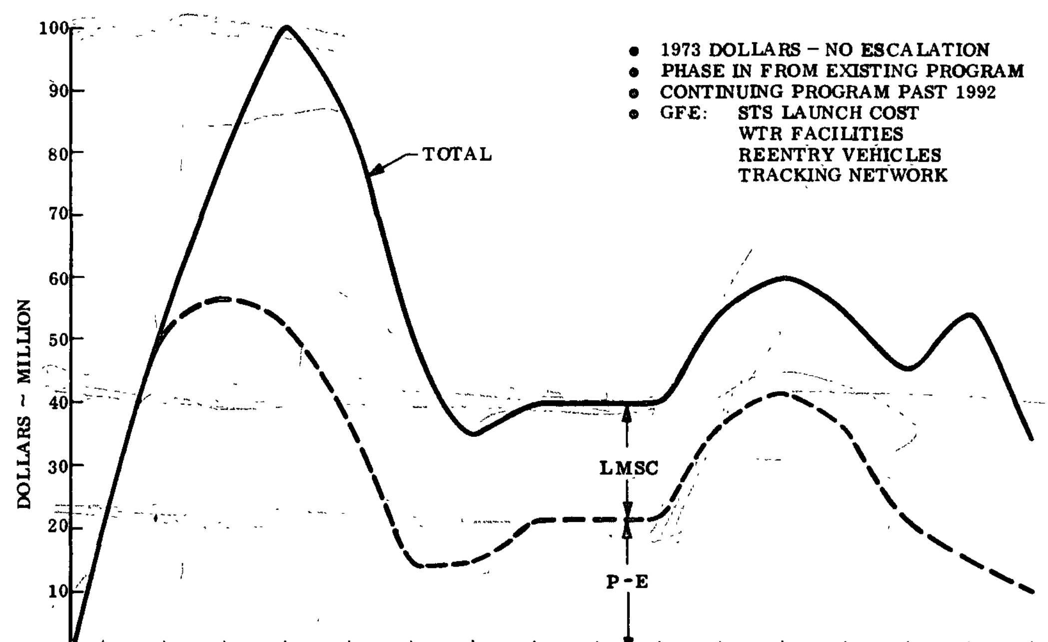
maybe to short.

Fig. 2-10 Program Schedule

BIF003W/2-069331-73

2.7 COST SUMMARY

The program costs shown in Fig. 2-11 include LMSC and P-E cost, time-phased by fiscal year, to reflect total program cost. For pricing purposes, 1973 dollars were used and provisions for economic escalation during future years were not included. A phase-in from the existing program has been assumed. Additionally, a continuing program has been forecast past 1992. It was assumed that STS launch, WTR facilities, recovery vehicles, and tracking network will be furnished GFE to this program.



LMSC			23.0	46.0	48.0	37.0	20.0	18.0	18.0	18.0	18.0	18.0	18.0	24.0	38.0	24.0	$368.0M
P-E	20.0	51.0	57.0	53.0	36.0	14.0	15.0	22.0	22.0	22.0	37.0	42.0	36.0	21.0	16.0	10.0	$474.0M
TOTAL	20.0	51.0	80.0	99.0	84.0	51.0	35.0	40.0	40.0	40.0	55.0	60.0	54.0	45.0	54.0	34.0	$842.0M
	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91	92	TOTAL

FISCAL YEARS

large optics looks like a sales job.

Fig. 2-11 Program Cost Summary

The LMSC costs were computed by using Hexagon program cost history and profiles. Block I costs were adjusted to 1973 rates for major development items. Block III costs were used for existing or modified hardware. Two blocks of vehicles were quoted as non-recurring costs; Block A for operation in 1982-1992, and Block B for 1992 and beyond. Each block included two SVs and one complete set of spares. Recurring costs begin at the completion of the SV/STS qualification program.

2.8 SUGGESTED AREAS FOR ADDITIONAL STUDY

Future studies should consider the period during which the SV is transitioning from an SLV-launched to an STS-launched program. Methodology, design approach, backup concepts, and logistics should be part of the study.

Firm and detailed SV/STS interface requirements should also be generated. Although STS development paces the development of any SV configuration for STS, detailed Hexagon SV interface requirements will facilitate early STS design guidance.

Testing of equipment and materials should begin, to verify their capability to survive the orbital lifetime requirement of the STS mode of operation.

The SV reuse and lifetime requirements should also optimally accommodate foreseeable changes in technology and mission requirement. Such changes could affect SV useful calendar lifetime, and at some point in time make it economically advantageous to build new SVs rather than update existing configurations. Methodology should be established for determining useful life span of a block of SVs.

Section 3
BASELINES AND GROUND RULES

This section provides a detailed description of the Block III Hexagon Satellite Vehicle, which is the point of design departure, and a description of the reference STS. The section concludes with a statement of criteria and ground rules applied in this SV/STS study.

3.1 BLOCK III HEXAGON SATELLITE VEHICLE

The SLV-launched Hexagon SV performs a photographic search and surveillance mission over specified global areas, with data retrieval accomplishment by multiple recovery vehicles. The SV can also conduct a mapping, charting, and geodesy mission through use of a stellar index (SI) terrain camera. The SV is also capable of carrying two subsatellites into orbit and ejecting them on command. Capability also has been reserved for vulnerability reduction devices.

Operations of the primary and SI cameras are independently programmed to satisfy user community requirements. The orbit is maintained throughout the mission by the orbit adjust capability of the SV. After ejection of the RVs, the SV is deorbited into a desired ocean area. Redundancy is provided in the SV to ensure that no single failure will abort the primary mission. A reliability of 0.85 after 60 days is the design goal for the Block III SV.

The following paragraphs provide a description of the SV and its subsystems.

3.1.1 SV General Configuration

The overall length of the SV is approximately 56 ft. At launch the length is 59 ft with the shroud and the Titan IIID booster adapter. The SV consists of three major structural sections:

- Aft or Satellite Control Section (SCS), which contains the SV control and support provisions
- Mid Section, which houses the stereo reconnaissance cameras, their film supply, and support equipment
- Forward Section, which accommodates the four RVs used for return of the exposed film.

An auxiliary payload, the Mapping Camera Module (MCM), contains the stellar index camera and one Mark 5 Satellite Recovery Vehicle (SRV) for retrieval of the exposed film. The main camera system is provided by Perkin-Elmer, the four RVs by Mc Donnell/Douglas, the SI by Itek Corp., the doppler beacon system by Applied Physics Laboratory, and the SRV by General Electric, Reentry Systems Division. All other SV structure and equipment, except for the GFE command system and piggy-back payloads, are provided by LMSC. Integration and system test of the SV is accomplished by LMSC.

In the LMSC factory the SV is brought to flight readiness by acoustic, thermal/ vacuum, and vacuum collimation testing of the assembled vehicle. Subsystems are designed for system-level testing through RF command and data links. The SV is shipped, essentially flight-ready, to the launch base for validation prior to launch.

The SV configuration permits modifications to meet specific mission requirements. The MCM and subsatellites can be omitted, and propellants and RVs can be off-loaded at WTR. Table 3-1 presents the weight breakdown for the SV. It is important to note that approximately half of the vehicle weight is in expendables (fuel, film, RVs, etc.). This factor was found to be important in the operational concept study (resupply/ maintenance vs. refurbishment-only). Most studies to data on resupply/maintenance have concentrated on vehicles that operate at higher altitudes and have a much smaller portion of their total weight allocated to expendables.

Table 3-1

BLOCK III HEXAGON SV WEIGHTS

		Percent Launch Weight	Percent Launch Weight
Fixed Equipment			
Forward Section	1, 078 lb		
Mid Section	1, 433		
Aft Section (W/Adapter)	3, 627		
Shroud	2, 912		
Subtotal SBA* @ Launch	9, 050		
Subtotal SBA @ Injection	6, 206		
Camera System (Less Film, Gas & TUs)	4, 344		
RV Permanent Provisions	240		
Total Fixed @ Launch	13, 634	57	
Total Fixed @ Injection	10, 790 lb		51
Expendables			
RVs (Less Fixed Portion) Plus TUs	4, 540 lb		
Camera System Film and Gas	1, 825		
SV Propellant and Gases (60 Days @ 82 x 144)	3, 890		
Total Expendables	10, 255	43	49
Total SV @ Launch	23, 889		
Total SV @ Injection	21, 045 lb		

* SBA is the LMSC-provided hardware

3.1.2 Payloads

3.1.2.1 Camera System. The camera system provides high-resolution stereoscopic coverage of selected areas of the earth's surface by using two independently-controllable panoramic cameras. The system is capable of providing target resolution of 139 lines per millimeter (2.27 ft) or better at scan nadir when operating at primary mission orbital altitudes, with a target contrast of 2 to 1 at the entrance pupil and a 30 deg sun angle. Design characteristics include:

Optics	60 in. focal length, f/3 camera
Film	6.6 in.-wide ultra thin base film
Film load	104,000 ft per camera
Scan modes	30, 60, 90, and 120 deg at full scan
Frame format	(120 deg scan) 6 in. x 125 in.
V/h range	0.018 to 0.054 rad/sec

General arrangement of the camera system, which is installed in the SV, is illustrated in Fig. 2-2.

3.1.2.2 Recovery Vehicle (RV). The four large RVs provide a takeup capability for the camera system. Each RV provides for the return of 450 lb of exposed film from the orbiting SV to the earth's surface at a given time and location. Film is initially routed through each RV and taken up on the forwardmost RV. As the RVs are filled with film, film is cut at the RV entrance, the capsule sealed, the film taken up on the next RV in line, and that RV's exit film paths are sealed. When RV retrieval is scheduled, the SV is put into the optimum pitch orientation, and at the correct time the RV separation is commanded. The reentry sequence consists of spin-up, retro rocket burn, de-spin, propulsion assembly ejection, ballistic reentry into the atmosphere, parachute deployment, and heat shield separation at the design altitude. The RV protects the takeup assembly and the film load during the reentry/recovery sequence. The RV flight is normally completed by aerial recovery of the payload capsule, which is suspended on the RV main parachute. Surface (water) recovery capability and a delayed sink capability are also provided. After recovery, the takeup assemblies are returned to P-E for refurbishment and reflight.

3.1.2.3 Mapping Camera Module (MCM). The MCM, which is not part of the SV/STS configuration, is installed on the front of the SV Forward Section. The MCM is an integrated photographic payload, complete with film retrieval capabililty. The module includes the SI mapping camera system, two film supplies, and a dual takeup located in the SRV, interconnecting film paths, and a doppler beacon system (DBS) for accurate ephemeris determination. Functional and environmental testing of the MCM can be conducted on or off the SV.

3.1.3 SV Structure

The SV structure has been designed to support all equipment and launch loads in a cantilevered fashion off the front end of a Titan IIID. For ease of manufacturing, assembly, and test, it is divided into three main sections: aft, mid, and forward. During launch the structure is enclosed by an ejectable shroud and mated to the Titan by a booster adapter which contains a separation joint. Neither of these latter two items are required for the STS-compatible SV. Materials used for fabrication of the SV include titanium, aluminum, and magnesium.

3.1.4 Aft (Satellite Control) Section

Figure 3-1 shows the Aft Section arrangement. The structure provides equipment housing, environmental protection, mounting interfaces, load support and distribution, and booster interface and separation mechanisms. The electronic equipment located in the SCS is grouped functionally on separate removable honeycomb panels designated as modules. The following subsystems are principally installed on modules in the Aft Section (portions of each subsystem may be elsewhere on the SV as dictated by functional considerations).

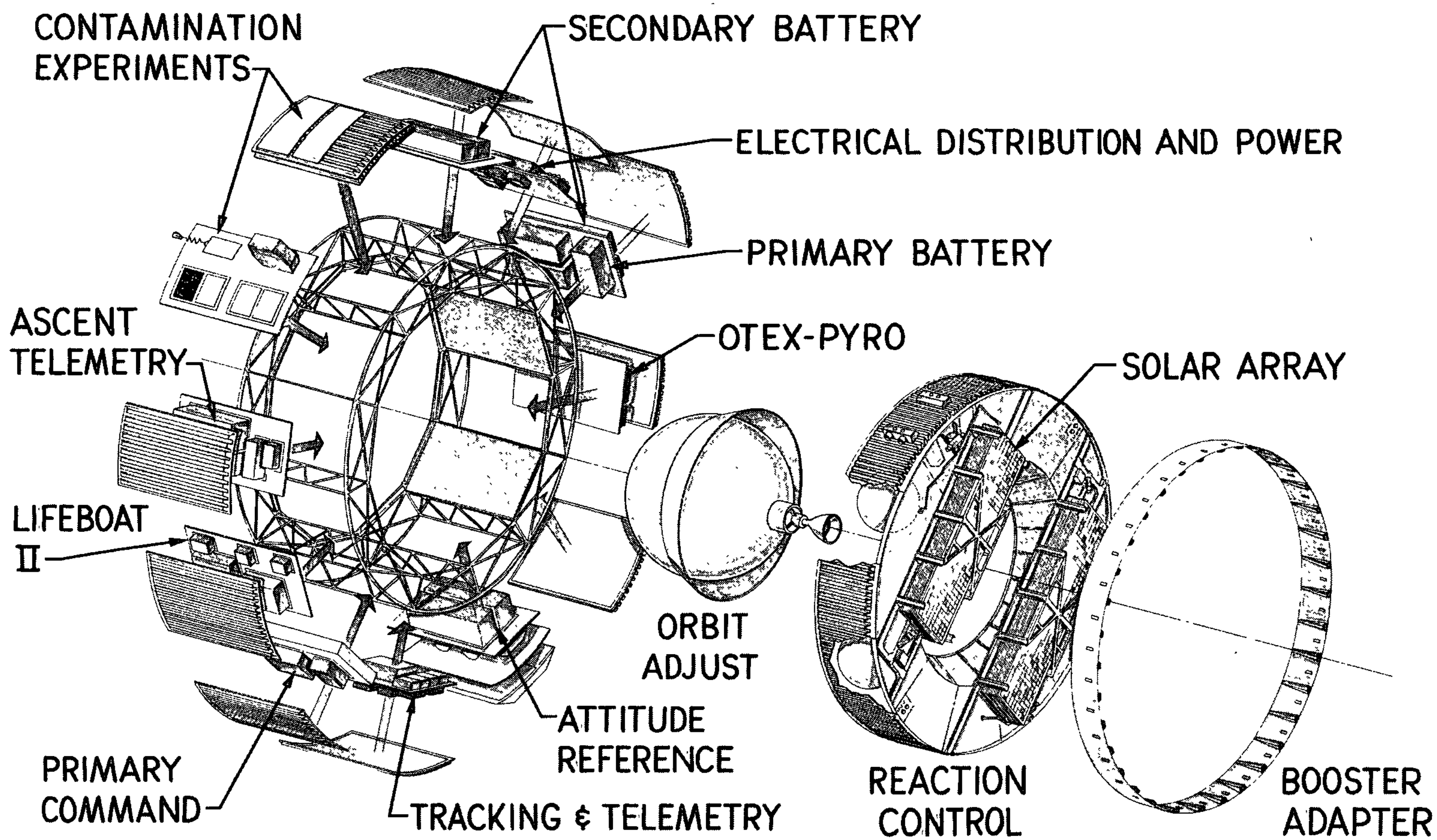


Fig. 3-1 Satellite Control Section

BIF003W/2-069331-73

~~TOP SECRET/H~~/10116

Electrical Distribution and Power (EDAP). Power to operate the SV is provided by solar arrays deployed from the Aft Section. Rechargeable batteries provide energy storage to meet dark-side-of-earth power requirements. Unregulated power within a 24 to 33 VDC range is distributed throughout the vehicle to using equipment. The power system is capable of providing 11, 000 watt-hours per day of usable power. Capability to operate with a solar beta angle ranging from -60 to +60 deg is attained by adjusting the solar array position about the vehicle roll axis. Power for the Lifeboat emergency system is provided by a primary battery. Equipment necessary for RV and SV deorbit can be switched to this battery for emergency operations. Pyro power is provided by redundant, isolated primary batteries and distribution circuits, with backup power provided by two of the four main batteries.

Attitude Control System (ACS). The ACS provides earth-oriented attitude reference and rate sensing. It develops reaction control system (RCS) firing signals to bring the SV to a commanded attitude and to maintain attitude and rate within the required accuracies. The ACS also provides measurements of vehicle attitude and rate during camera system operations. The reference element is a 3-axis rate gyro integrator system, with horizon sensor updating in pitch and roll, and gyrocompassing in yaw. Thruster firing signals are generated by the sensors, combined with the flight control electronics, and modulated in each axis to provide the impulse bit control necessary to meet the rate control and settling time requirements.

Orbit Adjust System (OAS) and Reaction Control System (RCS). The OAS and RCS provide the forces necessary to control the vehicle orbit and the vehicle attitude in orbit. Orbit control comprises injection error correction, drag and perigee rotation makeup, and deorbit of the SV at the end of the mission.

The OAS and RCS use catalytic decomposition of monopropellant hydrazine to generate thrust. The systems are pressure-fed with the pressurizing gas enclosed in the propellant tank with the hydrazine. This results in a declining ("blowdown") pressure characteristic such that the thrust level of the orbit adjust engine declines from 280 to 125 lb and the reaction control engine from 5 to 2 lb.

Telemetry and Tracking. The SGLS-compatible telemetry system provides PCM realtime data (ascent) at 48 KBPS, engineering analysis (realtime) at 128 KBPS, orbit realtime at 64 KBPS, and PCM tape recorded data (48 KBPS played back at 256 KBPS). The PCM telemetry provides status for normal mission operations, decision making, test operations and evaluation, command acceptance confirmation, and post-flight evaluation. Each tape recorder provides a minimum of 60 minutes of continuous recording of data, including the storage of payload status information and SV operational, attitude, and rate data.

Command. The GFE command system provides realtime and stored program primary and emergency capability. The SGLS-compatible primary (extended) command system (ECS) includes a dual remote decoder and complete redundancy. It provides operation commands to perform primary and secondary missions, the capability to configure the vehicle into various modes, test and checkout, security provisions for critical functions, vehicle system time to PCM and payload, and protected stored program commands to service certain SV functions. Included with the primary system is a backup 375 MHz receiver-demodulator to receive and provide commands to the ECS. The emergency (minimal) command system (MCS), with a single remote decoder, receives commands from another 375 MHz receiver-demodulator. The emergency system provides an independent capability to control the Lifeboat functions.

Lifeboat System. The Lifeboat system provides an emergency capability to initiate the recovery of two RVs and to deorbit the SV in the event of a complete failure of the main power system, the ACS, or the ECS. Emergency operation control is provided by the 375 MHz receiver, the MCS, and the SGLS-compatible telemetry system. SV attitude control, required to accomplish RV separations and SV deorbit, is provided by earth field sensing magnetometers, rate gyros, and a cold gas control system. Power to keep the system in a ready state for use and for emergency operation is provided by a primary battery. The OAS engine, and other necessary equipment for RV drop and SV deorbit, are switched from the main power system to the Lifeboat bus for emergency operations.

3.1.5 Computer-Controlled Test Concept

The program philosophy is to test the SV in essentially the orbital configuration during ambient tests and environmental exposure. This results in the use of an RF interface rather than extensive hardline interfaces during SV factory and pad testing.

The SV incorporates test commanding and monitoring capability primarily in the TT&C subsystem, resulting in a simple RF interface between the SV and the computer. That is, the vehicle command system is used to exercise all vehicle functions during test, and the PCM telemetry is used to acquire all test data. This arrangement ensures maximum flexibility of a computer-controlled test and reduced complexity of the AGE required to provide umbilical control and monitoring, and duplicates on-orbit operation as closely as possible.

Factory and pad testing, through the RF links, uses the same computer and AGE hardware designs, thus allowing identical test software to be used in all factory test locations and the launch pad.

3.2 DESCRIPTION OF STS

The STS operating characteristics and its payload accommodation provisions employed in the study were derived from three principal documents and supplemented by technical discussions with the Customer and Aerospace Corporation. The three documents are:

(1) "Baseline Space Shuttle System Description", Aerospace Corp. Report No. TOR-0073(3421-03)-1, 2 Jan 1973

(2) "Space Shuttle System Payload Accommodations," NASA Document SSCO7700, Vol. XIV, 13 April 1973

(3) Rockwell International Report "Space Shuttle System Technical Review", 16 April 1973

The LMSC-derived description of the reference STS configuration was then presented at the mid term briefing for Customer and Aerospace review and approval. This section presents that reference STS description, but it should be noted that the STS design is still being evolved.

Figure 3-2 shows the reference STS. The location of the Remote Manipulator System (RMS) is of particular importance since placement on the STS has changed several times since the start of detailed STS design.

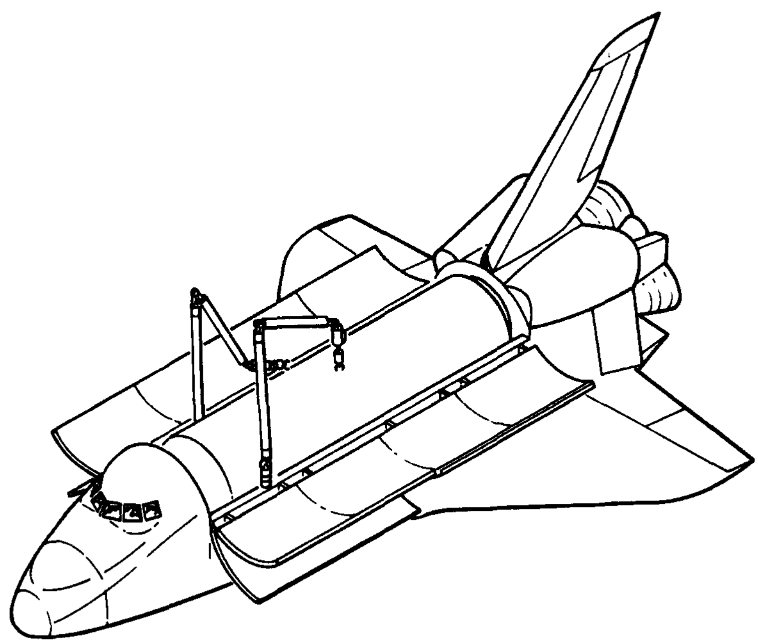


Fig. 3-2 Reference STS

Payload Weight. The payload weight carrying capability of the STS is 60,000 lb maximum as stated by NASA. This represents an easterly launch out of Kennedy Space Center. For the Hexagon program all launches will be from WTR and into a sun-synchronous orbit at 96.4 deg inclination. This case provides a payload weight capability of 36,400 lb into a 50 x 100 nm orbit. However, this is not all useful payload weight because an accounting system has been established that charges certain mission-required equipment/expendables to either STS or payload weight as follows:

Included within STS weight allocations:

- 250 ft/sec orbital maneuvering system (OMS) for reentry
- 3900 lb RCS propellant
- Rendezvous radar

- One remote manipulator arm
- Orbiter half of payload mounts and service panels

Items chargeable to payload weight allowance:

- OMS propellant for orbit adjustment from 50 x 100 nm orbit at 23 lb/ft/sec
- RCS propellants for rendezvous at 1800 lb/rendexvous
- Second remote manipulator arm at 600 lb
- Payload half of mounts, fluid, and electrical connections
- Any equipment required for EVA at 200 lb/person
- Docking collar at 2700 lb max
- Payload monitor equipment in Mission Specialist panel

A landing weight constraint also exists: 25,000 lb maximum payload weight for normal landing. This can be exceeded in emergency situations with the risk of operating at reduced STS margins.

Payload Mounting Provisions. The payload bay is rated as being capable of accommodating a 15 ft diameter by 60 ft long payload. This dimension must, however, include any payload dynamic excursions during launch, reentry, and landing. Accordingly, LMSC established a 174-in. maximum diameter payload envelope, which allows 3 in. clearance for dynamics. Mounting of payloads within the STS is accomplished by a family of standardized mounts. Figure 3-3 shows the type of mounts available, their locations, and their utilization to ensure that the payloads are mounted in a statically determinate fashion. Maximum load carrying capacity of any single mount is:

Side Rail Mounts:

250,000 lb ±X
160,000 lb -Z
75.000 lb +Z

Keel Mounts:

110,000 lb ±Y

All loads are provided as ultimate, and a 1.4 factor was employed to convert to limit loads.

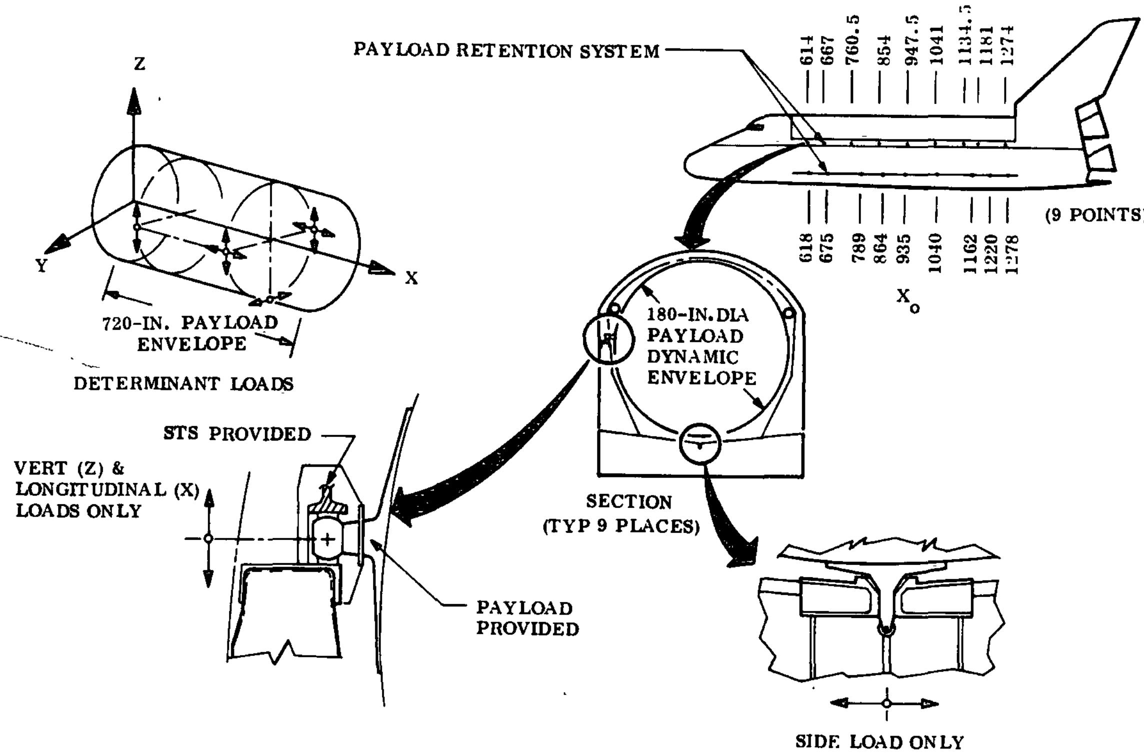


Fig. 3-3 Payload Envelope and Mounting

Payload Handling. Payload handling aboard the STS is to be by the Remote Manipulator System (RMS). The RMS operator has only limited direct visual access to the payload when in the bay or out of the bay above the STS (Fig. 3-4). Remote-controlled TV cameras and lights are provided for operator assistance: two sets are mounted in the payload bay and one camera and light are on each RMS.

The maximum reach of the RMS is 47.5 ft (Fig. 3-5), and it has a positional accuracy of ±2.0 in. and orientation accuracy of ±0.1 deg. Due to its length, its tip force normal to RMS centerline is limited to 10 lb. The length is not adequate to allow complete access to all payload areas. For example, if an SV is docked normal (parallel to Z axis) to the STS at the front of the payload bay, the RMS has access to less than 40 ft of SV length. This study considered SVs up to 60 ft in length.

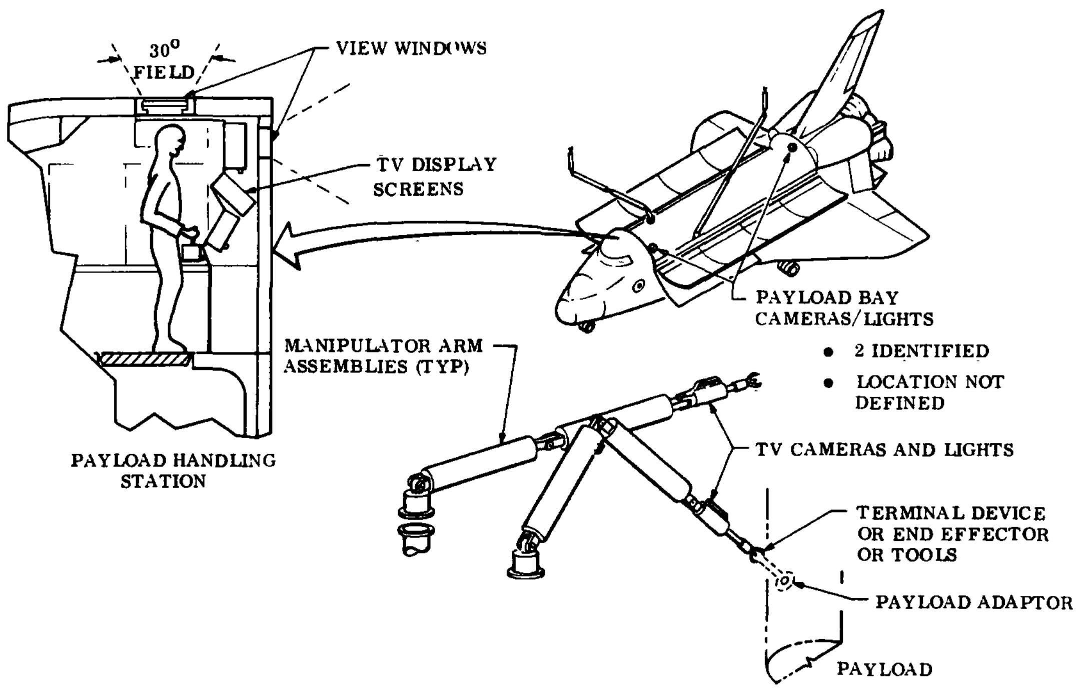


Fig. 3-4 Payload Visual Observation

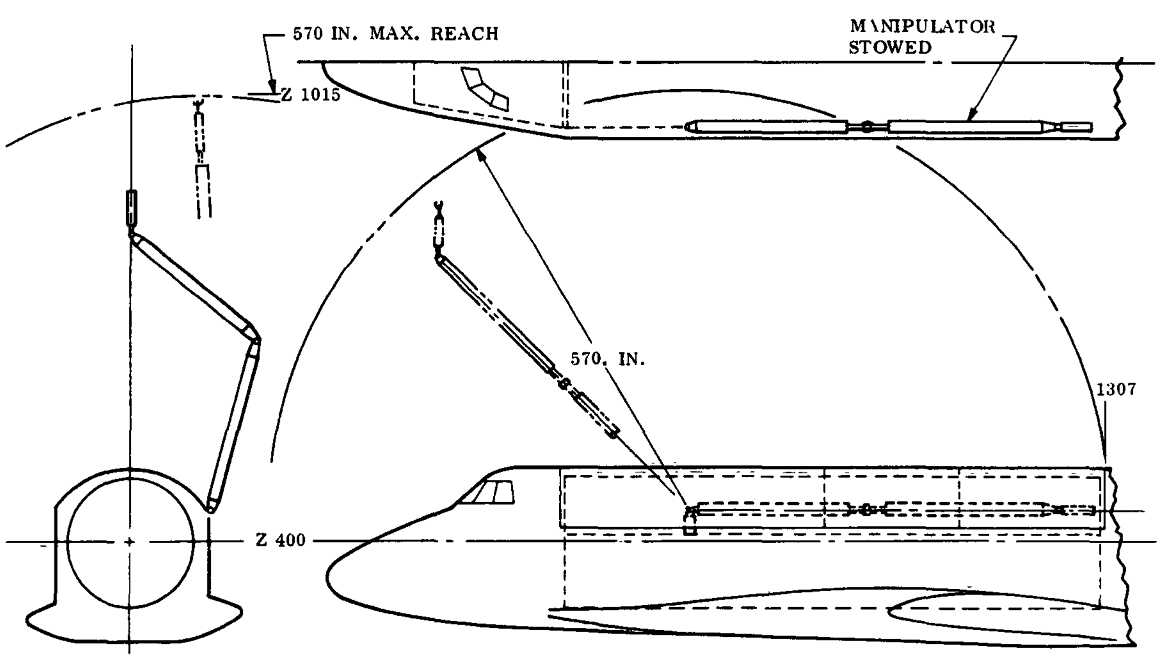


Fig. 3-5 Manipulator System Characteristics

Functional Interfaces. Any payload installed in the STS will have a variety of functional interfaces with the STS: power, command and control, monitoring, coolant, propellants, and pressurants. Figure 3-6 shows a simplified version of the power, command and control, and monitoring for SV/STS interfaces. Certain equipment required by a payload for display or recording of data at the Mission Specialist console must be payload-provided and compatible with other STS equipment. Not shown but also available is the provision for direct access to the payload from GSE through an STS skin-mounted umbilical.

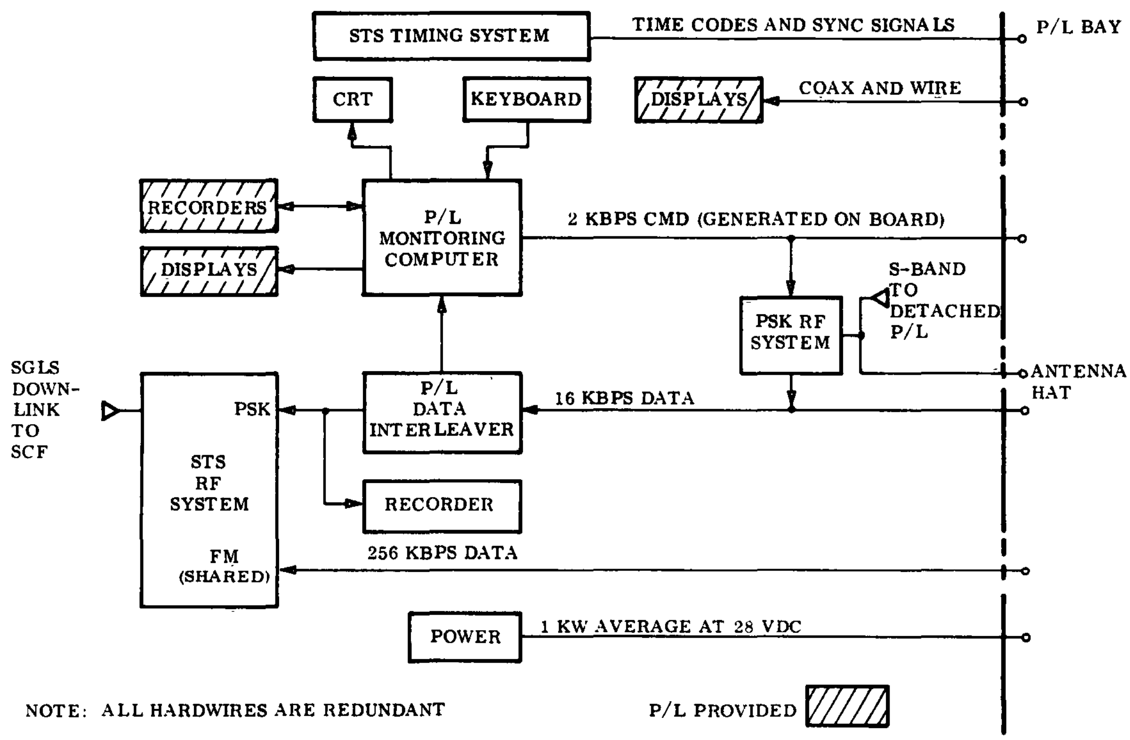


Fig. 3-6 SV/STS EDAP and TT&C Interfaces

Provisions for propellants and pressurant system umbilicals exist between the payload and GSE only through the STS preflight or launch umbilicals. There are no present provisions for connecting payload systems to STS flight-type plumbing for purposes of dumping payload storable propellants or pressurants. The only dump capability is for cyrogenics from upper stages.

Payload C.G. Constraints. Figures 3-7 and 3-8 illustrate the Orbiter longitudinal and vertical payload c.g. envelope. Curves are applicable for both launch and reentry/landing cases.

Environments. The critical environments affecting SV design were established as part of the reference STS. Table 3-2 presents the interface limit loads expected at STS payload retention points. Figure 3-9 shows the payload bay acoustic spectra and random vibration levels. The internal payload bay adiabatic wall temperature limits are identified in Table 3-3. Payload bay air conditioning will be provided during pre-launch activities, with the payload installed in the bay.

Table 3-2

SV/STS INTERFACE LOAD FACTORS

CONDITION	X	Y	Z
Lift-Off *	-1.7±0.6	±0.3	+0.8 -0.2
High Q Boost	-1.9	±0.2	+0.2 -0.5
Booster End Burn	-3.0±0.3	±0.2	-0.4
Orbiter End Burn	-3.0±0.3	±0.2	-0.5
Space Operations	-0.2 +0.1	±0.1	±0.1
Entry	±0.25	±0.5	+3.0 -1.0
Subsonic Maneuvering	±0.25	±0.5	+2.5 -1.0
Landing and Braking	±1.5	±1.5	+2.5
Crash**	+9.0 -1.5	±1.5	+4.5 -2.0

* Includes Dynamic Transient

** Ultimate Load Factor

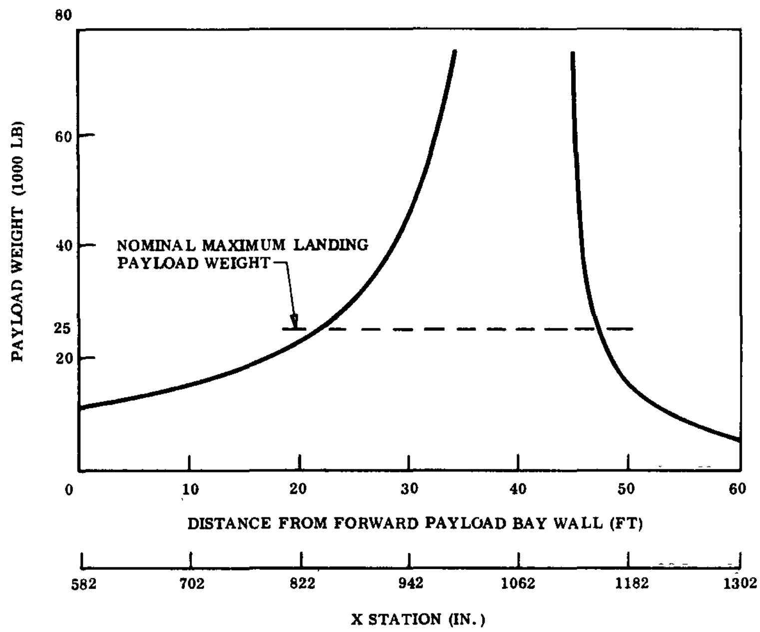


Fig. 3-7 Payload X C.G. Envelope

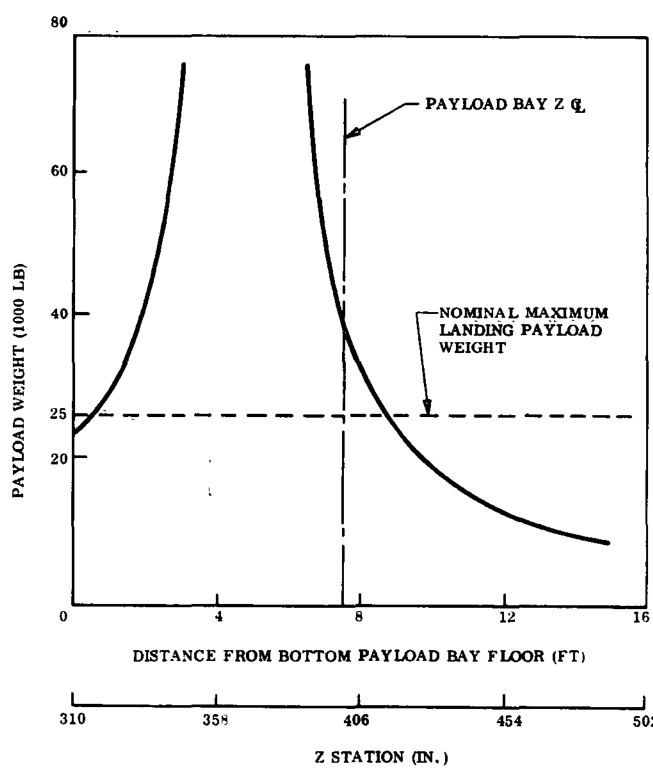


Fig. 3-8 Payload Z C.G. Envelope

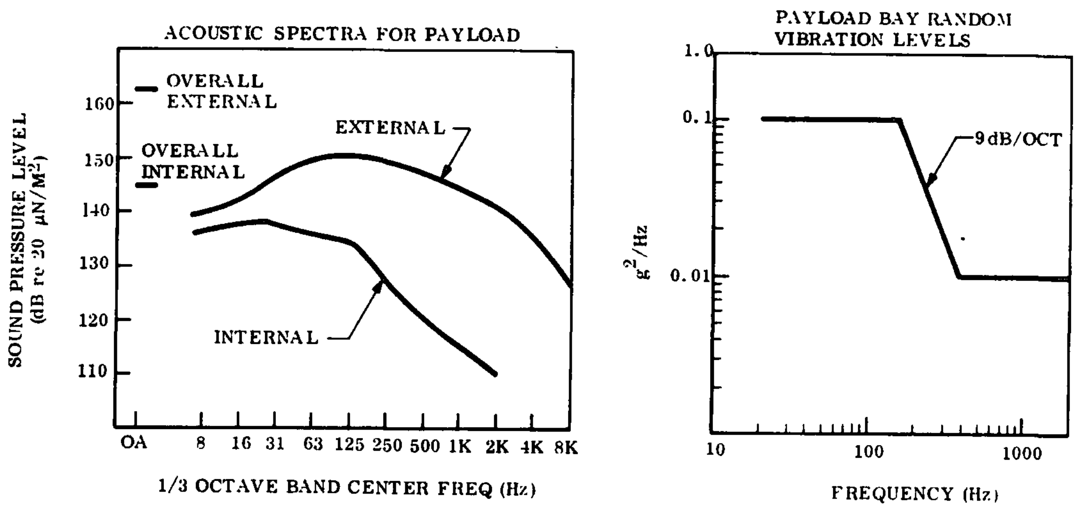


Fig. 3-9 Payload Bay Acoustic and Vibration Spectra

Table 3-3

PAYLOAD BAY WALL THERMAL ENVIRONMENT

Condition	Minimum	Maximum
Pre-Launch	+ 40°F	+120°F
Launch	+ 40°F	+150°F
On-Orbit (Doors Closed)	See C&D	See A&B
Entry and Post-Landing	-100°F	+200°F

A. Total Bay Heat Gain, Average ≤ 0 BTU/Ft²-hr

B. Heat Gain, Local Area ≤ 3 BTU/Ft²-hr

C. Total Bay Heat Loss, Average ≤ 3 BTU/Ft²-hr

D. Heat Loss, Local Area ≤ 4 BTU/Ft²-hr accept on-orbit use passive thermal control.

The on-orbit thermal control design used for sizing the nominal payload passive thermal control provided by the STS, is to size the payload bay to limit the heat leak over a finite area into or out of a 100°F constant temperature payload to 3 BTU/hr/ft^2, with the payload doors closed under worst-case orbital orientations. However, the payload bay doors are opened 30 minutes after orbit injection. The STS does not provide a thermal cover for protection of payloads during on-orbit operation. The payload will therefore be exposed to the space environment and must provide for its own passive and/or active thermal control. It is assumed that the STS can be oriented to provide the desired thermal environment during on-orbit operations.

Contamination. The STS reaction control system utilizes hydrazine fuel and represents a potential source of payload contamination during deployment and retrieval. However, analysis of or solution to any resulting problems was beyond the scope of this study. The payload bay prior to launch will be purged with Class 100,000 or better dry nitrogen. During reentry the bay will be repressurized with a Class 100, 000 medium by an STS system.

3.3 SYSTEM CRITERIA AND GROUND RULES

Following are the basic criteria and ground rules that have been used throughout this study:

3.3.1 General Criteria

- First SV/STS flight in the first quarter of CY 1982
- Two missions per year, each 120 days minimum; continuous coverage desired
- 10-year operational program with follow-on
- Same film load per mission as present: 104,000 ft per camera
- Each quarter's coverage to be recovered separately
- Best resolution: 2.27 ft GRD at nadir
- Operational orbit: inclination 96.4 deg with perigee located 45° North latitude
- Film retrieval normally by RVs, but portion of take may be returned by STS on resupply/retrieval mission
- All operations from WTR

- Assume no data relay satellites
- ? Survivability aids: 5 percent of all SV fixed weight
- ? Provide for piggyback payloads same as present Hexagon SV 800 lb, but no Mapping Camera Module
- All SVs to contain controlled deboost capability
- Minimum reliability goal for SV (excluding payloads and RV) of 0.85, with greater reliability of deployment/retrieval
- No pipeline/backup vehicle concepts *

3.3.2 Costing Ground Rules

- 1973 rates used for pricing; no rate escalation for subsequent years
- Phase-in from existing SLV Hexagon program
- No major changes
  - Class II changes included, i.e., design deficiencies, parts substitution, normal repair
  - Class I changes excluded
- Assume continuing program past 1992

  Block A flights 1982 - 1992

  Block B flights 1992 -

3.3.3 Safety Criteria

SV/STS safety criteria have been developed, based upon the safety philosophy stated in the previous LMSC Hexagon/STS Study, and on a detailed review of the SV safety problem areas. System safety philosophy may be stated as follows:

- The SV and its support equipment must be designed and operated so as not to endanger the STS crew or its mission.
- SV elements containing hazardous materials must have self-contained provisions to protect the STS from SV-generated potential hazards.
- Equipment and procedures must minimize personnel exposure to hazardous materials and functions.

This philosophy led to the following specific safety criteria which were followed in the study for all SV systems, including the camera system.

3.3.3.1 Safety Design Requirements

Structural

- Design all pressure vessels with a 2:1 burst/operate pressure ratio
- Design all tanks to withstand return environments/loads while fully loaded (abort condition)
- why → No use of magnesium for structure fabrication

Propulsion

- Provide a propellant/pressure dumping capability for both normal operations (to dump residuals) and for an STS abort.
- Provide system status monitors to enable continuous monitoring of system safety parameters (temperatures, pressures, propellant leak detector, etc.)

idiot lts

Pyro/Electrical

- Provide both electrical and mechanical safing (pyro train interruption or blockage) of all high hazard pyros. (High hazard pyros are those which could endanger the STS crew or mission if operated prematurely.)

3.3.3.2 Operational Safety Requirements and Constraints

- Follow best current practice during pre-flight ground test, propellant loading, pyro installation, etc.
- Monitor SV system safety parameters during flight. Maintain safe condition on all pyros when SV is in or near the STS
- Dump excess propellants, depressurize high pressure pneumatics, and safe pyros before SV retrieval. Dump remaining propellant after retrieval.
- In case of an STS mission abort, dump propellants through STS dump system and depressurize high pressure pneumatics before landing STS.

Section 4
SYSTEM OPERATIONAL CONCEPT AND CONFIGURATION SELECTION

This section presents the results of that part of the study work which involved the determination of a system operational concept and selection of a vehicle configuration for detailed subsystem design and costing.

The approach to this phase of the study was to define possible candidate configurations and to conduct a parallel investigation of resupply and maintenance concepts. Tradeoff analysis of the candidate configurations (both resupply and non-resupply versions) and their mission profiles determined an operational concept. A configuration for further study was selected and the operational concept of the SV/STS was determined.

4.1 CANDIDATE CONFIGURATIONS

4.1.1 Approach

The first step in determining an operational concept was to define possible vehicle configurations and their mission profiles. Six Hexagon SV baseline configurations were developed for use in the trade studies to determine the optimum mode of operation; either resupply and maintenance or refurbishment-only.

4.1.2 Considerations

It was initially assumed that the STS could deliver 35,000 lb to, and rendezvous in, an 80 x 100 nm sun-synchronous orbit. Subsequently-supplied data showed that 35,000 lb exceeds the useful STS capability. However, for purposes of comparing resupply versus non-resupply, the total weight does not have a first-order effect on the results. No weight allocations were made for subsatellites or survivability aids. It was also assumed that the SV was not limited to an 82 x 144 nm operational orbit if equivalent camera system performance could be maintained.

The constraints on developing the vehicle configurations were weight (35,000 lb) and size (15 ft dia, 60 ft length).

The primary consideration for resupply and maintenance of Hexagon-type vehicles is the weight of expendables (propellant, pneumatics, film, and RVs) to be resupplied. It was felt that the replacement of worn-out or failed equipment was less complex than replacement of expendables, therefore was not considered in this portion of the study.

4.1.3 Selection of Configurations

Greater payload weight and size capability of the STS naturally leads to a longer life consideration for an optimum SV design. Preliminary analysis showed that an 8-RV configuration was the maximum size SV that could fit into the STS payload bay. Certain operational constraints must be considered, however, before taking advantage of the full 15-ft-diameter STS bay in redesigning the SV.

Figure 4.1-1 shows the effect of SV frontal area on propellant usage. It is desirable, especially at lower altitudes, to hold frontal area to a minimum to reduce the propellant required for drag makeup. For large vehicles such as an 8-RV configuration and its associated 1-year orbit life requirement, weight available for orbit maintenance propellants is very restricted. The SV design is weight-limited by the 35,000 lb STS payload weight capability. The large SV configurations may require operational orbit perigees as high as 140 nm if periodic propellant resupply is not performed. 6-RV configurations with a 9-month orbit life will also require operational orbits higher than the present Block III requirements. Perkin-Elmer developed concepts to modify the optics so that the ground resolution at the higher altitudes is equivalent to the Block III Hexagon ground resolution at 82 x 144 nm.

✓ with Geo. Hess

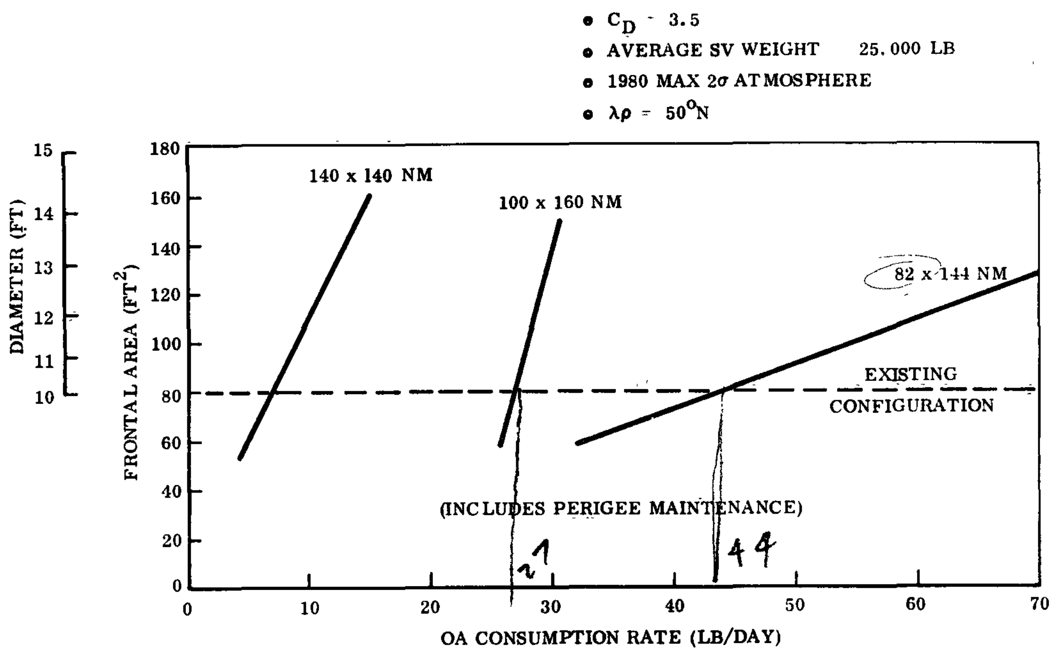


Fig. 4.1-1 Effects of Frontal Area on Propellant Usage

A preliminary reliability analysis was made to determine the extent of added equipment redundancy required to maintain a constant reliability for periods of up to one year. Normal design improvements of the Block III hardware and piece part failure rates consistent with present failure rate goals were assumed for the 1980 era. A dormancy factory of 10 percent was assumed for the analysis. As shown on Fig. 4.1-2, the reliability goal of 0.85 can be met for an on-orbit life of up to one year by adding approximately 100 lb of equipment redundancy. If the vehicle configuration includes a general purpose computer, a higher reliability can be achieved without weight penalty.

GPC

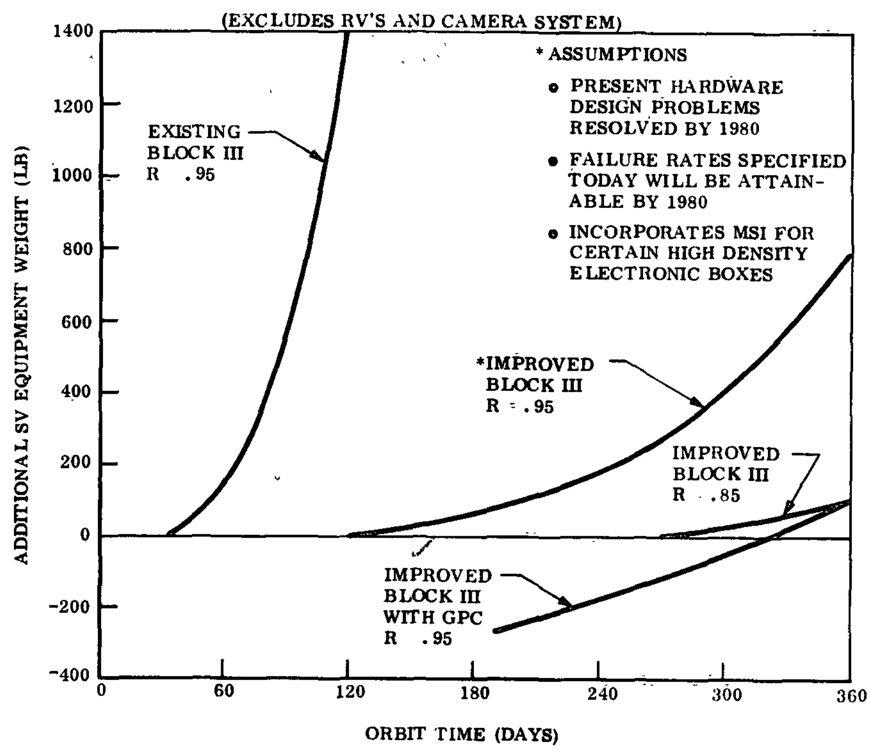


Fig. 4.1-2 SV Equipment Weight Required to Hold Constant Reliability

SV configurations were developed by considering the balance which must be maintained between expendables. That is, as the number of RVs and associated quantity of film is increased, the weight available for propellants required for orbit maintenance decreases to remain within the 35,000 lb weight limit. With the abovementioned considerations in mind, 4, 6, and 8-RV configurations were developed as shown in Fig. 4.1-3.

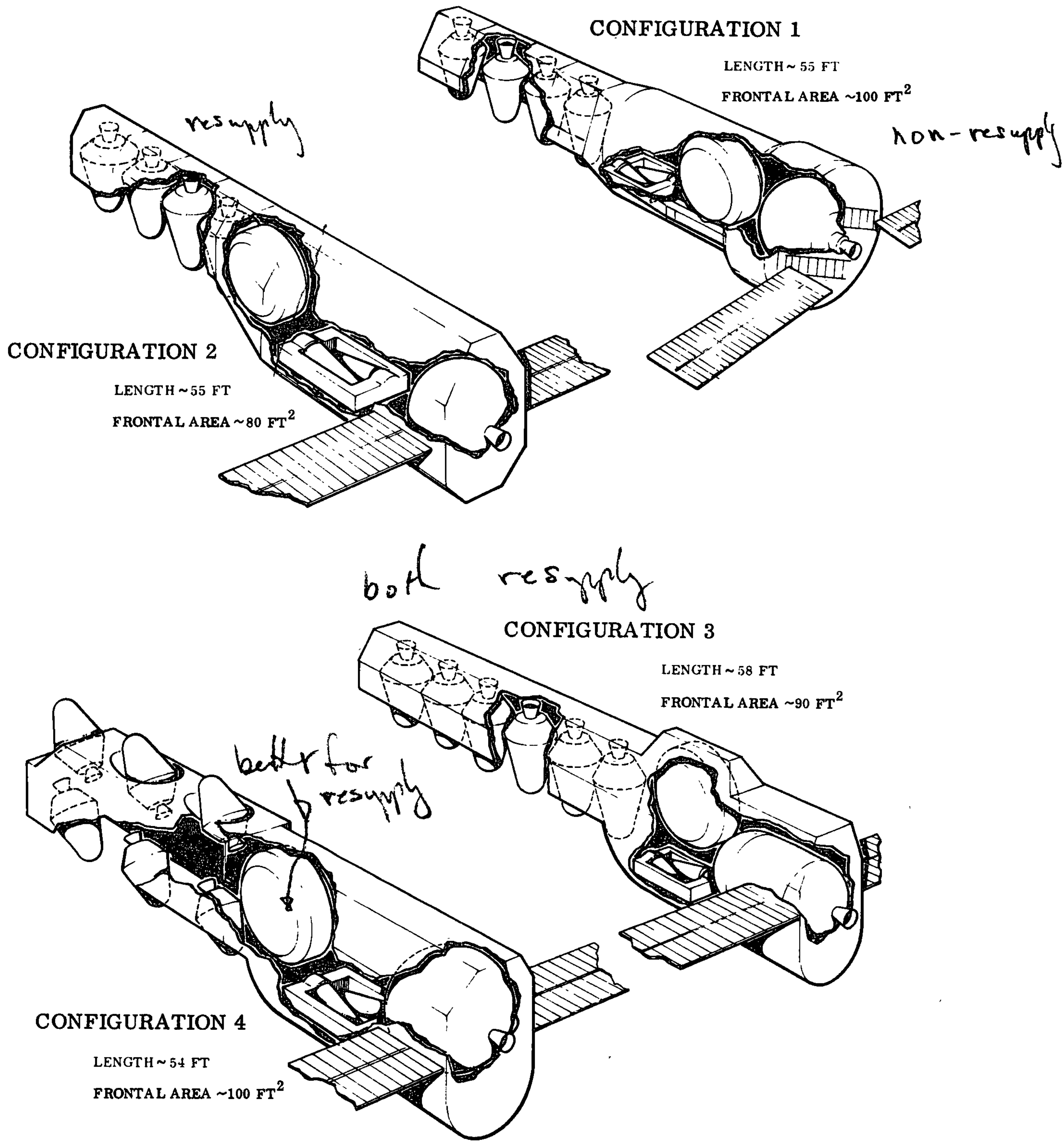


Fig. 4.1-3 Satellite Vehicle Configurations, 4-RV and 6-RV

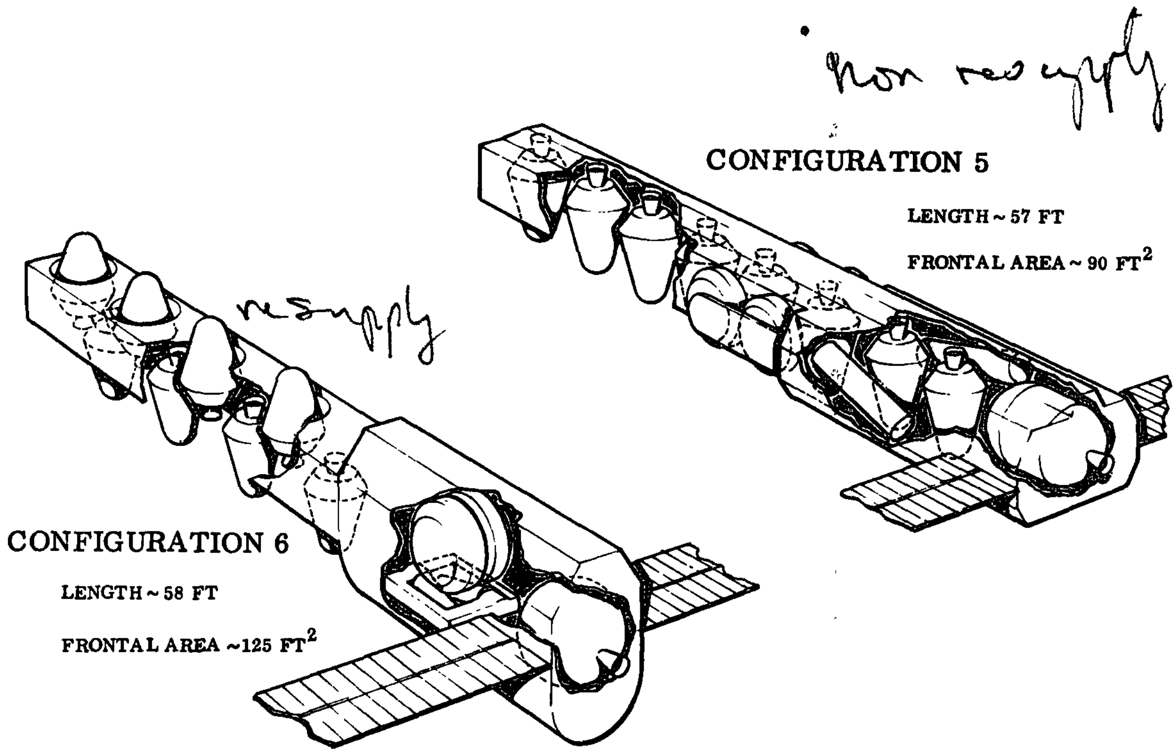


Fig. 4.1-3 (Cont) Satellite Vehicle Configurations, 8-RV

4.1.4 Configuration Descriptions

4-RV Configurations. Configuration 1 is similar to that developed during the Sv/STS Minimum Modification Study, with the propellant tank size increased to allow a 6-month life. This is a non-resupply configuration with the film supply unit in the same location as present Block III vehicles. For ease of resupply the film supply unit is located in a replaceable section with the RVs, as shown in Configuration 2. On this and subsequent configurations, the solar arrays are relocated to allow space on the aft bulkhead for a docking collar to be used during resupply operations. The SV cross-sectional profile is modified from the circular to minimize frontal area.

6-RV Configurations. ~~Both 6-RV concepts~~ can be considered resupply configurations, but Configuration 4 would have a simpler resupply interface because the film supply unit is located in the same replaceable section as the RVs. This allows one operation for replacement of the RVs and film. When the film supply unit is located forward of the camera, the RVs must be stacked back-to-back to remain within the 60-ft-long STS

thermal effects on RV's?

payload bay. This adds some complexity to the film path design and increases the frontal area. The RVs are shown slanted in Configuration 4 to minimize frontal area. Each configuration contains one large film supply unit of sufficient size to accommodate six RVs.

8-RV Configurations. Configuration 5 is designed primarily for non-resupply, with the principal camera components mounted external to the vehicle main structure, thus providing easy access for the refurbishment operations. In this configuration, all RVs may be placed in a single row for simplicity of film path design and minimal frontal area. This hardware arrangement may be applied to the 6-RV or 4-RV configurations to ease the refurbishment operations. Configuration 6 requires that the RVs be stacked back-to-back to remain within the 60-ft-long STS payload bay, resulting in a larger frontal area. This configuration may be resupplied, but the resupply interface is more complex. Two standard-size film supply units are required, and switching of the film path from one supply to the other would be necessary halfway through the mission. Two different RV separation maneuvers are also required due to the back-to-back stacking of the RVs.

None of the configurations shown consider individual resupply (replacement) of RVs on orbit. This approach was ruled out by P-E. The re-establishment of a film path after individual RV replacement required the introduction of numerous cut and splice devices, each of which represents a single point of failure. By replacing an entire section, which includes the film supply, the film path is interrupted at only two points: into and out of the camera system. good

A weight summary of the foregoing configurations is shown in Table 4.1-1, which determines the weight available for propellants. The non-resupply configurations are summarized but, as noted for resupply configurations, approximately 700 lb must be added to the total dry weight for the SV portion of a docking collar and resupply mechanisms. 500 lb is assumed for equipment that must be added to the STS for umbilicals and on-orbit checkout equipment at the Mission Specialist Station. As shown, the weight available for propellants on the larger vehicles is greatly reduced; this requires higher orbit altitudes for these vehicles. A weight breakdown of the kits required to perform resupply operations is shown in Table 4.1-2. The kit weights listed include both hardware and expendables required to resupply the associated SV configuration.

Table 4.1-1
SV WEIGHT SUMMARY

Weight Breakdown (lb)	4-RV		6-RV		8-RV	
Total Gross Weight		35,000		35,000		35,000
Mission Specialist Equipment	500		500		500	
Satellite Vehicle Gross Weight		34,500		34,500		34,500
Camera System (film?)	6125		7450		8650	
Takeups	960		1440		1920	
Recovery Vehicles	3900		5850		7800	
SV Equipment	2400		2550		2650	
Structure	6900		6900		6900	
* Total Dry Weight		20,285		24,190		27,920
Weight Available for Propellants & Gases		14,215		10,310		6,580

whats increase

*Resupply configurations add 700 lb to dry weight

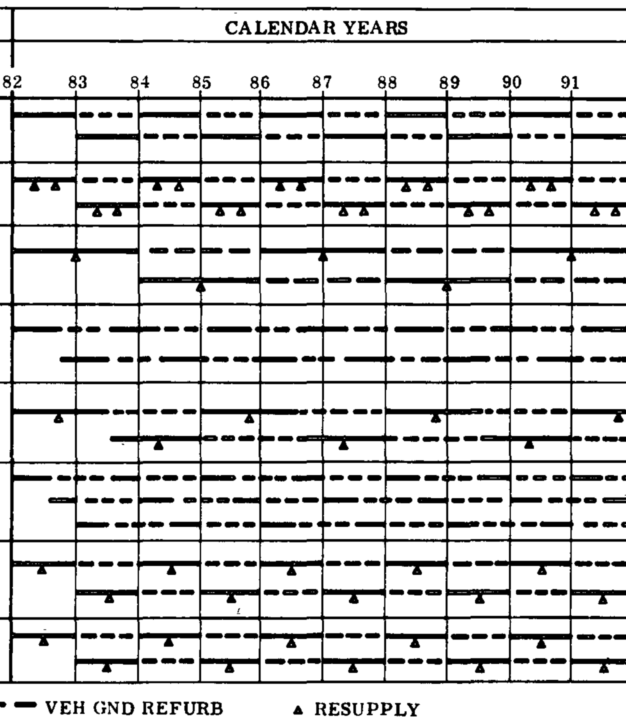
Table 4.1-2
SV RESUPPLY KIT WEIGHT

Weight Breakdown (lb)	8-RV Total Resupply	8-RV HYD Resupply	6-RV Total Resupply	4-RV Total Resupply	4-RV RV Resupply
Resupply Hardware (Boxes, Harnesses, Fwd Section, SU, Etc.)	6950	-	5125	3450	3450
STS-Mounted Equipment, (Plumbing, Docking Collar, Elevators, Tracks, Controls, Tanks, Supporting Structure, etc)	5375	2150	6250	7100	3500
Kit Dry Weight	12,325	2150	11,375	10,550	6950
Expendables	18,600	6600	18,875	18,950	6550
Total Kit Weight	30,925	8750	30,250	29,500	13,500

Dor't understand this one

Figure 4.1-4 is a summary of the foregoing vehicle configurations and their mission profiles selected for operational tradeoff analysis. In the case of the 4-RV, 6-month, non-resupply configuration it is assumed that vehicle refurbishment would take longer than 6 months and that three SVs would therefore be required to support the operational program.

RE-SUPPLY	RV'S	LIFE MOS.	SV WT	KIT WT	RELATIVE ORBIT	STS FLTS
-	8	12	34.5K	-	HIGH	10
HYD	8	12	34.5K	8.7K	MEDIUM	30
HYD RV'S FILM	8	24	34.5K	30.9K	HIGH	10
-	6	9	34.5K	-	MEDIUM	14
HYD RV'S FILM	6	18	34.5K	30.2K	MEDIUM	14
-	4	6	34.5K	-	LOW	20
HYD RV'S FILM	4	12	34.5K	29.5K	LOW	20
RV'S	4	12	34.5K	13.5K	MEDIUM	20



allows 270 days to refurb.

3 SV 180 day 182

assumes one resupply the bring down

Fig. 4.1-4 Mission Profiles

4.2 RESUPPLY AND MAINTENANCE CONCEPTS

This section describes the study of operational and associated design concepts for resupply (on-orbit replacement of expendables) and maintenance (on-orbit replacement of failed or life-limited items).

4.2.1 Approach

LMSC selected a parametric approach to scope the study problem; specifically, the task was approached by identifying the key variables in resupply and maintenance of a Hexagon-type SV. The defined variables fall into three categories: (1) candidates for resupply/maintenance, (2) SV/STS operational modes for resupply/maintenance, and (3) methods of resupply/maintenance.

For convenience in addressing the subject, candidates for SV resupply/maintenance were assumed to consist of three sizes of Space Replaceable Units (SRU): modules, subsections, and sections, as visualized in Fig. 4.2-1.

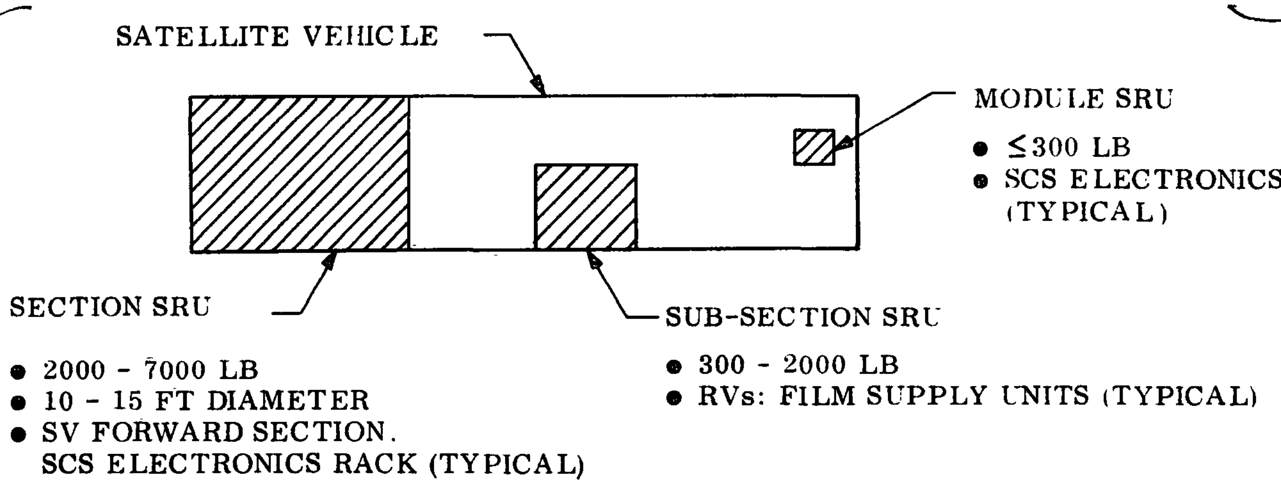


Fig. 4.2-1 Space Replaceable Units (SRU)

Transfer of fluids and pressurants was also considered. For this portion of the study there was no requirement to distinguish between resupply and maintenance. The parametric approach eliminates sensitivity to the purpose of the SRU replacement, whether it be for resupply or maintenance.

Replacement of the SRU could possibly be achieved under several SV/STS operational modes:

Formation Flying	• SV and STS maintain constant separation and relative attitudes • No physical coupling exists between SV and STS • Active reaction control system on each vehicle
Soft Dock	• SV-to-STS spatial orientation is provided by a minimum of one RMS arm
Hard Dock	• SV is rigidly attached to the STS by a docking collar – NASA neutered type

The methods of accomplishing the SRU replacement were considered to consist of:

EVA	• A suited astronaut operating exterior to the STS flight deck
RMS	• Employ one or both RMSs for removing and replacing SRUs
Special Equipment	• SV program-provided equipment mounted on the STS that will be employed for SRU removal/replacement

Other candidates, operational modes, or methods can obviously be identified and questions asked as to why their exclusion, e.g., the "teleoperator" system. This concept is a free flying spacecraft equipped with manipulators and controlled by an operator on the STS. Its exclusion was based on the fact that if NASA does develop it, first availability of a reliable system will be somewhat later than when the STS becomes operational.

The study approach is shown in Fig. 4.2-2. The three key families of variables were combined into a 3 x 3 x 4 decision matrix of 36 cells: 4 types of SRUs, 3 modes, and 3 methods. This matrix made it possible to clearly scope the problem.

The matrix is shown in more detail by Fig. 4.2-3. Under EVA, RMS, and special equipment, there are submethods shown as 1, 2, 3 . . . . . N . Later in this section various alternative sub-modes are presented and discussed. In the evaluation process, sketches of more than 40 sub-modes were developed for study. Impossible/infeasible cases were discarded by inspection. Design and operating concepts were developed for surviving cases which were subjected to the criteria shown below in decreasing order of importance, i.e., weighting factors assigned:

- Safety
- Reliability
- Technical risk
- Cost
- Weight
- Human ease
- Time to perform operation

question order of importance

Leading candidates from the process were selected for further design and detailed study.

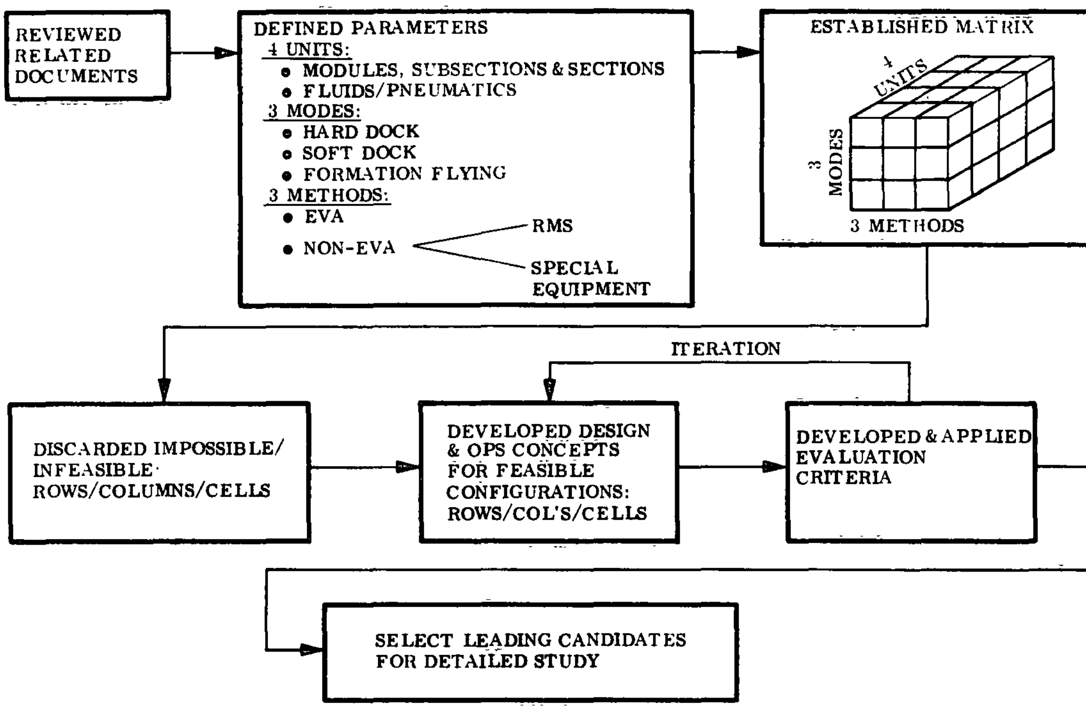


Fig. 4.2-2 Resupply and Maintenance Approach

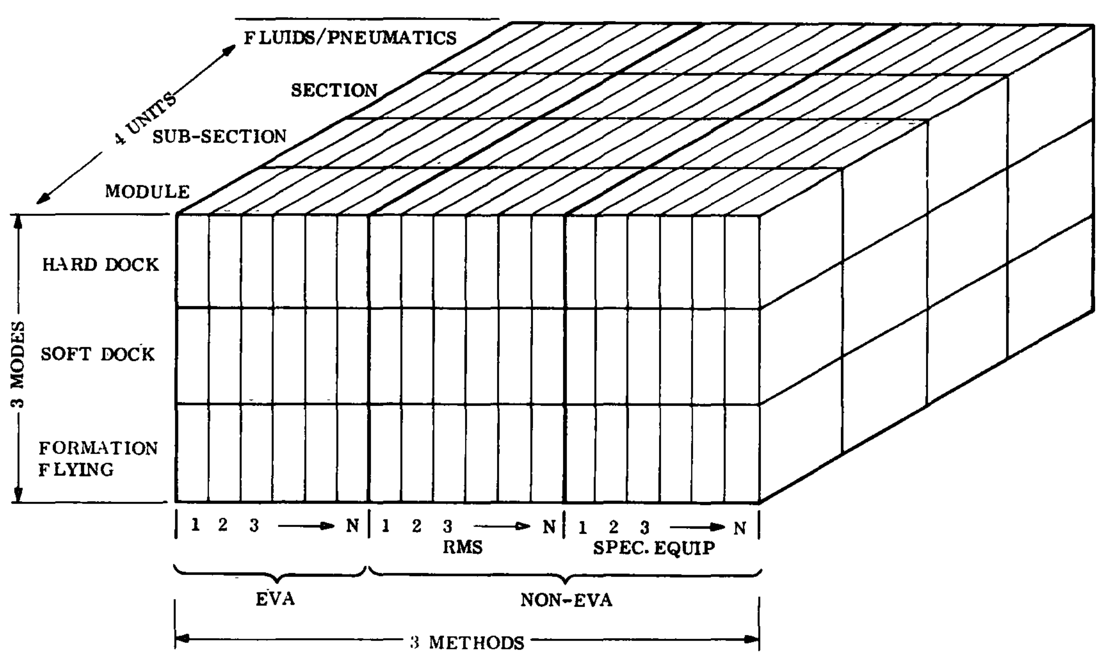


Fig. 4.2-3 Matrix of Alternatives

4.2.2 Evaluation of Operational Modes

Three operational modes were evaluated: Formation Flying, Soft Dock, and Hard Dock.

Formation Flying Mode. The study showed a number of major problem areas with the Formation Flying mode.

- Maintenance of a spatial relationship would be difficult at the altitudes considered for Hexagon SV resupply/maintenance (~100 nm); differential drag due to atmospheric forces will tend to separate the vehicles. Flying at higher altitudes to eliminate this problem incurs a direct penalty on payload weight because the weight of the OMS propellant needed to achieve higher orbits is charged to the payload.
- The vehicles must fly close enough that SRU replacement can be accomplished by one or a combination of the three methods studied. The requirement therefore exists that the SV fly above the STS open payload bay. Mutual contamination of the vehicles by their reaction control systems then becomes a problem.
- SV dynamic response when a SRU is removed or replaced is an unknown and was not studied. Potential hazards do exist in that the SV could go unstable and impact the STS.
- The last area of concern is one that affects the design of the SV. The requirement exists that removal of one SRU will not cause the SV to be disabled. Redundant capability must exist in other portions of the vehicle to ensure that the STS can control/monitor the SV during this operation and that the SV reaction control system is functional.

LMSC therefore discarded the Formation Flying mode of operation on the basis of feasibility and technical risk in combination with STS and crew safety.

Soft Dock Mode. In the Soft Dock mode the SV and STS are coupled by at least one RMS arm. Reaction control systems on both the SV and STS are shut down to avoid the mutual contamination problem. Two additional considerations argued in favor of disabling the STS reaction control system: its 1000-lb thrusters would perturb spatial orientations, and the STS RCS fuel consumption would be excessive for small deadbands (114 lb/hr at 0.1 deg and 2 lb/hr at 0.5 deg).

A two-fold, parallel study approach was taken as shown in Fig. 4.2-4: (1) design requirements and concepts for EVA and non-EVA methods were developed and evaluated, and (2) concurrent analysis was undertaken of two-body system dynamics, differential and aerodynamic drag, and RMS capability.

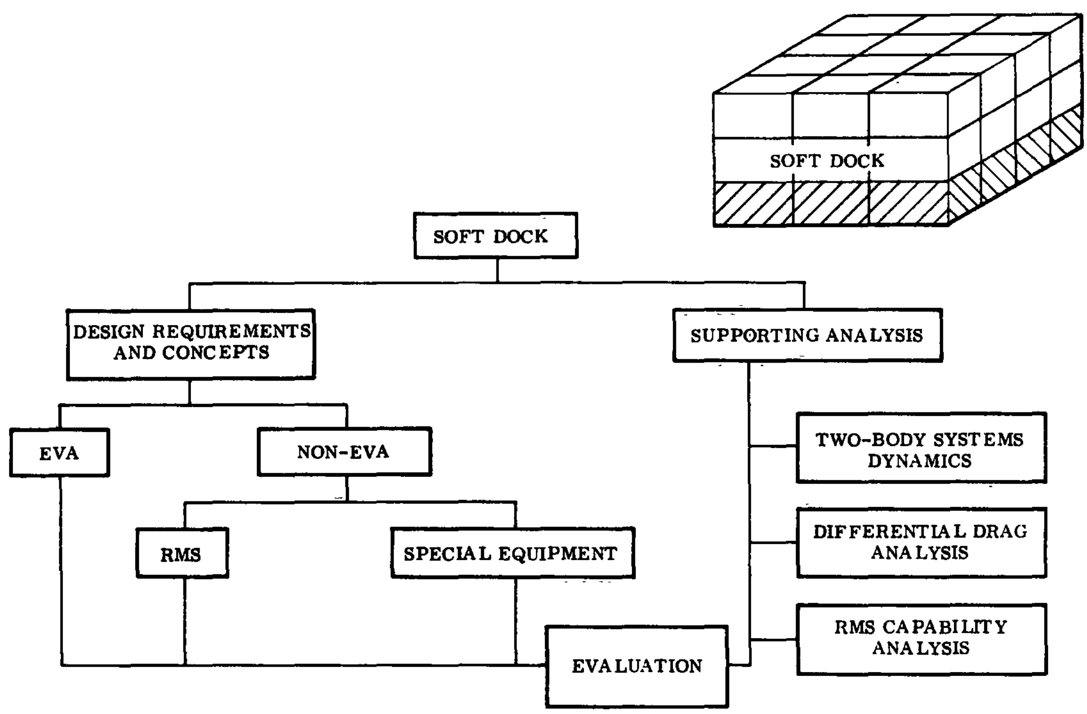


Fig. 4.2-4 Soft Dock Mode

For the two-body system dynamic analysis, the RMS was represented by a 47.5-ft-long aluminum tube 1 ft in diameter, with no joint flexibility considered. The RMS in turn was attached to the SV c.g. and the combined system frequency was found to be 0.1 Hz. Using residual SV-STS attitude rates of 0.1 ft/sec and assuming no damping in the RMS, the excursions for a typical target point on the SV relative to the STS were computed as:

X ±1.80 in.

Y ±1.80 in.

Z ±0.02 in.

This typical target point is representative of a point on the SCS. Any SRU replacement system must be able to accommodate this range of SV excursions. The problems are magnified if the SV is not held at its c.g. by the RMS. Target point excursions up to ten times those identified above will exist if the SV is held at one end by the RMS.

In consideration of the area exposed to the low-orbit wind stream, it was recognized that the differential drag would cause the SV/STS relative positions to change. The RMS would therefore require preprogramming to compensate for drag variance so that the SV target point would be held in constant position relative to the STS. It is important to note that the RMS maximum force capability normal to its tip is 10 lb. The question arises whether differential drag has been considered in NASA Missions 3A and 3B timelines where deployment/retrievals are accomplished at altitudes of 50 to 100 nm. The analysis results summarized below demonstrate that the drag forces will either impede or restrict RMS operations below certain altitudes.

SV/STS Flight Direction	Force Developed Normal to RMS Tip*		
	At 100 nm Altitude	At 80 nm Altitude	
X-Axis	0.16 lb	0.62 lb	Both bodies parallel
Y-Axis	0.66 lb	2.61 lb	
Z-Axis	0.40 lb	1.55 lb	
STS in X-Axis Flight, SV Normal to STS	0.97 lb	3.80 lb	

*50 ft by 15 ft diameter SV.

Any RMS capability analysis is subjective in that no RMS system exists as of this date. However, the RMS design entails a significant state-of-art advancement. Available publications, LMSC expert opinion, and comments from NASA personnel raise a number of unanswered questions, particularly as to whether the as-built RMS will satisfy currently stated performance specifications.

In addition to the above, when the RMS attaches to the SV, it cannot orient the SV to all the attitudes required for SRU replacement, i.e., if the RMS attaches to the bottom side of the SV the upper side is not accessible. Therefore, several RMS attach points would be required on an SV to ensure complete access to the entire SV periphery.

Based upon the above analysis and evaluation, the Soft Dock mode was discarded, except for possible use during resupply of propellants where propellants are transferred to the SV from STS-mounted storage tanks.

Hard Dock Mode. In the Hard Dock mode, the SV is assumed to be rigidly attached to the STS via a docking collar of the NASA neutered type. It is immediately obvious that the Hard Dock mode overcomes many of the deficiencies of the two operational modes previously discussed. Relative motion of the SV to the STS due to differential drag and RMS dynamics is eliminated. No contamination problems exist because both SV and STS reaction control systems can be shut down after docking. LMSC therefore selected this concept as the operational mode for resupply and maintenance. As shown in Fig. 4.2-5, design requirements and concepts for EVA and non-EVA methods were developed and evaluated. The leading candidates which emerged from this process were then refined to permit final evaluation.

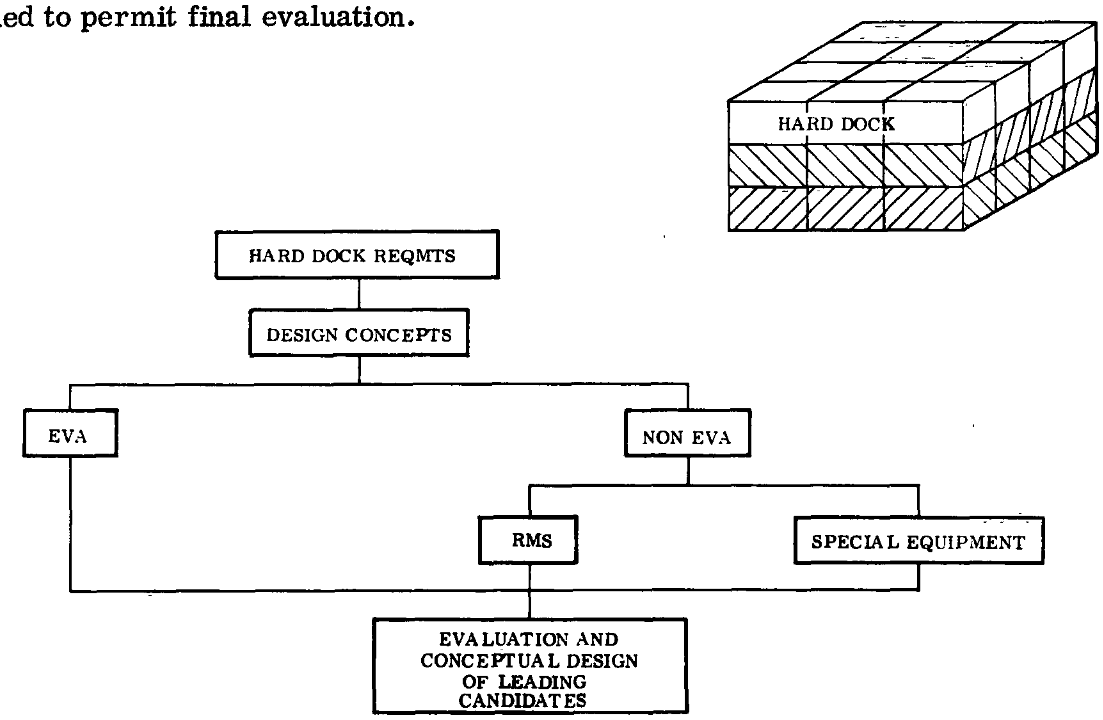


Fig. 4.2-5 Hard Dock Mode

4.2.3 Evaluation of Hard Dock Resupply/Maintenance Methods

Hard Dock/EVA Evaluation Findings. Hard Dock/EVA cases were developed and documented by design sketches. The best EVA cases were evaluated against non-EVA (RMS and special equipment) methods. In only one case (the replacement of modules under 100 lb) did EVA rank essentially in a tie with RMS and special equipment. Except for that single case, the EVA mode was consistently inferior to RMS and special equipment methods. The EVA evaluation necessitated a consideration of the pertinent limitations and capabilities of a suited EVA astronaut, some of which are illustrated below:

- Force range (push/pull, up/down) with Dutch shoes is from 10 lb (4 sec duration) to 40 lb (1 sec duration)
- With body harness, 100 lb mass maximum handling capability
- 1 to 4 hours at 500 kilocalories per hour maximum rate of energy expenditure
- Plus or minus 0.01 in. positional accuracy
- High resolution and good depth perception with direct vision.

Of the possible EVA functions evaluated, the most plausible is the enhancement of the RMS/special equipment method in an active role (e.g., sensor in the loop), monitor role (e.g., sensor and manual override), troubleshooter role, or a combination thereof. Based on this evaluation, the EVA method was eliminated from further study and the method of RMS/Special Equipment while hard-docked was pursued.

Hard Dock RMS/Special Equipment. Two SV/STS docking relationships (parallel and right angle) were explored. In the case of the parallel hard-docked relationship, it was assumed that initial docking would be at right angles and that the docking mechanism, after capture, would move the SV into a parallel relationship with the STS.

In the exploration of options in the hard docked relationships, several significant constraints became apparent. (1) The SRUs must be held in a storage unit in the STS prior to transfer to the SV. (2) The supporting mount in this storage unit can be similar to the supporting mount in the spacecraft; and the transferring mechanism can attach to some other location on the SRU. If the SRU is not mounted in the STS storage unit in a mounting similar to its mounting in the SV, it must be mounted in the STS storage unit on a transferring mechanism that plugs it into the SV supporting mount. (3) A storage location must also be provided for each size SRU replaced, and it must be available to store the old SRU prior to handling the new SRU. This constraint means that one location for each size SRU must be available in the storage unit, or the transferring mechanism must remove the old SRU from the spacecraft and support it in some manner until the new SRU can be removed from the storage unit and installed in the SV (reference Fig. 4.2-6). It is also a requirement, in all cases, that a suitable means be provided by the transfer mechanism to properly position the equipment for replacement and to operate all latching mechanisms and electrical connectors.

? might like next system

A number of handling and transfer concepts appeared to be possible. Of these concepts, those that appeared to be practical are discussed in the following paragraphs.

(a.) Remote Manipulator System Concept. The RMS (Fig. 4.2-6a) is adaptable for replacing equipment when the vehicle is hard-docked. The most efficient SRU storage unit appears to be a rotating unit, in the STS bay, which allows access to all SRUs. The RMS, equipped with proper controls and viewing devices, can remove the old SRU from the SV and place it in a storage location in the storage unit, then remove the new SRU from a storage unit location and place it in the SV. It is readily apparent that the SV must be rotated on the docking ring to allow proper access for this operation.

why not external to STS

The length of the RMS (47.5 ft) presents a restriction on this concept. A typical Hexagon SV is approximately 55 ft in length. The RMS will be able to provide access to approximately 37.5 ft of the SV. The other 10 ft of RMS length is accounted for by the fact that the SV docks approximately 10 ft above the payload bay rail, where the RMS is mounted. All SRUs must be located toward one end of the SV or docking

that's the way it is designed

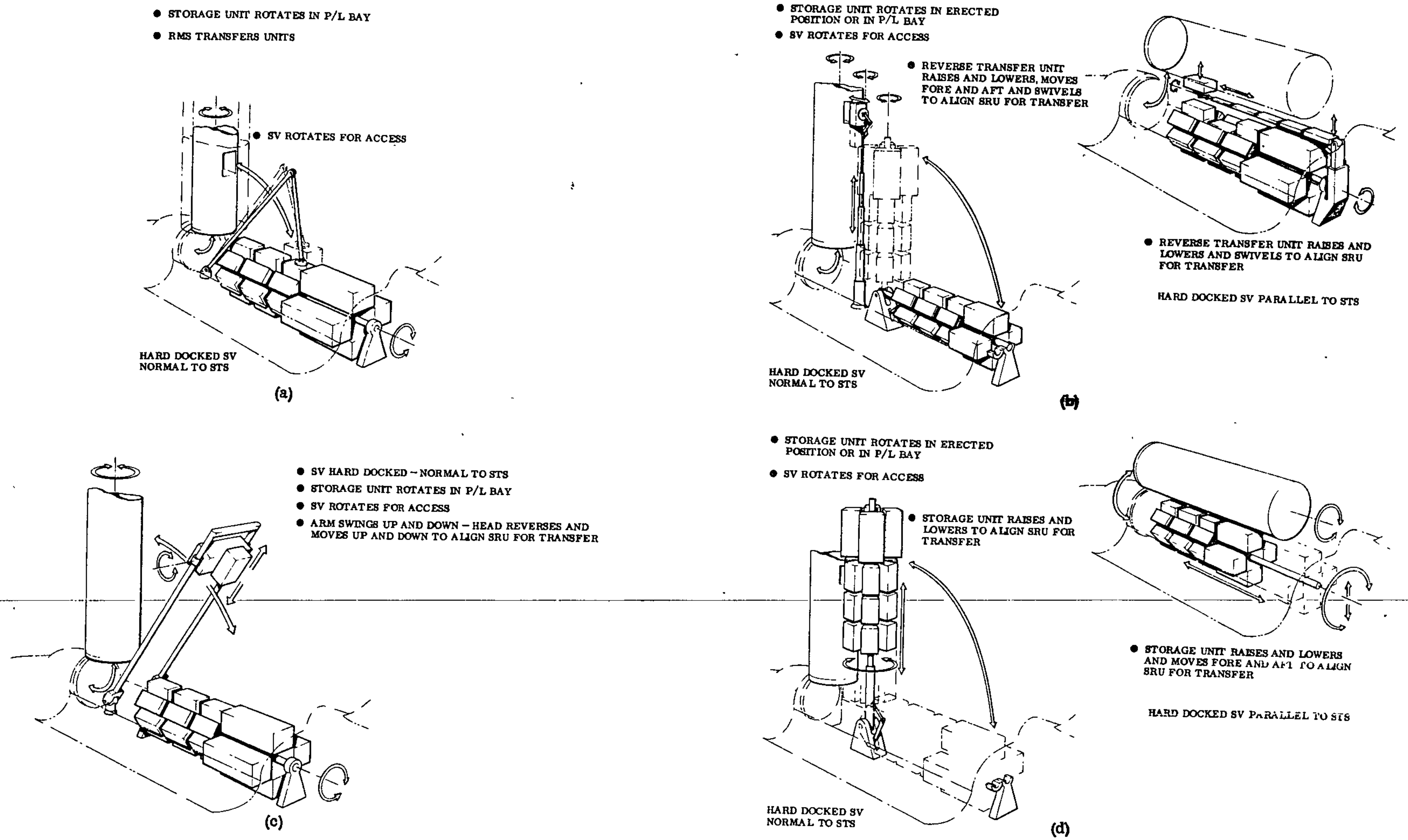


Fig. 4.2-6 Hard Dock Mode Handling and Transfer Concepts

is that really needed.

collars must be provided on both ends. With the latter concept, once initial SRU replacement is completed, the SV could be redocked in reverse orientation and the RMS would provide access to remaining portions of the SV.

(b.) Reverse Transfer Concept. This concept (Fig. 4.2-6b) brings the rotating storage unit and the rotating SV into parallel relationship, either by swinging the SRU storage unit out of the STS bay or by swinging the hard-docked SV down to the storage unit in the payload bay. The transfer of SRUs between the SV and the storage unit involves a positive position type mechanism that rotates the equipment 180 deg during the transfer, to align with the attachment points. The same removal and storage activity must precede the installation of the new SRU. In this concept the same SRU mounting interface is used in the storage unit and SV.

(c.) Swing Arm Concept. This concept (Fig. 4.2-6c) has a rotating SRU storage unit in the STS bay and has a sturdy mechanism which swings between the SV and the storage unit. The transfer head on the mechanism reverses, and moves up and down, to gain access to the storage unit and SV locations. In this concept the transfer head can (1) remove the old SRU; (2) reverse while still holding that SRU; (3) travel to the storage location and pick up the new SRU; (4) reverse while holding both SRUs; (5) place the old SRU in the same storage location as that just vacated by the new SRU; and (6) then travel to the SV location and install the new SRU. In this concept a spare set of storage locations is not required. The same SRU interface is used for attaching to the SV and the storage unit.

(d.) Direct Transfer Concept. This concept (Fig. 4.2-6d) eliminates the intermediate transfer mechanism. The SRUs are stored in the storage unit on direct transfer mechanisms that extend from the storage unit to install the SRUs in the SV. In this concept a full set of open bays in the storage unit is required for each size SRU. The storage unit and the SV must rotate and the storage unit must swing out of the STS, or the SV must swing down to parallel the storage unit in the STS bay.

4.2.4 Selected Resupply and Maintenance Concept

Good looks.

The selected concept uses the best features of the prior-described concepts and also provides the lowest complexity design approach. (See Fig. 2-3). The SV is hard docked at right angles to the STS and rotated down parallel to the payload bay. The SV rotates on the docking mechanism. The SCS SRUs are stored on a rotating storage unit. A direct transfer mechanism at each location on the unit extends to remove the old SRU, stores it in a spare location in the storage unit and installs the new SRU. The recovery section is replaced as a complete section. The new section is carried in the lower half of a storage assembly, also located in the STS bay. The upper half of the assembly is open to receive the old section when it is detached from the spacecraft.

With this concept, the storage assembly extends from the payload bay to engage and support the old section. When the old section is held securely, the recovery section SV connections are withdrawn, and the storage unit rotates to separate the old section from the SV and bring the new section into position. When the new section is in position the connections are extended to secure the new recovery section. The old section is, at this time, in the lower half of the storage assembly. The assembly is then withdrawn back into the payload bay. This concept allows replacement of recovery sections that contain failed RVs or the last RV filled with film. This latter case avoids the expenditure of an RV for film recovery if an STS mission is scheduled for resupply/maintenance. Propellant resupply can also be best accomplished when the spacecraft is parallel with the STS. A probe will be extended into the spacecraft to enable the transfer of propellants and gases as shown.

*

Further Studies. As indicated in Section 2 of this report, LMSC selected an SV/STS operational concept that does not use resupply and/or maintenance. Retrieval and refurbishment was found to be the most economical concept. However, if the answers had been reversed, then LMSC would have continued to study resupply and maintenance during the second half of the study. Areas of work would have included:

need to check cost analysis.

Space Replaceable Units

- Mechanical interfaces – alignment, fasteners, sizes
- Electrical interfaces – power, data bus
- Test and diagnostic provisions
- Environmental requirements
- Film path considerations

Resupply and Maintenance Support Equipment

- Design of special equipment for SRU replacement
- SRU storage provisions in STS
- SV/STS docking collar
- SRU-to-STS interfaces
  - Control and monitoring
  - Diagnostic
  - Safety

EVA Crewman Requirements and Role

- Active role (sensor and man in loop)
- Monitor role (sensor and manual override)
- Troubleshooting role
- Combinations

4.3 OPERATIONAL TRADEOFF ANALYSIS

4.3.1 Approach

All resupply and non-resupply configurations developed in Section 4.1 meet the same mission requirements. In order to derive an operational concept, other criteria must be identified for use in the concept tradeoff and selection. Technical feasibility is a possible tradeoff area; however, all configurations, both resupply and non-resupply, are technically feasible in concept. The only other area which appears to be a significant variable among configurations is cost. Cost was thus chosen as the tradeoff parameter to be used in the operational concept selection.

4.3.2 Costing Method

In terms of costing techniques, a system may be estimated from the bottom up or from the top down. Since bottom-up costing requires a design more detailed than those developed in Section 4.1, the top-down technique was chosen.

The first step is to determine how funds are apportioned among various segments of a space program. The data presented in Fig. 4.3-1 is derived from annual NASA budget justification documents available to LMSC. The figure shows the budget allocations for the major segments of NASA unmanned space programs by year. For purposes of this study, the funding allocations for the year 1971 were chosen as representative.

Not sure I understand this one.

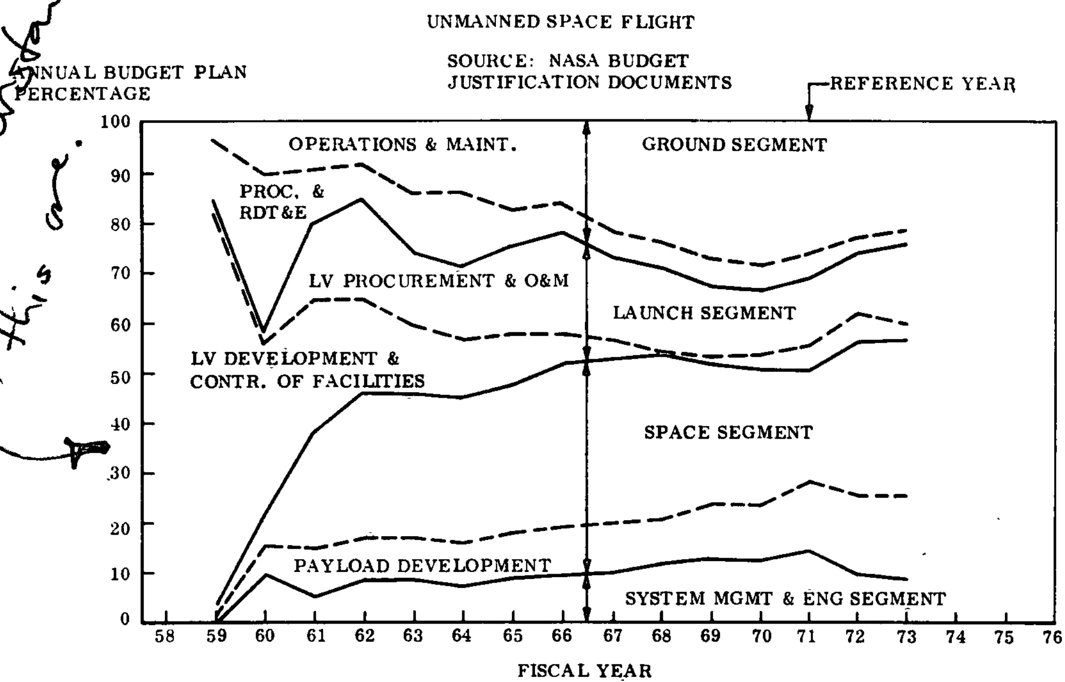


Fig. 4.3-1 Sample Budget Allocation

The reference year data was then divided into allocated and program-related categories, from information contained in the NASA budget justification documents, as shown in Table 4.3-1. Allocated funding covers the portion of items such as tracking networks, launch base operations, and central control operations that are charged to the program. These charges ~~were not~~ included in the tradeoff.

One adjustment was necessary since NASA uses an electronic rather than mechanical data recovery system. It was assumed that the value apportioned to the electronic data recovery system is the same as that which would apply to a mechanical system such as on the Hexagon vehicle. As shown in the table, 62 percent of the NASA unmanned space flight budget is for program-related items. This information will be used to determine those program-related costs of the configurations which cannot otherwise be estimated (e.g., launch cost and camera system costs).

Table 4.3-1

NASA SPACE PROGRAM FUNDING – FY 1971

	Allocated	Program-Related
Program Management & Engineering	.175	.043
Space Segment		
• Spacecraft Integration		.221
• Payloads		.137
• Mechanical Recovery System(1)		(.089)
Launch Segment		
• Product Improvement & Stage Integration	.053	
• Hardware		.062
• Support & Vehicle Integration		.052
• Launch Base & Range Operations	.014	
Ground Segment		
• Tracking & Data Acquisition	.134	
• Electronic Data Recovery System		.089
• Program Data Analysis & Planning		.016
• Facilities	.004	
Total	.380	.620

(1) If used delete electronic data recovery system

Source: NASA budget justification documents

Based on information available to LMSC, the cost through first launch of some representative programs are as shown below:

Program	Weight (lb)	Total Cost Through First Launch	Dollars/lb in Orbit
OAO	3900	$166.6 M	42,700
110	7200	$311.4 M	43,200
Hexagon	20,600	$925.0 M	44,900

a number for a new undeveloped program.

A survey of the program weights results in the observation that the cost per pound in orbit for these various programs is remarkably close and, for purposes of this study, was assumed to be $45,000 per pound. Of this number, 62%, based on NASA unmanned space flight budget allocations, is program-related costs. Thus, the cost of development and the first flight system into orbit is $28,000 ($45,000 x 62%) per pound.

reduce number by considering use of hardware already developed.

4.3.3 Procedures

The cost of development, procurement, integration, and refurbishment of the SV equipment, STS-borne equipment, and the resupply kit were included in the cost estimates of each vehicle configuration. The estimating procedure required dividing each vehicle configuration's weight into that which could be attributed to the existing program and that which would be new (Table 4.3-2).

they did it

Table 4.3.2

CONFIGURATION WEIGHT BREAKDOWN

Configuration			Vehicle		Kit	
No. of RVs	Life (mo)	Resupply	Existing (lb)	New (lb)	Existing (lb)	New (lb)
8	12	-	10,450	4350	-	-
8	12	HYD	10,450	4350	825	1325
8	24	HYD RV & film	10,450	5050	6950	5375
6	9	-	10,450	3775	-	-
6	18	HYD RV & film	10,450	4475	5125	6250
4	6	-	10,450	1900	-	-
4	12	HYD RV & film	10,450	4100	3450	7100
4	12	RV	10,450	4100	3450	3500

How does this compare with figure 4.1-1 4.1-2 ? on page 4.1-8

That portion of the configuration attributable to the existing program was costed as continuing procurement of the existing program, that is, no development cost was charged against that portion of the vehicle. The new portions were charged with development cost at a rate of 4.79 times the first article cost. The factor of 4.79 was based on experience with the Hexagon program. Camera system development, procurement, and refurbishment cost estimates were furnished by P-E. The cost of an STS launch was ground ruled for the study at $10.5 million. All other recurring cost charges were computed to match the NASA profile as specified in Table 4.3-1. The development, procurement, and refurbishment charges were time-spread per DoD program experience data as published in Congressional hearing records. A learning curve of 83 percent was employed for procurement and refurbishment, which requires no design changes. This rate has been experienced on some program runouts in the past, and is considered conservative for this tradeoff analysis. Design changes would reduce the benefits of this effect. It was assumed that the SV can be refurbished for at least 10 years. Also, the STS would be available when needed and no unscheduled resupply missions would be required. The cost of expendables was omitted as negligible.

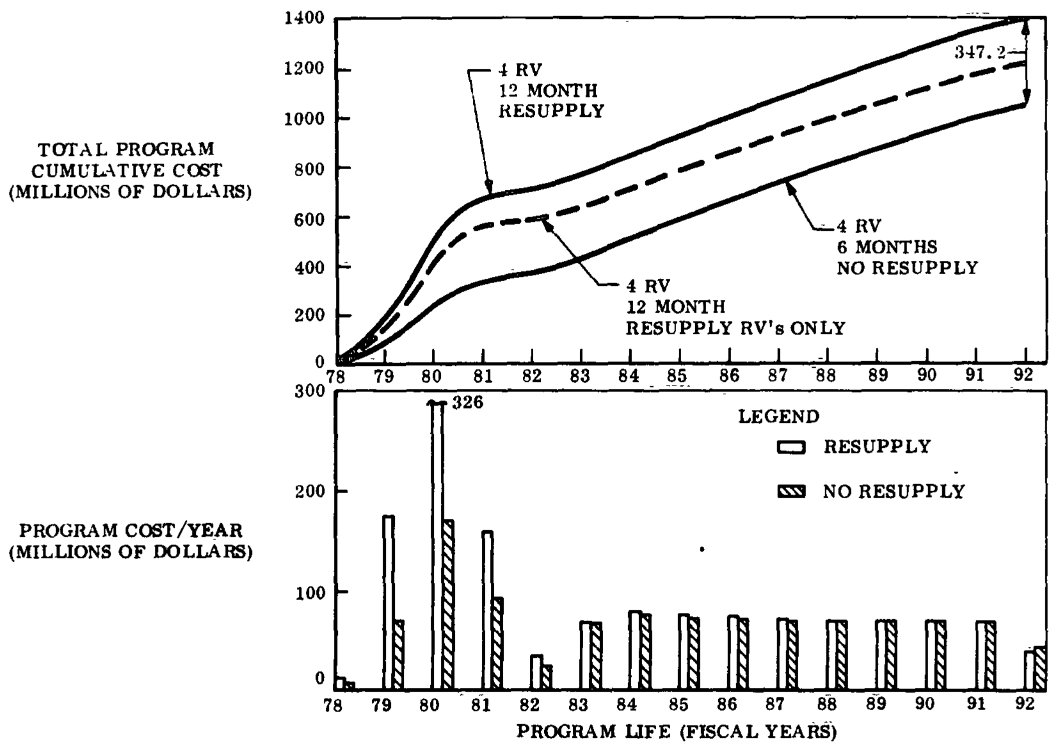
Show me.

The cost estimates are in current dollars. Finally, the costs presented using the above-described technique are considered valid only for configuration-to-configuration comparison and are not intended as a precise total cost estimate.

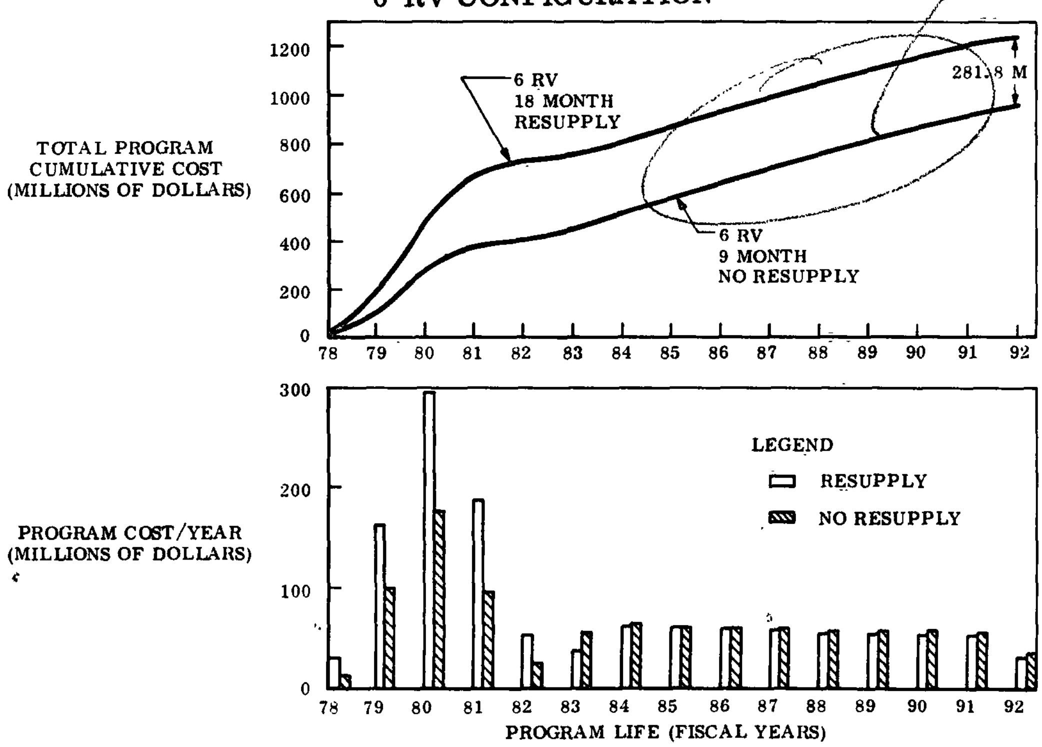
4.3.4 Results

The cost estimate results for the various configurations are presented in Fig. 4.3-2. Both total program cost and annual cost are shown for the 10-year operational period. In all cases, the cost of a resupply configuration is higher than a comparative non-resupply configuration. This is due primarily to the greater costs involved in developing a resupply kit and an SV capable of being resupplied.

4-RV CONFIGURATION



6-RV CONFIGURATION



why are these parallel? resupply has less factory outs

Fig. 4.3-2 Program Cost Summary

8-RV CONFIGURATION

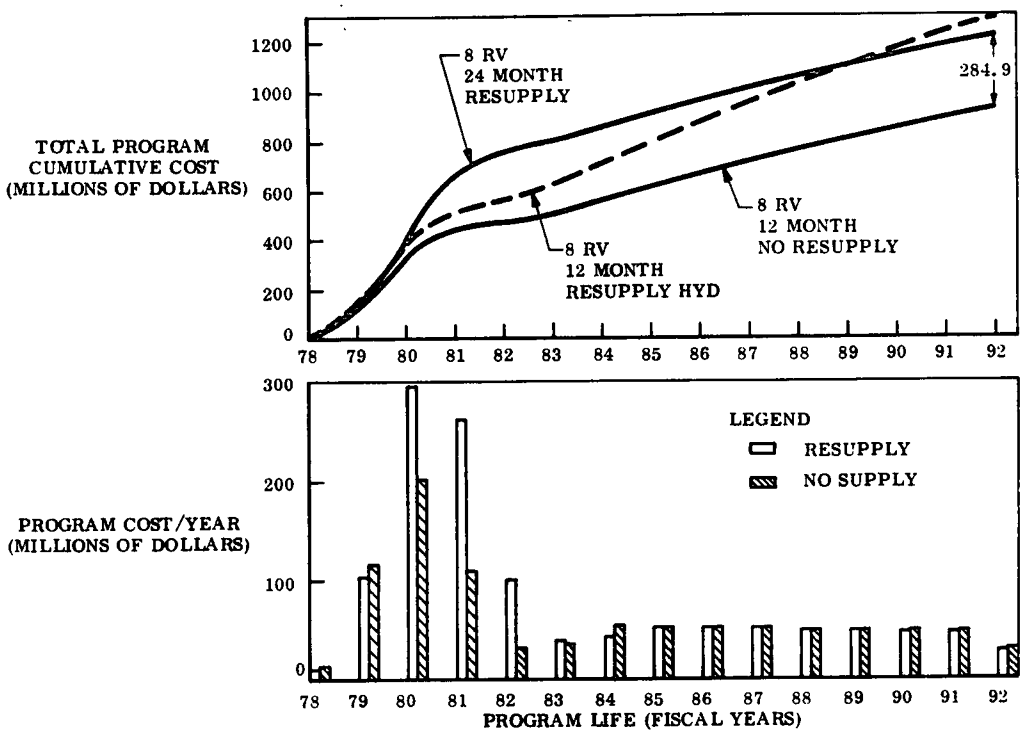


Fig. 4.3-2 Program Cost Summary (Cont)

In the 8-RV comparison an additional configuration of resupplying hydrazine only was investigated and, although development and procurement costs are lower than the total resupply case, the recurring costs are higher due to more STS launches.

In the 4-RV comparison an additional configuration of resupplying only RVs was investigated. As could be expected, the cost of this configuration is less than that of the total resupply configuration because of the lower development costs.

The development phase of the program requires large annual expenditures of funds, but the results show that the annual recurring costs of the resupply and non-resupply configuration are nearly the same.

Table 4.3-3 summarizes the estimated costs of all the configurations studied, including the total development and procurement costs and the total recurring cost for this 10-year period.

Table 4.3-3
COST ESTIMATE SUMMARY

COST ESTIMATE		CONFIGURATION							
	RVs	8	8	8	6	6	4	4	4
	LIFE	12	12	24	9	18	6	12	12
	RESUPPLY		HYD	HYD RV & FILM		HYD RV & FILM		HYD RV & FILM	RV
• INVESTMENT									
DEVELOPMENT		274.4	327.5	501.3	225.4	464.9	141.5	459.2	357.2
PROCUREMENT		194.5	220.1	287.1	178.4	265.5	223.3	248.5	227.2
SUBTOTAL		468.9	547.6	788.4	403.8	730.4	364.8	707.7	584.4
• OPERATIONS – 10 YEARS									
REFURBISHMENTS, REPLACEMENT, AND RE-LAUNCHES		451.5	733.9	416.9	554.2	509.4	679.9	684.2	633.7
• TOTAL COST		920.4	1281.5	1205.3	958.0	1239.8	1044.7	1391.9	1218.1

4.3.5 Sensitivity Analysis

Since many assumptions were necessary which affect the final conclusion, a sensitivity analysis was performed on the variables in the economic model. The technique used was to vary one variable about the value used for the analysis and note the impact on the delta cost between the resupply and non-resupply configurations. The results are presented in Fig. 4.3-3. The distance between the two heavy lines represents the delta cost between configurations for the conditions stated.

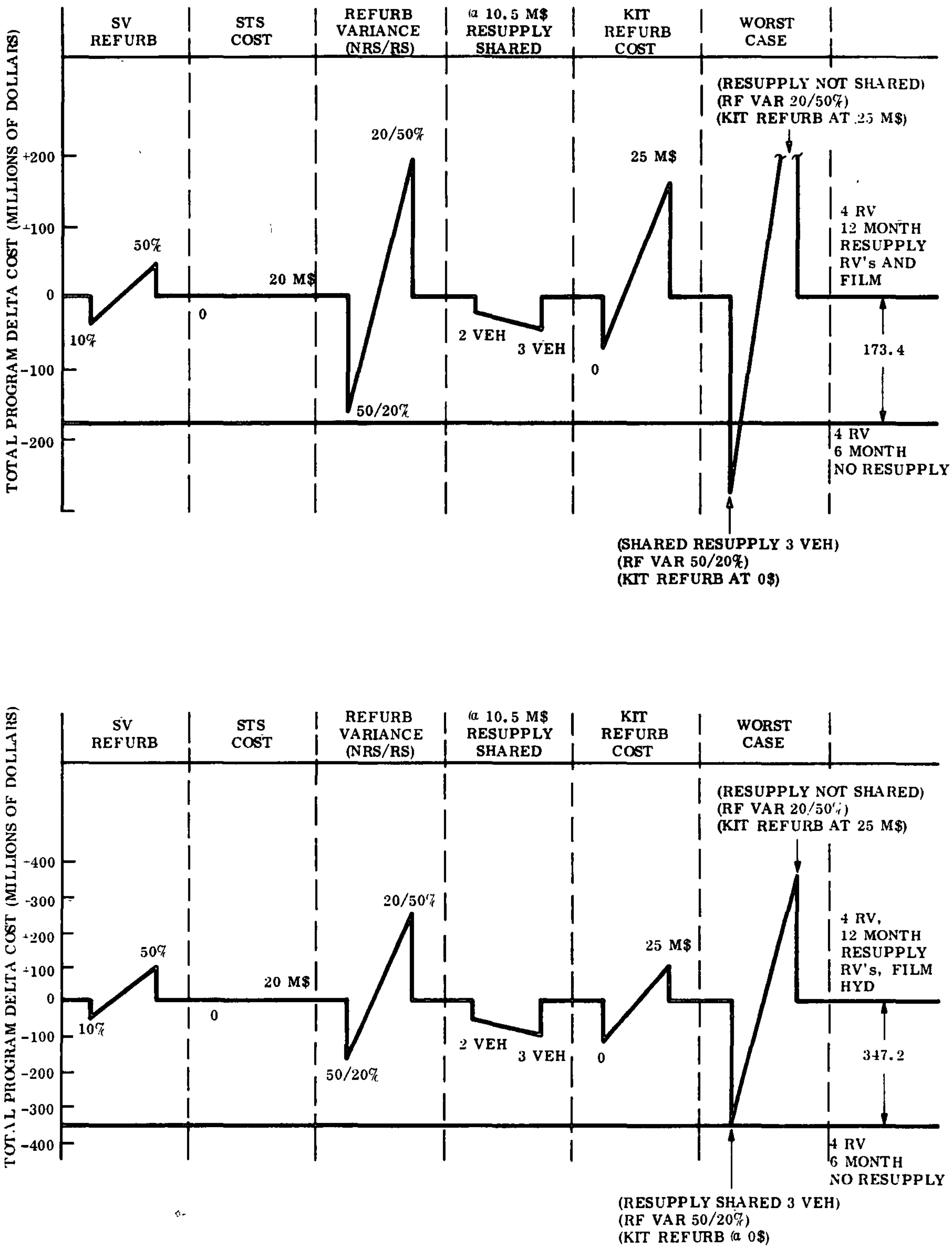
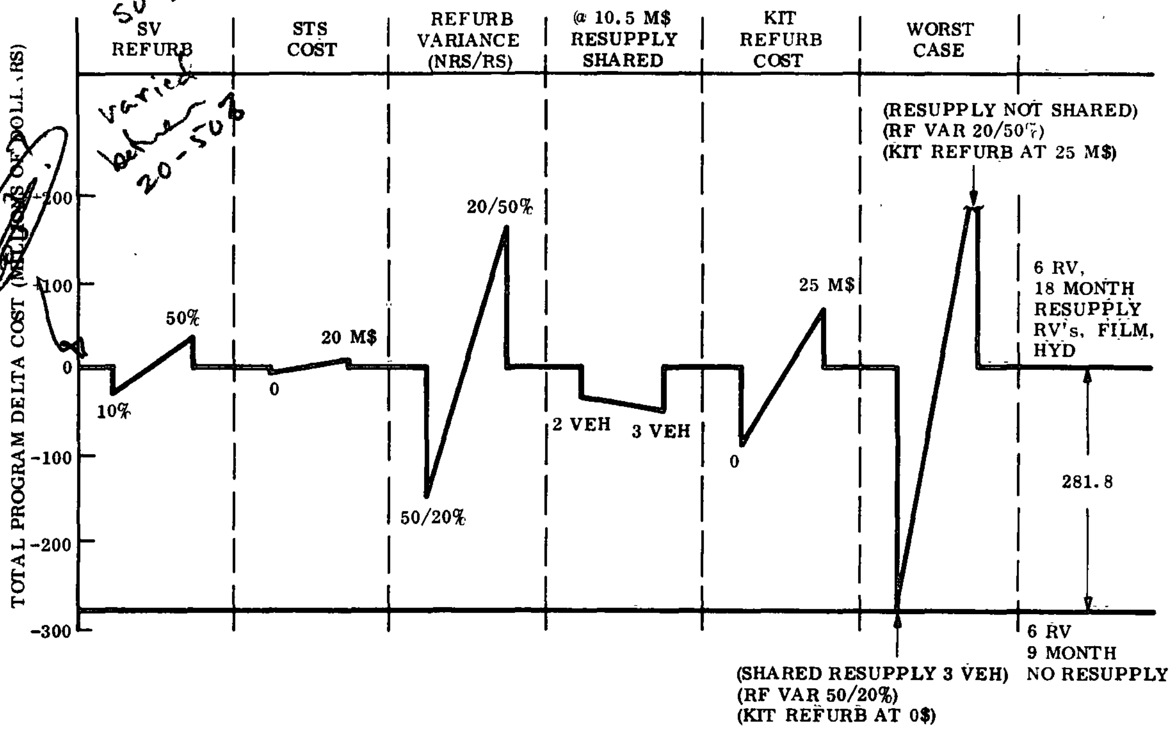


Fig. 4.3-3 Cost Sensitivity; 4-RV Configurations



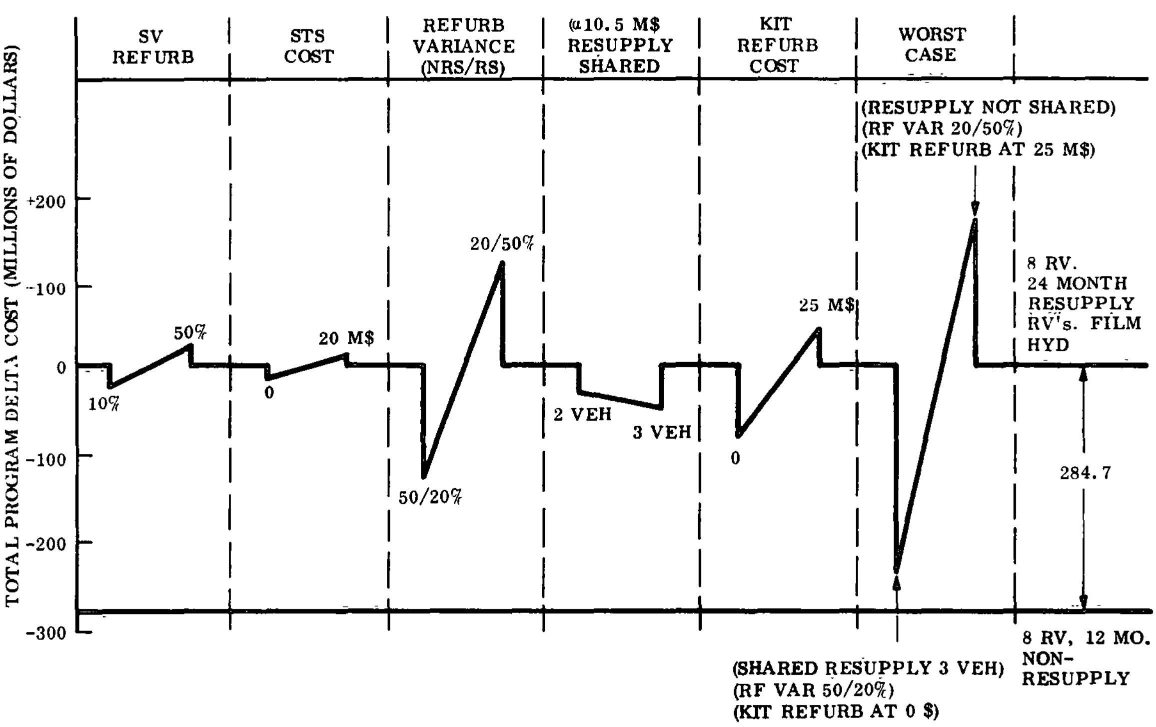


Fig. 4.3-3 (Cont) Cost Sensitivity; 6-RV and 8-RV Configurations

For the cost estimates, SV refurbishment was assumed to be 20 percent of the initial vehicle cost. Camera costs were not included in this estimate but were handled independently, based on inputs received from Perkin-Elmer. The value of 20 percent for a primarily electronic vehicle was derived from data developed by LMSC in its payload effects study for NASA. The refurbishment cost of 20 percent was taken as an average over 9 refurbishments on an 83 percent learning curve. Perkin-Elmer estimated their refurbishment costs at 50 percent with no learning curve adjustment. In this analysis the refurbishment factor of 20 percent, as applied to the vehicle and resupply kit (in the case of a resupply configuration), was varied from 10 percent to 50 percent. The effect of this on the delta cost between the resupply and non-resupply configuration is shown in the "SV Refurb" column. The variation of the refurbishment factor over a wide range has a relatively minor effect on the total cost differential between the two configurations.

Bull!

The cost of an STS launch was ground ruled at $10.5 million, including amortized learning curve effects. The effect of varying STS launch cost can be seen only when there is a different number of launches between configurations being compared. Thus, a change in launch cost has no effect on the delta cost between configurations if both configurations require the same number of STS launches.

The "Refurbishment Variance" depicts delta cost, assuming that one configuration cost 50 percent to refurbish and the other cost 20 percent. As could be expected, the delta cost is quite sensitive to this condition. For example, if the non-resupply (NRS) configuration refurbishment cost is 50 percent and the resupply (RS) configuration costs 20 percent to refurbish, the total cost between the two configurations is significantly reduced. However, this condition seems rather unlikely since it is expected that refurbishment costs for each configuration would, regardless of the final number, be nearly the same from configuration to configuration.

? why? Based on how many uses before SV return?

The cost benefits from sharing resupply launch costs with other programs were investigated. The only launch costs to be shared are the resupply launches since an SV nearly fills the payload bay and approaches the weight limitations. Incidentally, most resupply kits would also nearly fill the bay and approach the weight limits. The analysis shows that even if sharing were physically possible, the economic benefits are relatively slight.

So, what.

The impact of the resupply kit refurbishment cost was investigated. The analysis shows that even if the cost to refurbish the resupply kit were zero, the resupply configuration still does not become economically feasible.

The combination of the lower peaks results in the worst-case plot, which on some configurations reduces the delta to zero or less. However, since this worst case really assumes the additive effects of three worst cases, it is considered a highly unlikely situation. In addition, a delta cost of zero or less does not eliminate the high one-time costs required for development of a resupply capability.

4.3.6 Conclusions

A review of the cost estimates clearly shows that the resupply configurations are not cost-effective; therefore, a non-resupply configuration was selected for further detailed study.

The total costs of the non-resupply configurations are so close that a choice based on cost alone is not clear, but it is evident that longer life configurations have lower recurring costs.

4.3.7 Verification of Approach

At the conclusion of this study, the cost of the LMSC portion of the 6-RV configuration was estimated as described in Section 9. After making adjustments for differences in ground rules (such as earlier program go-ahead, no learning curve applied, and second block buy) and correcting to current year dollars, the spacecraft and integration (LMSC) portion of the top-down estimate was compared with the bottom-up estimate. The two values agreed within one million dollars of each other, which provides additional verification of the top-down approach.

4.4 CONFIGURATION SELECTION

As stated in the preceding section, it was determined that the longest-life vehicle configuration would minimize recurring costs by reducing, over a 10-year period, the number of STS launches and SV refurbishments. This section presents the SV/STS mission analysis that was conducted to select such a vehicle configuration for further subsystem design and costing.

4.4.1 Ground Rules

As a result of coordination meetings with the Customer some initial ground rules and guidelines were updated:

- STS can deliver a 36,400-lb SV into a 50 x 100 nm sun-synchronous orbit.
- The STS weight includes 250 ft/sec ΔV OMS not chargeable to the SV. This 250 ft/sec ΔV OMS includes that ΔV required to deorbit from a 100 nm circular orbit.
- The STS weight includes 3900 lb of RCS propellant not chargeable to the SV.
- 1800 lb of additional RCS is required to effect a rendezvous. This weight is chargeable to the SV.
- Sufficient propellant is included within the RCS for 7 days of STS drag make-up at altitudes equal to or greater than 100 nm circular.

4.4.2 Analysis

Trade-off
Life - vs, injection

As seen in Fig. 4.4-1, there is essentially no orbital lifetime ~~for a~~ at vehicle in a 50 x 100 nm orbit. Therefore, the STS must place the SV into a higher lifetime orbit for deployment. The existing Hexagon program requirement of 8 revs of tumbling life at injection was taken as the requirement for this study. To meet this requirement, the SV must be deployed in at least an 80 x 100 nm orbit. The STS must therefore carry additional propellant, chargeable to the SV weight budget, to achieve this orbit from the 50 x 100 nm injection orbit.

Can this be lower

could some other altitude be looked at.

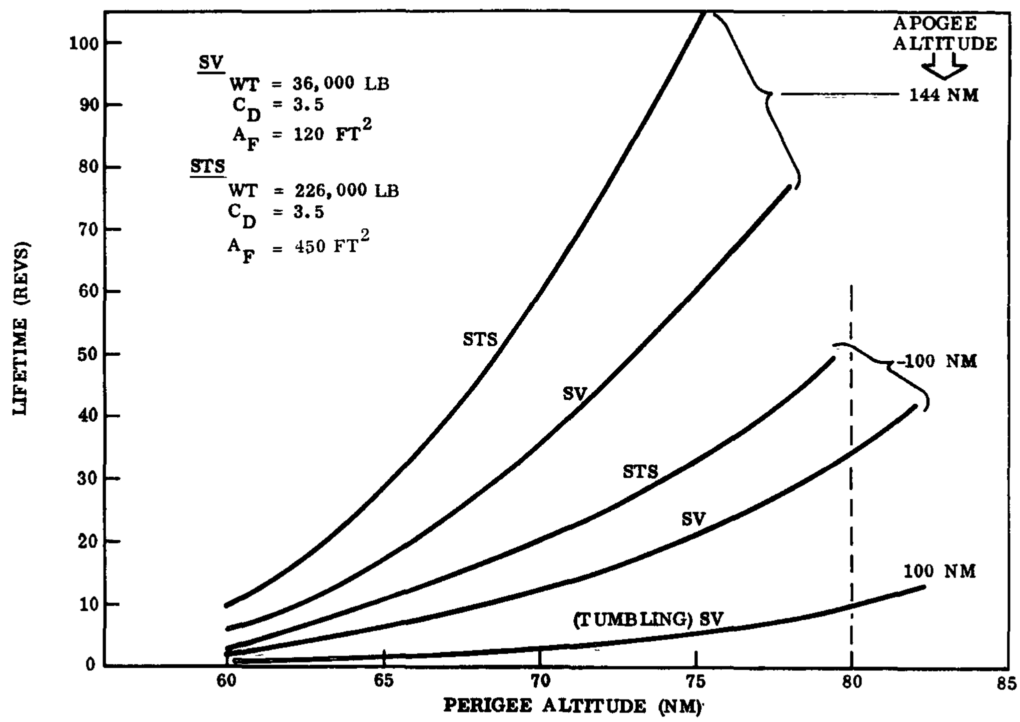


Fig. 4.4-1 Drag Lifetime vs. Orbit

250
16
5 | 1250

As shown in Fig. 4.4-2, the STS requires approximately 5 times the propellant weight required by the SV to achieve the same orbit from a 50 x 100 nm injection orbit. Therefore, the use of the STS for maneuvers will be kept to an absolute minimum. As shown, approximately 1250 lb of OMS propellant is required to achieve the 80 x 100 nm deployment orbit. Additional propellant required to place the SV in the chosen operational orbit from the deployment orbit will be allocated to the SV propellant weight.

Since all launches (after the initial one) will require rendezvous with the expended SV, some consideration must be given to the timing problems involved in such a rendezvous. Figure 4.4-3 depicts the effects of a delay in launch time of the STS. As an example, an STS launch delayed 10 minutes from nominal, without booster trajectory correction, would require approximately a 2.5 degree plane change to arrive at the orbit plane of the expended SV. This plane change would require a change in velocity of approximately 1000 ft/sec. For each ft/sec of velocity change,

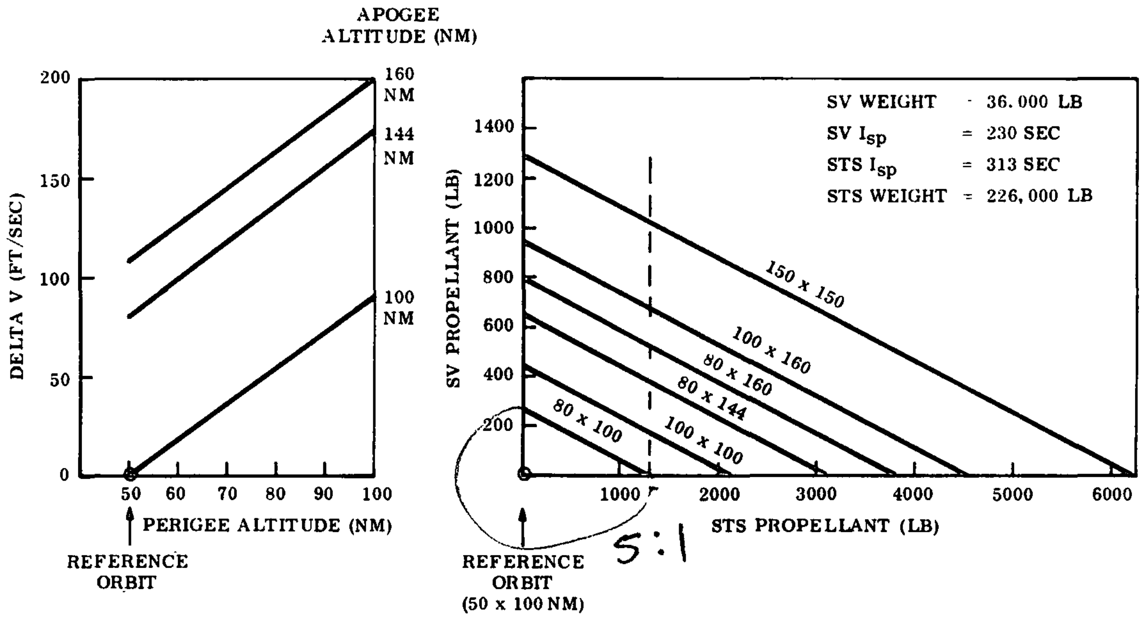


Fig. 4.4-2 Propellant Requirements for Orbit Changes

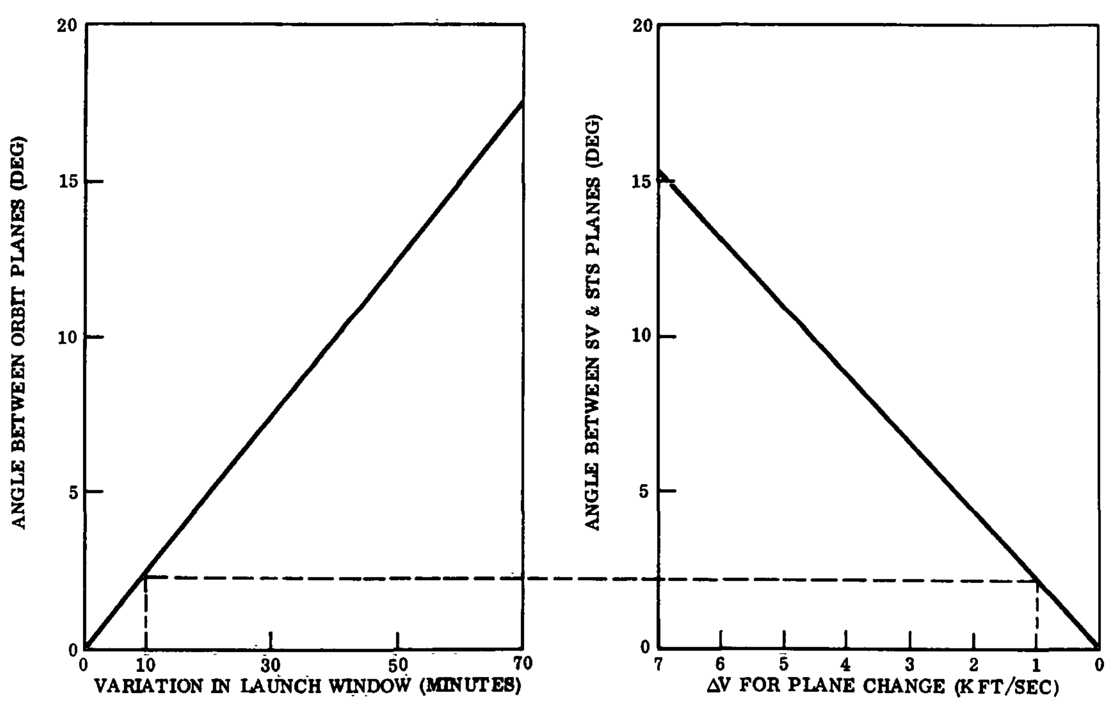


Fig. 4.4-3 Penalties Associated with Orbit Plane Variations

about 23 lb of STS propellant is required. Thus a 10-minute delay in launch would require about 23,000 lb of OMS propellant to correct the orbital plane. This weight penalty is prohibitive. It was therefore assumed that the launch window would be determined by the booster capability to correct the orbital plane during ascent without penalty to the SV weight.

In-plane orbital phase differences must also be considered to effect a rendezvous for retrieval of an SV. Assuming a maximum launch window of 10 minutes, and with the booster correcting the orbital plane, an in-plane phase difference of approximately 40 deg will exist between the Orbiter and SV. If two days are allowed to correct this phase error, using a Hohmann transfer, it will take approximately 125 lb of SV propellant (based upon RVs, film, and the majority of propellants expended) or 1400 lb of OMS propellant (Fig. 4.4-4). For minimum impact on payload weight, propellants required for orbital phase corrections of this type will be allocated to the SV propellant weight. Additionally, pre-phasing corrections due to launch delays of one or more days could be corrected by the SV, using this propellant allocation.

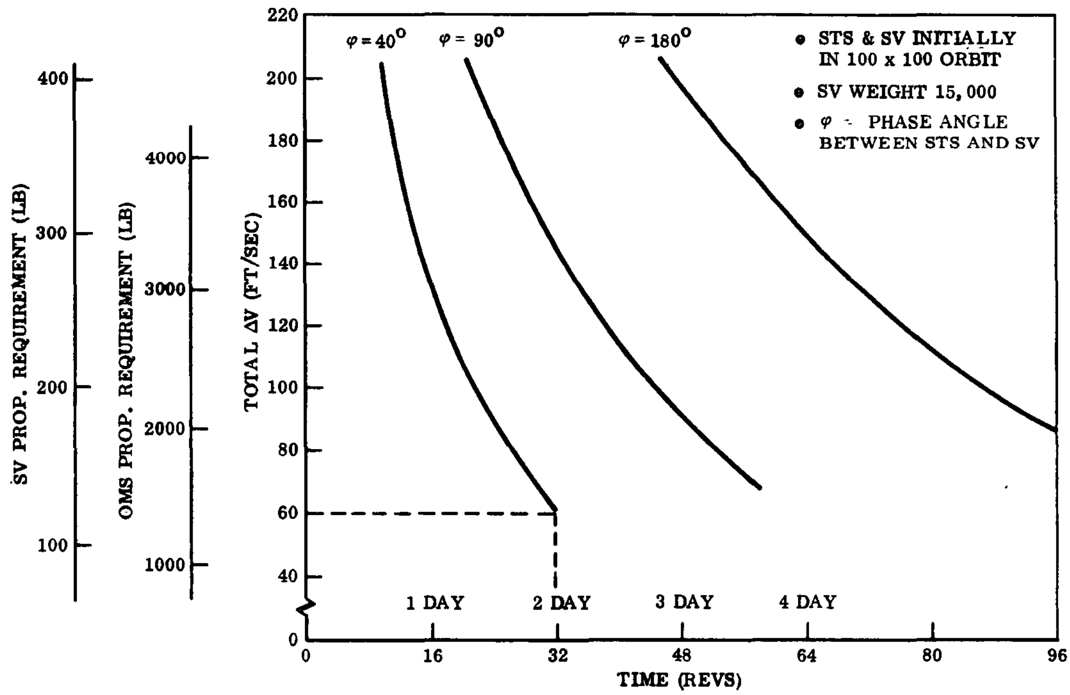


Fig. 4.4-4 Delta V Required for In-Plane Phasing Corrections

Table 4.4-1 shows that the effective weight available for the SV is 33,350 lb, when subtracting (1) the OMS propellant weight required to transfer to the deployment orbit, and (2) the STS RCS propellant weight required to effect a rendezvous with the expended SV for retrieval. To determine the actual weight available for all SV propellants, the weight available for the total SV is adjusted by subtracting the SV dry weight and Mission Specialist payload-provided equipment. An additional weight allowance for subsatellites and survivability aids was made at this time. The results show the amount of weight available for SV propellants and gases for each configuration.

In order to determine propellant available for SV mission orbit maintenance, the total weight available for propellants and gases, is adjusted by that propellant weight required for attitude control and performing orbit maneuvers. As shown in Table 4.4-2, an 8-RV configuration with subsatellites and survivability aids does not allow any weight for orbit maintenance propellants. A configuration without subsatellites and survivability aids was therefore considered. This allowed 1550 lb to be added to the total weight available for SV propellants and gases, which provided some weight margin for orbit maintenance propellants. Finally, the propellant usage on a daily basis was calculated for each configuration.

Knowledge of the daily propellant usage allows determination of the range of orbit parameters (Fig. 4.4-5) for each configuration. It was assumed that a photographic access repeat cycle greater than 9 days was not desirable. The orbit parameters derived from this figure are summarized in Table 4.4-3. Even though the 8-RV configuration does not include subsatellites or survivability aids, the lowest orbit possible is 140 nm circular with a repeat cycle of 9 days. To achieve lower repeat cycles, even higher altitudes are required. The 8-RV configuration was therefore rejected.

The 6-RV configuration provides the maximum on-orbit life for a reasonable orbit altitude and repeat cycle, and was therefore selected for subsystem conceptual design and costing. This configuration requires only a reasonable enlargement of the optics system to provide the required resolution at the higher altitude.

Table 4.4-1

WEIGHT BREAKDOWN

Weight Element (lb)	SV Configuration					
	4-RV		6-RV		8-RV	
STS Delivery Weight		36,400		36,400		36,400
STS Orbit Transfer to 80 x 100 nm	1250		1250		1250	
STS RCS for Rendexvous	1800		1800		1800	
Weight Available for SV		33,350		33,350		33,350
Mission Specialist Equipment	500		500		500	
Satellite Vehicle Gross Weight		32,850		32,850		32,850
Camera System	6125		7450		8650	
Takeups	960		1440		1920	
Recovery Vehicles	3900		5850		7800	
SV Equipment	2400		2550		2650	
Structure	6570		6570		6570	
Allowance for Subsats and Survivability Aids	1550		1550		1550	
Total Dry Weight		21,505		25,410		29,140
Weight Available for SV Propellants & Gases		11,345		7440		3710

NOTES:

- Insertion orbit is 50 x 100 nm, 96.4 deg inclination
- Delivery and rendezvous at 80 x 100 nm orbit
- SV will transfer to and from operational orbit to rendezvous orbit for retrieval
- No propellant allocated for plane changes
- Propellant required to perform orbit pre-phasing corrections will be allotted to SV propellant reserves.

Table 4.4-2

SATELLITE VEHICLE PROPELLANT ALLOCATIONS

Weight Element (lb)	4-RV	6-RV	8-RV	8-RV(4)
Total Weight Available for SV Propellants & Gases	11,345	7440	3710	5260
Transfer to Maximum Operational Orbit(1)	850	800	1500	815
Transfer to Rendezvous Orbit(2)	390	365	670	375
Phasing Corrections Due to STS Launch Window(3)	125	125	125	125
RCS and Pressurants	920	1000	840	940
Deboost	600	600	600	600
Total	2885	2890	3735	2855
Remaining Propellant for Orbit Maintenance	8460	4550	-25	2405
Max Propellant Usage (lb/day)	47.0	16.8	-	6.7

(1) Based on 32,850 lb
(2) Based on 15,000 lb, includes rendezvous pre-phasing
(3) Maximum phase angle assumed to be 40 deg, based on 10 minute launch window and booster correction of associated 2.5 deg plane change without penalty to payload. Maximum time for phasing corrections — 2 days
(4) Without survivability aids and subsatellites (1550 lb)

16.85

270/4550
270
1850
1620
2300
216
140

Table 4.4-3

NON-RESUPPLY — ORBIT CHARACTERISTICS SUMMARY

Configuration	Orbit Parameter Range (nm)	Repeat Cycle (days)
4-RV	82 x 135	4
	85 x 190	9
6-RV	110 x 125	5
	115 x 163	9
8-RV*	140 x 140	9
	200 x 200	5

like this one

*Without survivability aids and subsatellite

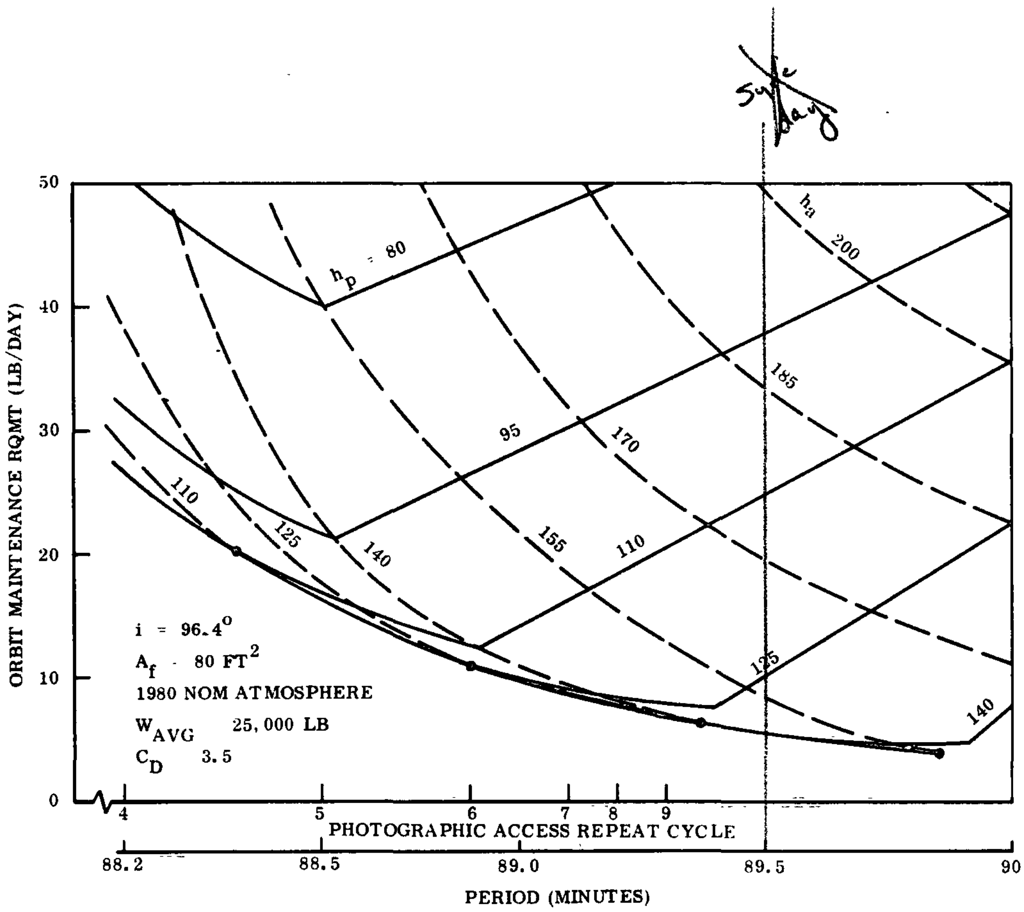


Fig. 4.4-5 Propellant Requirements vs. Orbit Parameters

4.5 SV/STS OPERATIONS

Autonomous SV/STS operation during ascent, deployment, and retrieval was chosen to minimize dependence on the Satellite Control Facility (SCF) and to allow direct control of SV functions by the STS crew. This approach was taken to maximize operational flexibility and crew safety due to the relatively infrequent and brief tracking station contacts.

During normal operations, when the SV is in or near the STS, the STS will monitor SV safety, status, and performance data, and will have full command and control of SV abort, test, and conditioning functions. During this time the SCF will operate primarily in a backup mode. The SCF can verify the SV command system memory loads if desired. If an SV problem occurs, the SCF can analyze SV real-time or recorder playback data and recommend corrective action.

Pre-Launch. Control over pre-launch testing and monitoring of the SV will be by computers located at the factory and connected by a data link to WTR. The STS will perform compatibility tests with the SV to verify that interfaces are properly connected and that the software routines are working properly. While in the STS payload bay, communication with the SV will be via equipment located in the STS or through the STS skin umbilical. Prior to the STS skin umbilical removal, the SV abort sequences will be loaded into the SV command systems and the SV clocks held. The abort sequences will then be verified by AGE. Subsequent to umbilical removal, SV status will be monitored via the 16 KBPS telemetry interleaved with the STS telemetry. Monitoring will be by both STS equipment (Mission Specialist Station) and by AGE. The ascent, deployment, and retrieval concepts are depicted in Fig. 4.5-1 and described below.

new guideline given.

Launch. During a normal launch the STS will not be required to exercise any command control over the SV. In the event of an abort, the STS will manually initiate the preloaded SV abort sequences. These commands, previously stored in the command system memory, will perform SV safing functions such as propellant dumping and pneumatic depressurization. Status of these abort activities will be displayed to the STS crew at the Mission Specialist Station.

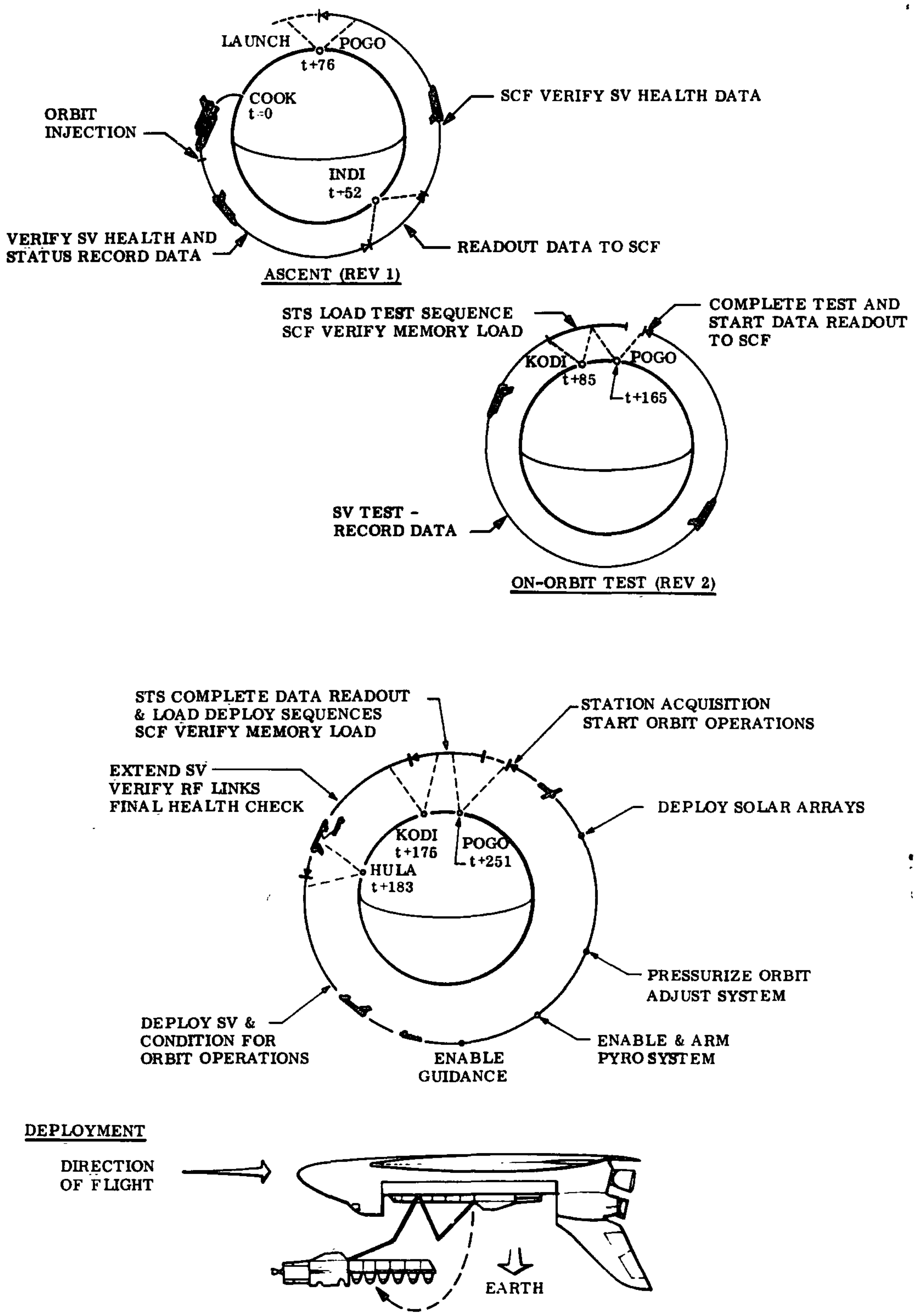


Fig. 4.5-1 SV/STS Deployment Operations

On-Orbit Test Concept. The capability to verify that the SV is functional, following exposure to the ascent environment and before committing the SV to orbital operations, is possible in the STS mode of operation. This concept of operation was evaluated to determine the extent of SV testing that should be conducted prior to deployment of the SV by the STS. The factors weighed included (1) impact on the STS hardware, software, and personnel; (2) time required to perform extensive testing; (3) impact on SV design; and (4) minimum tests required to verify safety of the STS and crew.

To perform on-orbit tests comparable to the ground SV systems test, extensive computer, software, and test time would be required. Considering this, it was decided that a comprehensive functional test of the SV prior to deployment was not practical. However, certain tests of the SV prior to deployment are considered mandatory to ensure safety of the STS and crew during SV deployment and to verify that the SV is capable of retrieval. It is also highly desirable to verify that the SCF can communicate with the SV prior to actual SV deployment. The STS may remain in a standby orbit for several hours after deployment and then, if a critical failure is detected in the initial SV orbit operations and engineering passes, the SV could be retrieved and returned to the ground for repair. If such a decision were made, another STS would be required to retrieve the expended SV that the first STS had been intended to retrieve.

A limited performance evaluation of mission-critical hardware is also desirable. This test, prior to SV deployment, will provide a high confidence that the SV is operational, and will reduce the risks of encountering a mission-critical failure after deployment. The on-orbit test objectives are:

- Verify that the SV will not endanger STS during deployment operations
- Verify that the SV communications system performance is adequate for SCF station acquisition and control
- Verify that the SV payload and subsystems are operational
- Verify that the SV is capable of retrieval.

A summary of the pre-deployment test by subsystem, as currently envisioned, is shown below.

How about only go/no-go info.

Subsystem	Test Objective
Electrical Distribution and Power	• Verify proper operation during other subsystem operation • Primary and Lifeboat bus switching
Pyro	• Pyro bus switching
Propulsion	• Pressures and temperature within specification values • Isolation, RCS & OAS valve configuration
TT&C	• Transponder operation (performed after SV extension on RMS) • Cmd transmission acceptance of primary, redundant, and backup command systems • Clock status • PCM system operation • Cmd readout capability of primary, redundant, and backup command systems
Guidance	• Gyro rate output (compare primary and redundant) • H/S output (compare primary and redundant) (Performed after SV extension on RMS) • Integrator operation (compare primary and redundant) • Guidance mode status • Valve driver activity (compare primary and redunant)
Lifeboat Guidance	• None (unless primary or redundant guidance system has failure)
Camera System	• Pneumatic system press./temp within safety limits • Film transport verification test • Configuration status verification
RVs	• Pyro safe/arm status • Configuration status verification

Pre-Deployment. Most of the first rev will be devoted to STS post-ascent functions and monitoring of SV telemetry health data to ensure that all systems survived ascent. A data readout at INDI will allow the SCF to review the ascent data and confirm SV status at the next station contact. Following SCF confirmation of SV health data at POGO, the STS will load the SV command system with the pre-deployment test sequences. Commanding will be from a payload-provided digital tape recorder at the Mission Specialist Station via the STS Payload Computer. These initial command loads can be confirmed by memory readout to POGO or KODI if desired. During all SV/STS orbit operations, an abort load will be available in one of the command system memories for immediate activation by the STS if required.

During the second rev, the pre-deployment tests are conducted and the telemetry data limit-checked by the STS. Due to the magnitude of the data required, two 16 KBPS telemetry formats will be selected (automatically) as part of the checkout sequence. During the SCF tracking station pass (POGO), at the completion of these tests, the STS tape recorder containing the 16 KBPS telemetry data will be read out to the SCF for backup analysis. The SCF can then assist the STS with corrective action in the event a problem is disclosed during the test.

safety only

primary

Deployment. If no problems have occurred, the STS will load the SV command system with the deployment sequences. The STS will then attach the Remote Manipulator System (RMS) and extend the SV. Final health tests can then be conducted to verify proper horizon sensor response and RF link operation (with the STS and SCF) prior to release. After release the STS will move a safe distance away from the SV before conditioning the SV for orbit operations. The SV systems will then be enabled and verified under STS control. At the first tracking station contact following these activities, the STS can turn SV control over to the SCF for orbit operations. The deployment attitude shown is desired (but not mandatory) so that the SV is released in the normal operational attitude. Prior to this time the STS should be in an attitude to provide the proper thermal environment for the SV.

change in guidelines

Retrieval. Once the Orbiter has completed deployment of the new SV and is ready for retrieval of the expended SV, the SCF will transfer the expended SV to the rendezvous orbit and condition it for retrieval as shown in Fig. 4.5-2. The STS will then complete terminal rendezvous with the SV, attach the manipulator, and stow the SV in the payload bay. The SV/STS umbilicals will be reconnected, residual propellants dumped, and the SV conditioned for reentry. The FM telemetry system should remain on during reentry to monitor the environmental exposure to which the SV is subjected. This data can be recorded and subsequently analyzed to ensure that specification values have not been exceeded and that no special refurbishment operations are required.

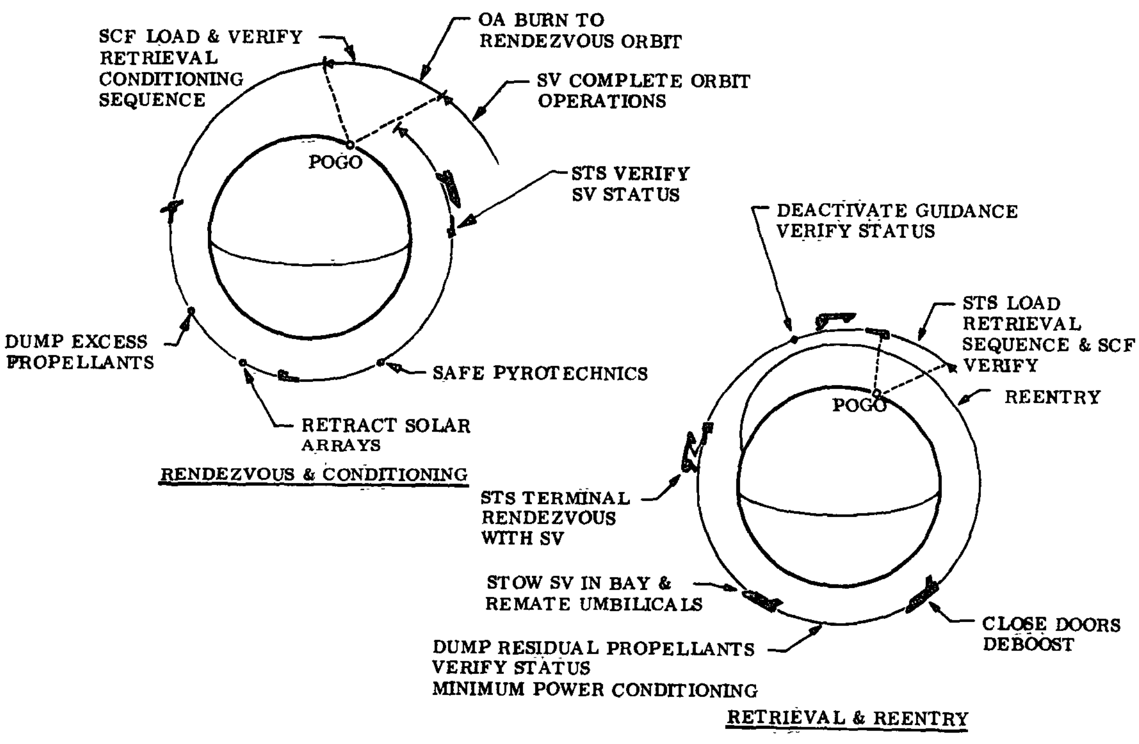


Fig. 4.5-2 SV/STS Retrieval Operations

Section 5
SATELLITE VEHICLE DESIGN

This section describes a Hexagon Satellite Vehicle specifically designed for use with the STS. Included is a description of the interfaces between the SV and STS and the SV-provided equipment installed in the STS.

The SV design described herein satisfies the objectives of this study; however, this design may not be the best design for transition of the Hexagon program from the SLV to the STS mode of operations. Additional studies are required to define an SV design which can best transition from the existing configuration to an STS-compatible design.

5.1 SYSTEMS DESCRIPTION

This section provides an overall description of the Satellite Vehicle selected in Section 4.4 for conceptual design and costing. A detail discussion of each subsystem is presented in subsequent paragraphs.

5.1.1 System Configuration

The complete Satellite Vehicle design is shown in Fig. 5.1-1. As shown the overall length of the vehicle is 56 ft, 4 in., with a frontal area of 100 sq. ft. All film takeups, including the sixth takeup, are installed within RVs to allow individual reentry and recovery of exposed film. This capability was retained for the STS-compatible configuration to allow film recovery at the desired time, regardless of STS availability or in the event the SV could not be retrieved due to an SV or STS problem. The propellant tank and supply unit are located and positioned as shown to conserve overall vehicle length. The frontal profile of the vehicle is designed for minimum area.

The SV structure is entirely different from that currently planned for Block III. The primary considerations for the structural design were compatibility with the STS attachments and loads, and the provision for good access to equipment during the SV refurbishment operations. The STS attachment structure shown is integral to the basic SV structure and is also used for ground handling support points. The truss structure provides excellent access during refurbishment. The thermal design concept is similar to that currently used on Block III except that here the SCS thermal design can be simplified by using lightweight shields since ascent heating protection is no longer required.

The components have been relocated on several of the SCS electrical/electronic modules so that most equipment is installed on external surfaces to provide easy access during refurbishment operations. The attitude reference module and one battery module are the only exceptions. The new module designations and their assigned equipment are shown in Table 5.1-1.

In addition to these modules; the propulsion system and reaction control components, including the propulsion system J-box, and the solar array modules are located in the SCS.

Three umbilicals are provided for ground test, servicing, and interfacing with the STS during flight operations. Umbilical No. 1 contains hardline instrumentation, controls, and battery charging; Umbilical No. 2 contains coaxial cables carrying serial-digital command and telemetry data; and Umbilical No. 3 contains the propellant dump and battery cooling interfaces.

The Forward Assembly contains the camera system, film supply unit, recovery vehicles, and film path articulators, as well as the following electrical/electronic components:

- PCM remote units #4 and 5
- ECS remote decoders #1 and 2
- Backup command decoder
- Forward instrumentation J-box #1 and 2
- Forward power distribution J-box
- Temperature control electronics assembly
- Interface J-box
- Data interface unit

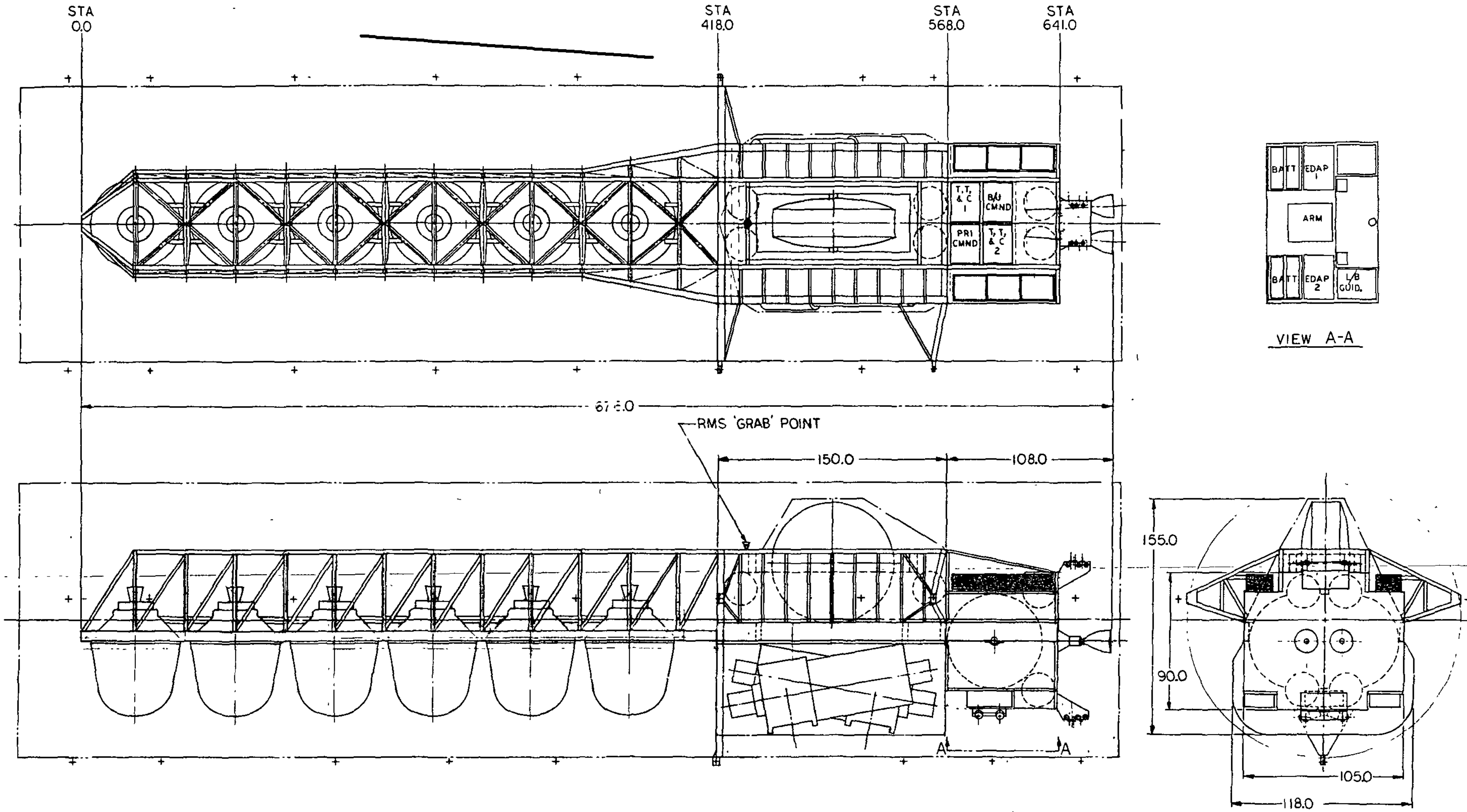


Fig. 5.1-1 Vehicle Design Drawing

Table 5.1-1

MODULE DESIGNATIONS

Module	Components Located on Module
Battery Module #1	2 secondary main bus batteries
Battery Module #2	2 secondary main bus batteries 1 secondary Lifeboat battery
Electrical Distribution & Power #1	Power distribution J-box Pyrotechnic J-box PCM remote unit #1
Electrical Distribution & Power #2	5 charge current controllers
Attitude Reference	Primary & redundant inertial reference assemblies Primary & redundant horizon sensor assemblies Primary & redundant flight control electronics
Tracking & Telemetry #1	Primary SGLS transponder Primary tape recorder Primary PCM master Type 1 control J-box PCM remote #2 Multicoupler
Tracking & Telemetry #2	Redundant SGLS transponder Redundant tape recorder Redundant PCM master Type 4 instrumentation J-box PCM remote #3 Multicoupler
Primary Command	Extended commanded system Type 2 control J-box
Lifeboat Guidance	Lifeboat electronics Lifeboat J-box Magnetometer Magnetometer electronics 3 rate gyros
Backup Command	Backup command receiver Lifeboat command receiver Backup command computer Caution & warning instrumentation J-box FM telemeter

All camera system components, except the film path pneumatics and supply unit, were modularized onto a removable platform, thus providing a simple interface for camera system removal and reinstallation during the refurbishment operations. The camera system optics have been increased in size to provide the required ground resolution of 2.27 ft at altitudes of 120 nm.

The existing McDonnell/Douglas RVs were employed during this study. The following modifications should be incorporated to meet system safety requirements and to provide ease of refurbishment:

- Add an electromechanical safe/arm device for retro rocket and other hazardous pyrotechnics.
- Install a reusable SV/RV electrical umbilical instead of the present guillotine design.
- Modify RVs as necessary to enhance refurbishability.

5.1.2 Subsystem Description

Most of the vehicle subsystems were not significantly modified from the Block III configuration, but the propulsion system, solar array modules, and backup command system require extensive modifications to meet the safety and life requirements. A brief summary of the modifications required on each subsystem is provided below:

Attitude Control. The existing Block III attitude control subsystem can be used directly on the STS-compatible SV design. The cross-strapping provided in the Block III design provides the required redundancy for the reliability considerations of this design.

Propulsion. The orbit adjust/reaction control system required significant modification to meet the safety, life, and refurbishment requirements. The orbit adjust tank was increased in size to provide the required propellant capacity. The orbit adjust and reaction control system were integrated to conserve weight and to provide ease of refurbishment. A repressurization system was added to provide the capability for low pressure in the propellant tank during ascent and to minimize the tank size versus that required for a blowdown system. Propellant dump capability was incorporated to allow no problem on use cycle?

propellant dump during ascent aborts, on-orbit SV/STS operations, and retrieval. A redundant orbit adjust engine was added to provide increased reliability and redundancy during rendezvous maneuvers, and to eliminate the need for requalification of the existing engine. The Lifeboat reaction control system was modified to a hot gas configuration to conserve weight.

Other modifications were made to allow flushing and purging at the systems level and to provide ease of hardware replacement during refurbishment operations.

Power. The existing Block III main bus power system will meet the new power requirements because the deletion of the Mapping Camera Module more than offsets the increased power consumption of the larger camera and two additional RVs. The solar array modules were modified to provide retraction capability, with a pyrotechnic separation mechanism as backup for retrieval operations. A new solar cell currently under development was used in the solar array sizing.

To meet the life requirements, the Lifeboat primary battery was replaced with a secondary battery. The pyrotechnic bus primary batteries were deleted and backup power was provided from the Lifeboat bus.

The pyrotechnic J-box was modified because of the elimination of the shroud, booster, and horizon sensor fairing separation events and the addition of the propellant dump valve and solar array backup separate functions. In addition, remotely actuated safe/arm switching was provided to maintain safety during STS flight operations.

Tracking and Telemetry. Additional instrumentation is required primarily for caution and warning monitors and the two additional RVs. Extra slices were added to the existing remote units to meet these requirements. The master units were also modified to provide two 16 KBPS formats for STS compatibility during pre-deployment tests. Two formats are required because of the magnitude of data to be processed during SV checkout.

with no APSA Instrum. should be no problem to APSB use ixs commands.

An ECS remote decoder was added in the Forward Assembly to satisfy the command requirements of the additional RVs. The ECS was modified to provide secure real-time command capability and also to provide a secure word generator capability to meet the additional secure command requirements of the longer life vehicle, and to eliminate the need for ECS removal from the vehicle for secure word loading during refurbishment.

The backup command requirements have increased due to the abort and retrieval considerations and the addition of two RVs. The existing Block III backup command system (MCS) was replaced with a small general purpose computer to satisfy these requirements.

An FM telemeter was added to provide vibration data during ascent and reentry. Other modifications include the addition of line-drivers to drive telemetry signals to the STS, and a caution and warning instrumentation system to provide safety data to the STS during flight operations.

<u>Lifeboat.</u> The existing Block III lifeboat guidance system can be used essentially as is. The only changes required are a simple modification to the Lifeboat J-box to provide inertial operation during retrieval, and a modification of the thruster harness to provide compatibility with the hot gas thruster configuration.

5.1.3 Hardware Design Impact

A summary listing of the added and modified hardware discussed in the preceding paragraphs is shown in Table 5.1-2. As noted, much of the added hardware is of existing design.

Table 5.1-2

HARDWARE DESIGN IMPACT

New and/or Added Hardware	Modified Hardware: Major Modification	Modified Hardware: Minor Modification
Structure	Pyrotechnic J-box	PCM Master Units
Harnesses	Type 4 Inst J-box	PCM Remote Units
Plumbing	Propulsion J-box	Tape Recorders
OA & LB Propellant Management Devices	OA Tank	Type 1 Control J-box
*Additional OA Engine	Antennas	Pwr Dist J-box
*LB & Repressurization Tanks	Temperature Control Electronics Assembly	LB J-box
*LB Hot Gas Thrusters		Backup Command Receivers
*Pyro-Operated Valves		ECS
*Additional Fill Valves		Type 2 Control J-box
Solar Array Deploy/Retract & Separation Mechanisms		Fwd Pwr J-box
*LB Battery & Charge Controller		Fwd Inst J-box
Thermal Control Surfaces		Interface J-box
C&W Instrumentation J-Box		
Backup Command System & Remote Decoder		
*ECS Remote Decoder		
*Telemeter Unit		
*C&W Instrumentation		
Umbilicals		

*Denotes added hardware of existing design

5.2 STS INTERFACES

The following paragraphs present a summary of the SV/STS interfaces. Figure 5.2-1 shows the interface concept.

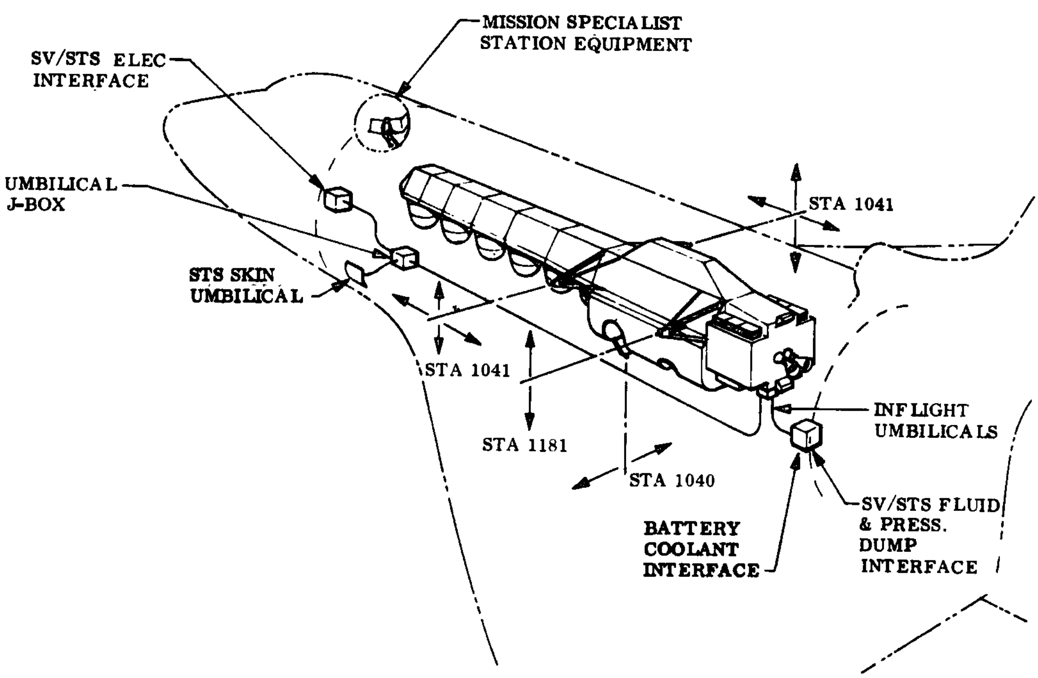


Fig. 5.2-1 SV/STS Interfaces

5.2.1 Mechanical and Environmental

The SV is mounted in a statically determinate fashion at STS stations 1040, 1041, and 1181. The SV is mounted with the +X axis toward the front of the STS and the +Z axis facing the payload bay floor. The STS-furnished remote manipulator system (RMS) will be used for SV deployment and retrieval.

The payload bay repressurization and cleanliness during reentry was not investigated during this study. It was assumed that payload requirements would be met by STS-provided equipment, as most payloads will require a clean, dry, environment during reentry to avoid extensive refurbishment. Another problem area not investigated is the effect of the STS RCS thruster plumes on the SV during deployment and rendezvous maneuvers. This problem is also common with all other payloads.

No constraints were identified to LMSC on the STS attitude with the payload bay doors opened. It was therefore assumed that the STS could fly in any attitude required to provide the correct thermal environment for the SV. These mechanical and environmental interfaces are discussed in detail in paragraph 5.3.

5.2.2 Umbilicals

Two electrical and one fluid umbilical are required and will be provided by the SV. These umbilicals will be retracted (by command from the Mission Specialist Station) prior to SV deployment and engaged following SV retrieval. The SV electrical umbilicals will provide command and telemetry information to the Mission Specialist Station and the STS skin umbilical via an umbilical J-box. Both the skin umbilical and umbilical J-box will be SV-provided. The SV fluid umbilical will provide the path necessary to dump SV propellant while in the STS payload bay, and will also provide battery coolants while on the launch pad. Although the capability to dump propellants is not now provided in the STS, it is recommended that such provisions be incorporated.

5.2.3 Electrical

The STS is capable of supplying 1 KW at 24 to 30.5 VDC. Since the SV requires up to 34 VDC, DC-DC converters will be provided in the umbilical J-box to up-convert the STS voltage. If higher voltage is available, the DC-DC converters will not be required.

The interface equipment which will be provided by the SV is shown in Fig. 5.2-2. In addition to containing the DC-DC converters, the umbilical J-box will provide the necessary signal conditioning for the instrumentation (such as umbilical status, hydrazine leak detector, etc.) located in the payload bay. The J-box will also route signals to the SV-provided STS skin umbilical for use during pre-launch checkout by the AGE. A command decoder sub-unit in the umbilical J-box will convert the output command signals of the STS payload monitoring computer to signals compatible with the SV. One such signal is the serial-digital commands to the SV command system via the SV coax umbilical (Umbilical No. 2). The other type of signal generated by the command decoder sub-unit is discrete signals which will control individual events within the SV (such as clock control, power on-off, etc.) and events within the payload bay (such as DC-DC converter on-off, umbilical retract, etc.).

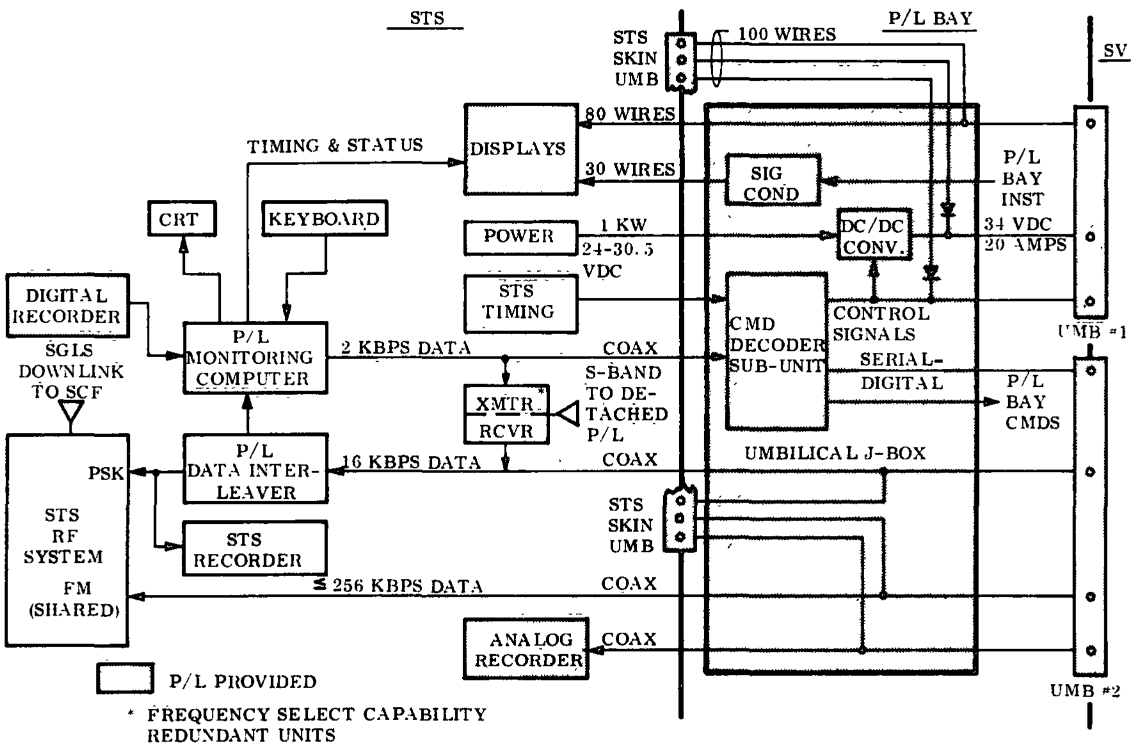


Fig. 5.2-2 SV/STS EDAP and TT&C Interfaces

Two SV electrical umbilicals have been provided. Umbilical No. 1 handles all discrete commands, discrete telemetry, and power. Umbilical No. 2 handles the coax inputs and outputs of the SV. These two umbilicals are configured such that the loss of either one of them (e.g., by inadvertent retraction) will not result in complete loss of control of the SV. That is; Umbilical No. 1 contains 70 hardwired caution and warning monitors with 10 selected returns and the control signals necessary to safe the SV, and Umbilical No. 2 contains the caution and warning monitors on the 16 KBPS telemetry data and SV commanding (via the serial-digital signal) to control the SV.

During the time when this SV is detached from the STS (e.g., post-deployment or pre-retrieval), the STS RF system will be used for transmission of commands and receipt of telemetry. It is assumed that the STS will have redundant transmitters with frequency select capability to enable commanding of the backup command system if required.

Telemetry. The vibration data (ascent and reentry) will be recorded on an SV-provided analog recorder at the Mission Specialist Station.

The STS FM Analog S-band link can be used to relay up to 256 KBPS data from the SV. However, in the event that additional data is required for SCF analysis, the maximum data rate will be 128 KBPS for the Hexagon vehicles. Normal operation will require the SV telemetry at a 16 KBPS rate to be interleaved with the STS telemetry for on-board recording and subsequent transmission to the ground. The 16 KBPS data will also be presented to the STS payload monitoring computer for CRT display of the caution and warning monitors (as backup to the hardwired display) and for limit-checking during the SV pre-deployment tests. The magnitude of the data to be processed for vehicle and payload checkout requires two 16 KBPS formats. If higher bit rates (e.g., 48 KBPS) were available, these new formats would not be required. Also, due to the 16 KBPS format constraint, the other PCM data formats (48, 64, 128 KBPS) cannot be tested by the STS system. The caution and warning monitors are discussed in more detail in Section 5.7.

Commanding. Commanding will be from an SV-provided digital recorder through the STS payload monitoring computer. For SV tests, individual cassette tapes will be placed in the tape recorder and the payload monitoring computer will be instructed to process the tape, send the necessary command load to the SV command system, and limit-check the appropriate monitors on the 16 KBPS telemetry from the SV. Any number of command routines can be handled in this manner. Individual commands (such as Start SV Abort) will be sent from the keyboard at the Mission Specialist Station.

<u>STS Payload Computer Utilization.</u> Sizing of the STS payload monitoring computer was based upon processing and displaying 70 caution and warning monitors and on processing, limit-checking, and displaying vehicle parameters during the SV checkout mode. This preliminary analysis (shown below) indicates that the amount of processing time that has been allocated for payload functions is not sufficient.

	Allocated	Required	
		Caution & Warning	Checkout
Size – (16-bit words)	5000	300	1400
Speed – (additions/second)	8600	1400	28000

If time cannot be made available in the computer, then the SV must provide its own computer in the Mission Specialist Station.

<u>Realtime Displays.</u> The Mission Specialist Station will contain the displays and controls necessary for operation of the SV. This concept is shown in Fig. 5.2-3.

Time of the STS and SV clocks will be displayed at the console. Meters will be provided to determine such things as voltage and pressure. Go/no-go lights will be provided to monitor the caution and warning instrumentation. Each caution and warning point will be hardwired to the Mission Specialist Station, where it will be level-detected. If the instrumentation point is within limits, the "go" light will be lighted; if the point is out of limits, the "no-go" light will be lighted; if the wire from the SV is broken, neither light will be lighted. In this case the Mission Specialist can select the applicable caution and warning monitor from the 16 KBPS telemetry data and display it on the CRT.

good

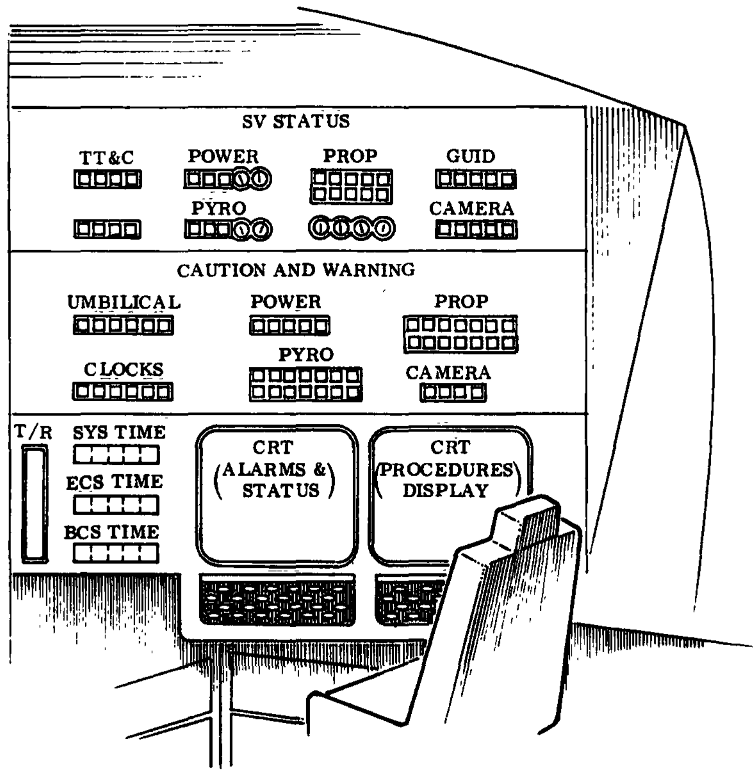


Fig. 5.2-3 Mission Specialist Station

5.3 STRUCTURAL/MECHANICAL SUBSYSTEM

This section describes the structural/mechanical subsystem and discusses the requirements; the loads, dynamics, and stress analysis; the thermal design; and the testing requirements. The mass properties of the SV are also provided.

5.3.1 Requirements

The STS structural/mechanical interface requirements for the SV were derived from information provided in JSC 07700, Vol. XIV, "Space Shuttle Program Payload Accommodations," Johnson Space Center, 4/13/73. The significant requirements are listed in Table 5.3-1. Those which have the greatest impact on the SV design are:

- The four-point statically determinate attachment to the STS, which does not provide X- and Z-axis reactions at the Y-axis attachment, and does not provide Y-axis reactions at the X- and Z-axis attachments
- High Y- and Z-axis acceleration during STS reentry
- Crash condition requirements
- Acoustic environment peak energy occurs at a significantly lower frequency than that experienced on current programs.

5.3.2 Structural/Mechanical Design

The primary concerns in the structural/mechanical design of the SV were: accommodation of the six RVs, increased film and propellant, enlarged optical bars, and adaptation of the vehicle to the STS mounting requirements. Several methods of mounting the SV in the STS were considered (see Fig. 5.3-1). Method (d) appeared to be impractical for the SV, and was therefore dropped from any further consideration. Methods (a), (b), and (c) do not present induced torsional and bending loads in the SV, and do not appear to provide any distinct advantages relative to each other. Hence, Method (a), which is identical to the one in the JSC 07700 document, was chosen as the mounting method. (If the STS can be designed to accommodate Y-axis loads on the door sills ($Z_o = 410$) and Z-axis loads on the keel ($Z_o = 307$) an SV mounting with simpler load distribution can be designed.) It should be

Table 5.3-1

STRUCTURAL/MECHANICAL REQUIREMENTS

Item	Requirement
Allowable Envelope	15 ft dia by 60 ft long
C.G. Envelope	$968 \leq X_o \leq 1121$, $314 \leq Z_o \leq 421$ ($W \leq 35,000$ lb)
Launch Vehicle Attachment	Four-point, statically determinate (see Fig. 5.3-1)
Loads	See Table 5.3-3
Quasi-Sinusoidal Vibration	Determined by booster and Orbiter engine shut-down transients; assume equal to or less than current boosters
Acoustic	145 dB overall (see Fig. 5.3-6)
Random Vibration (Components)	6.3 G_{RMS}; 0.1 g^2/cps, 20 – 200 Hz (See Fig. 5.3-7)
Pressure (Venting)	Not to exceed 1.0 psi burst and 0.5 crushing
Component Load Factors	≤10 lb: 15g; ≥ 200 lb: 8 g
Factor of Safety	Ultimate = 1.4 x limit.
Frequency Constraints	Ascent: no major resonances between 16 and 22 Hz Orbit: no major resonances below 1.5 Hz
Thermal Environment	Pre-launch +80 ±40°F Ascent +40 to +150°F Orbit +30 to +161°F (in cargo bay with doors closed) 70 nm minimum altitude, $i = 96.4^o$, $\beta = \pm 45^o$ Reentry and landing -100 to +200°F
Cleanliness	Fed Std 209A, Class 100,000
Humidity	0 to 50 percent RH

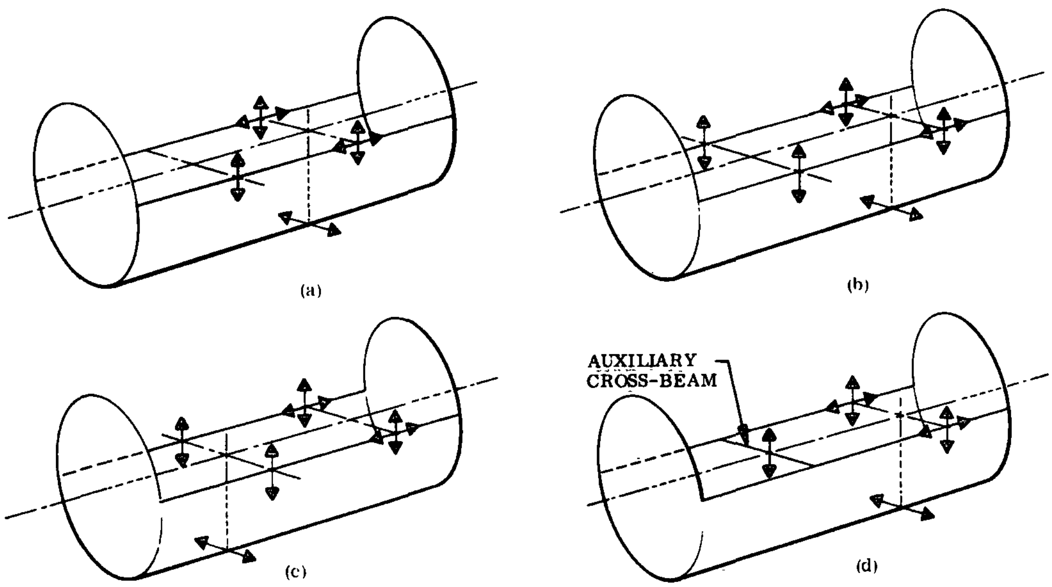


Fig. 5.3-1 Satellite Vehicle Support, Mounting Point and Reactions

noted that the mounting methods available at this time are predicated on rigid body behavior. However, since neither the SV nor the STS can provide this characteristic, Y-axis loads and displacements will occur at the STS door sills when lateral and vertical accelerations are applied to the vehicle. This area of concern cannot be resolved until the elastic behavior of the SV and STS is established.

The other key design problem was obtaining a vehicle short enough to fit in the payload bay. The major length reduction effort centered on the RV/camera system, which spans nearly 90 percent of the current vehicle length. LMSC, working with P-E, investigated several possible configurations of RV, supply unit, and optical bar locations. The arrangement chosen was a combination of in-line RVs for simplicity, and the supply over the optical bars for reduced length. The selection was based upon a reasonably short length, a film path very similar to the present demonstrated design, a high potential for facilitating initial assembly and subsequent refurbishment operations and, most important, a compatibility with the chosen

STS mounting scheme. Not only was a direct and simple load path for the primary STS attachment possible, but the center of gravity of the SV for all STS-mated conditions was very close to this same location. Thus, yaw-bending induced loads in the primary attachments were minimized with benefits to both the STS and SV. The final design is shown in detail in Fig. 5.1-1.

Figure 5.3-2 shows the SV installed in the STS payload bay. As shown, the Y-axis loads are carried at STS station $X_o = 1040$, the X-axis loads at $X_o = 1041$, and Z-axis loads at stations $X_o = 1041$ and $X_o = 1181$, with the support at $X_o = 1181$ located on one side of the vehicle only. All latch/unlatch mechanisms, including guides, are assumed to be provided by the STS.

The open truss structure (Fig. 5.3-3) supporting the RVs was chosen to provide ease of access to the film path, RV motor, and pyrotechnics for inspection and refurbishment, and to provide a lightweight, low-cost structure. The two large box beams in the central region were selected as the design approach to provide the support for the camera system, and to provide the torsional strength and stiffness required to react STS mounting loads. The box-like structure used for the SCS was chosen to provide sufficient external mounting space for subsystem components so that refurbishment can be easily accomplished. All primary structure is aluminum alloy. Lighter weight materials such as magnesium were not used because of the potential hazards to the STS and crew.

The propellant tank uses the domes from the existing orbit adjust tank, with an added cylindrical section to provide the required volume. Propellant management and slosh baffling devices will be included in the tank design. The new camera system mechanical interfaces are described in paragraph 5.9.

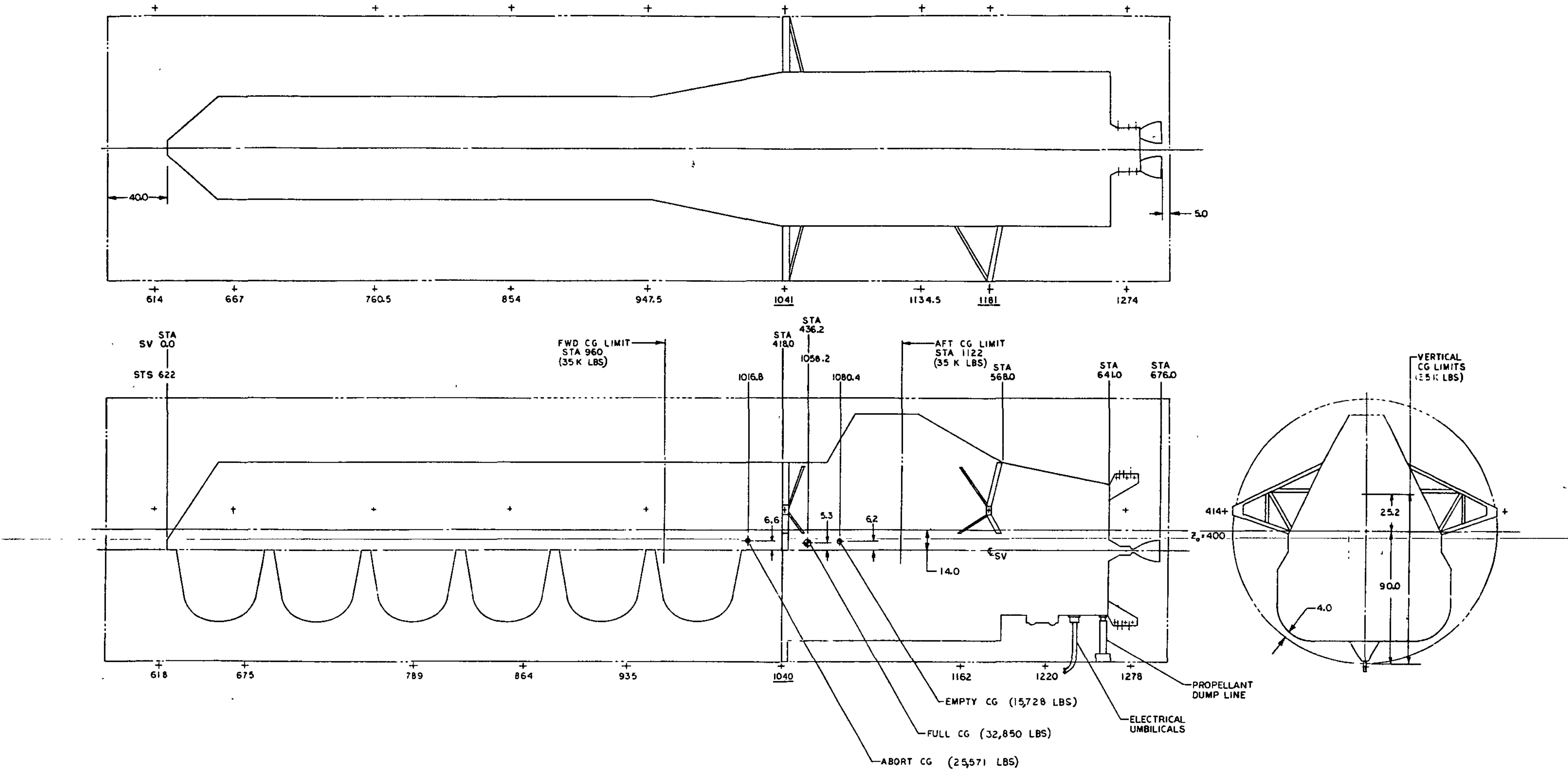


Fig. 5.3-2 SV/STS Installation and C.G.s

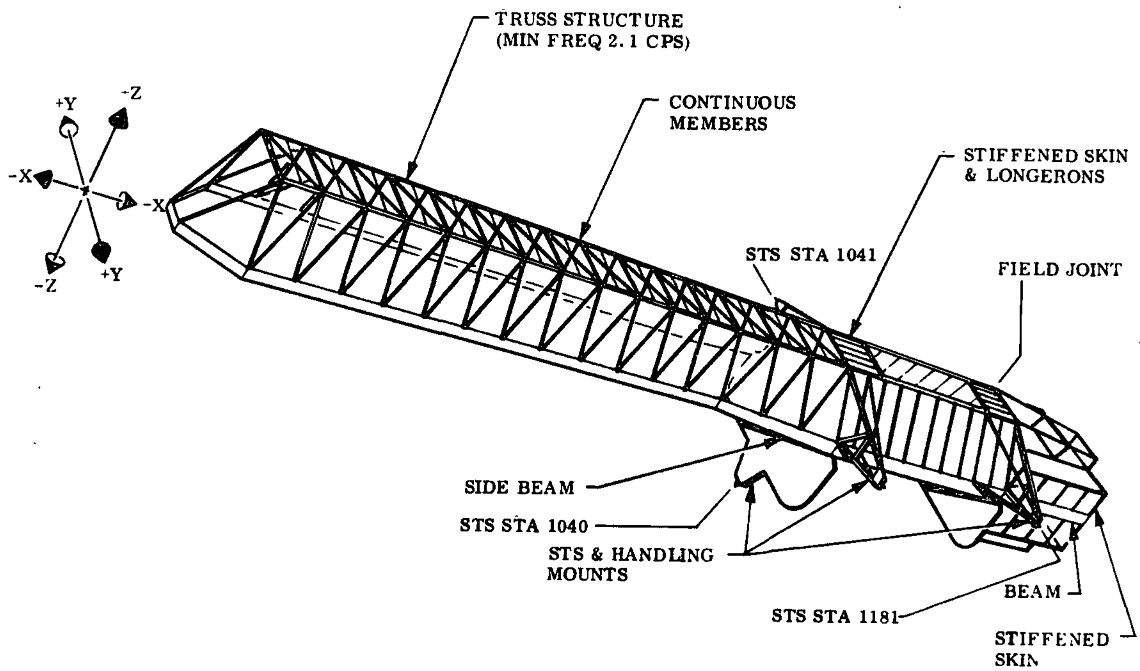


Fig. 5.3-3 Structure Design

5.3.3 Mass Properties

The weights and center of gravity for the SV were determined for three vehicle conditions: ascent, normal return, and abort. The abort condition assumed that all propellants would be dumped prior to reentry; the normal return condition assumed that all expendables, including RVs, had left the vehicle. The c.g. locations for the three flight conditions, as shown in Fig. 5.3-2, are within the allowable c.g. limits. The SV landing weight after a normal retrieval and reentry is 15,728 lb, well below the 25,000 lb limit. In the abort case the 25,000 limit is slightly exceeded as the SV weight, in that case following propellant dump, is 25,571 lb. These weights include contingency allocations, but do not include the 500 lb allotted to the payload bay and Mission Specialist Station equipment.

The SV weights and c.g. shift during the SV mission are shown in Table 5.3-2. The contingency for each vehicle category is also shown. The c.g. shift during the mission is equal to or less than currently experienced. Preliminary analysis indicates that no thrust vector control is required on the orbit adjust engines because of this c.g. shift.

5.3.4 Loads and Dynamics

Primary structure loadings were determined for the critical flight conditions. The reactions at the mounting points were calculated for these flight conditions and for the crash conditions. All loads are based on the values given in Table 5.3-3. The results are summarized in Figs. 5.3-4 and 5.3-5.

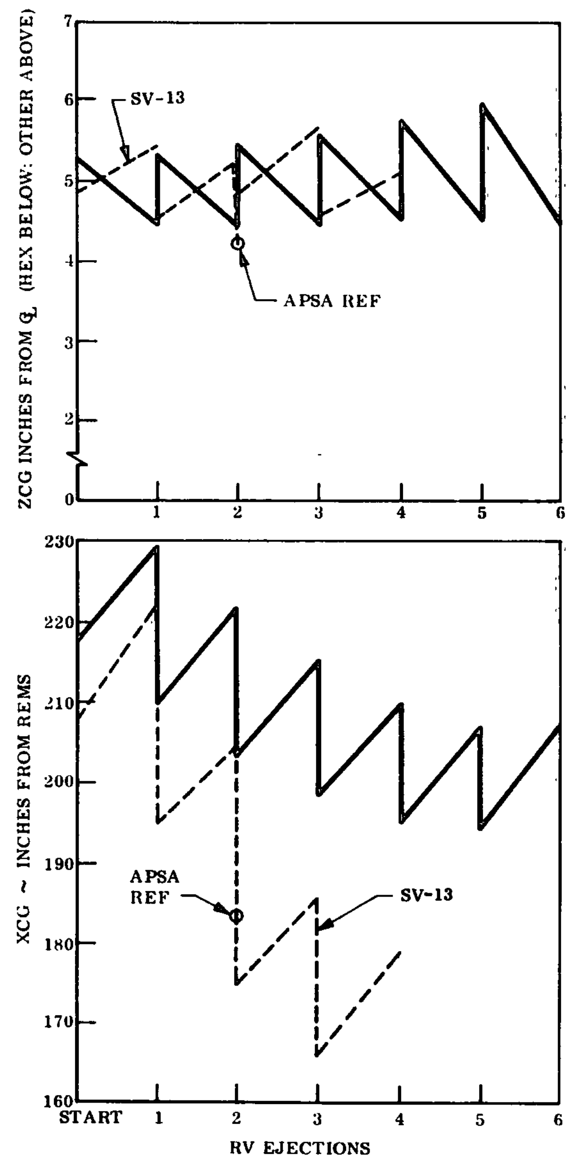
The first cantilever mode of the forward truss structure was estimated by the use of Rayleigh's method. The frequency of the first mode in Y-bending was calculated to be 2.1 Hz, implying second and third mode frequencies of 13.6 and 37.5 Hz. For comparison, the existing SV forward section design has a first mode frequency of approximately 4 Hz. The modal behavior of the new design may result in large deflections of the forwardmost RV under some dynamic loadings, and will require detailed dynamic analysis of the complete SV and STS system.

The random vibration environment (Fig. 5.3-7) was estimated from the acoustic spectrum (Fig. 5.3-6) and from acoustic response data obtained from ground and flight tests. It should be noted that this environment is different in frequency distribution from the environments encountered in present vehicles; therefore, many components may require retest for this environment. This can be conducted during the PRAT tests discussed in Section 6.1. The level of this environment (6.3g RMS) is well within current levels and is not expected to cause any severe problems.

Table 5.3-2

WEIGHT SUMMARY AND C.G. SHIFT

WEIGHT STATEMENT	PERCENT CONTINGENCY		
SATELLITE CONTROL SECTION			3621 LB
EQUIPMENT MODULES	5	1394	
SOLAR ARRAYS	15	158	
OAS ENGINES (2)	5	144	
OAS TANK	10	560	
LIFEBOAT TANK, THRUSTERS, PLUMBING	5	30	
RCS THRUSTERS, CONTROLS, PLUMBING	5	75	
OAS PRESSURIZATION SYSTEM	10	195	
WIRE HARNESS	25	100	
STRUCTURE AND MOUNTS	15	608	
THERMAL CONTROL	15	50	
CONTINGENCY	N/A	307	
FORWARD ASSEMBLY			17,747
SUPPLY CASSETTE	10	1030	
OPTICAL BARS, PLATFORM	10	2325	
PNEUMATICS (INCL. 175 LB GAS) (P-E)	10	375	
ELECTRONICS AND WIRE HARNESSES (P-E)	15	330	
STRUCTURE AND MOUNTS	15	4494	
ELECTRONICS AND WIRE HARNESSES	25	400	
THERMAL CONTROL	15	420	
RECOVERY VEHICLES (6)	0	5220	
TAKEUPS (6)	0	1380	
FILM CHUTES, ARTICULATORS	0	430	
CONTINGENCY	N/A	1343	
SURVIVAL AIDS, SUBSATELLITES			1550
DRY WEIGHT			22,918
PROPELLANT (INCL. 50 LB LIFEBOAT)		7154	
PRESS. GAS (INCL. 1 LB LIFEBOAT)		78	
FILM		2700	
CARGO BAY, MISSION SPECIALIST EQUIP.		500	
GROSS WEIGHT			33,350 LB



BIF003W/2-069331-73

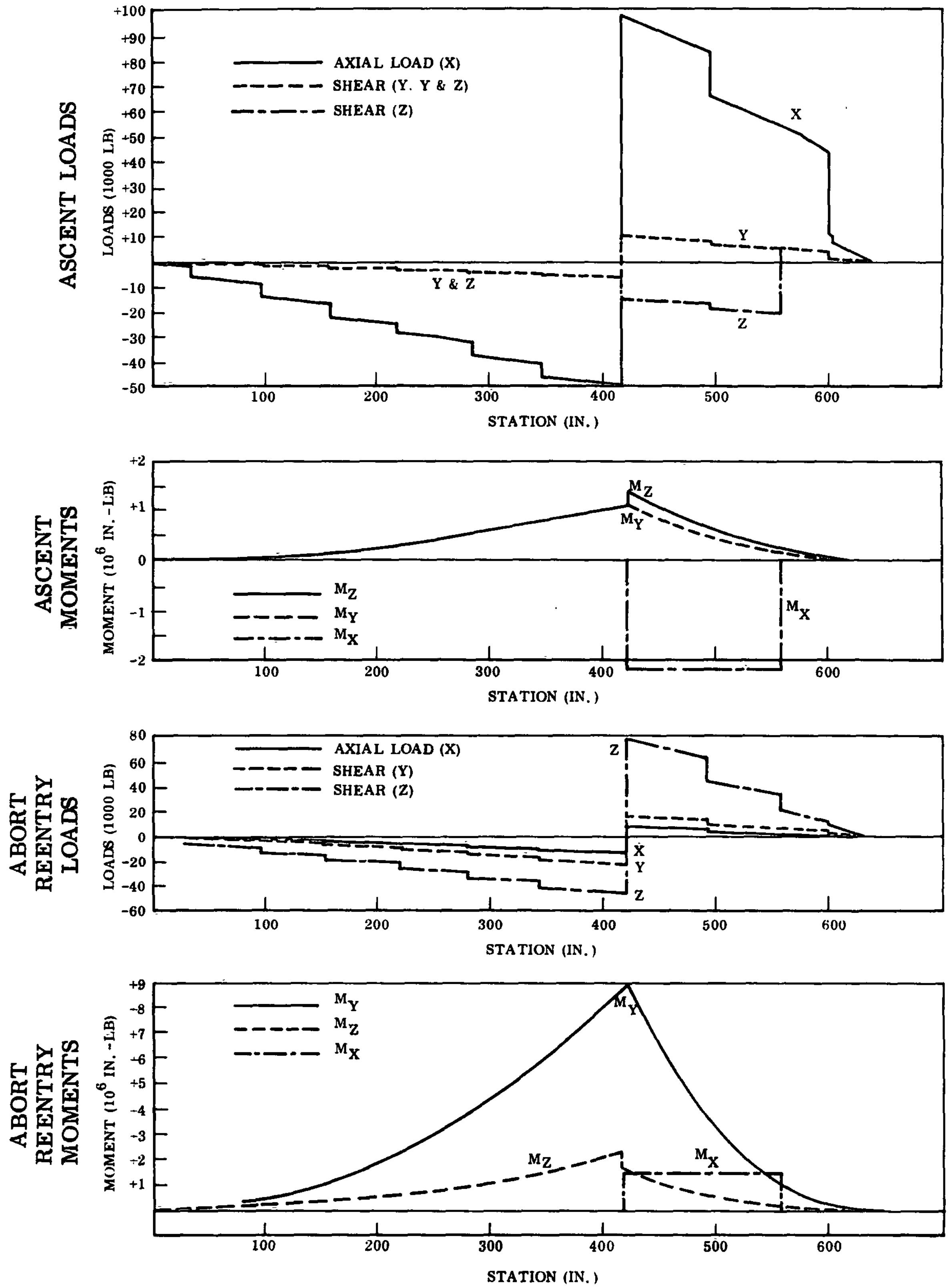


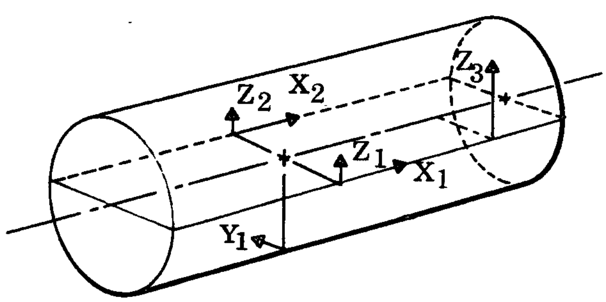
Fig. 5.3-4 Design Loads and Moment

Table 5.3-3

SV DESIGN LOAD FACTORS

Condition	Load Factor (g)		
	N_X	N_Y	N_Z
Ground Handling and Transportation	±3.0	±3.0	±3.0
Ascent, Liftoff	2.0 ±1.0	±0.5	±0.5
Burnout	4.0 ±0.5	±0.5	-0.5
Orbit Operations	±0.3	±0.2	±0.2
Reentry	±0.5	±1.0	+4.0, -1.5
Landing	±2.0	±2.0	+3.5
Crash	-9.0, +1.5	±1.5	+4.5, -2.0

1. The design load factors include an allowance for dynamic loads and are limit load factors except those for the crash condition, which are ultimate load factors.
2. The load factors in the three axes are to be considered to act simultaneously, except for the crash, and ground handling and transportation load factors, which act in any one direction along the major axes.



MAXIMUM REACTIONS, WORST CASE

X_1 = +105,900; -117,900*
X_2 = +105,900; -117,900*
Y_1 = ±73,350
Z_1 = +35,200*; -131,800
Z_2 = +22,324; -99,200
Z_3 = +35,800; -35,200*

(ULTIMATE VALUES IN LB)

*CRASH LOAD

Fig. 5.3-5 Reactions Into STS Payload Bay

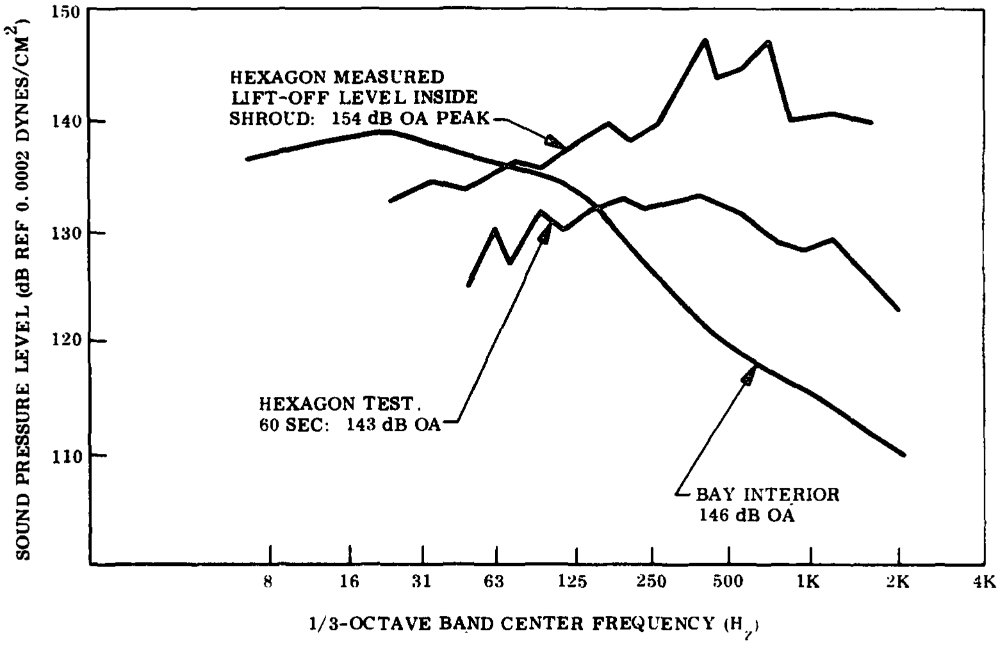


Fig. 5.3-6 Acoustic Environment

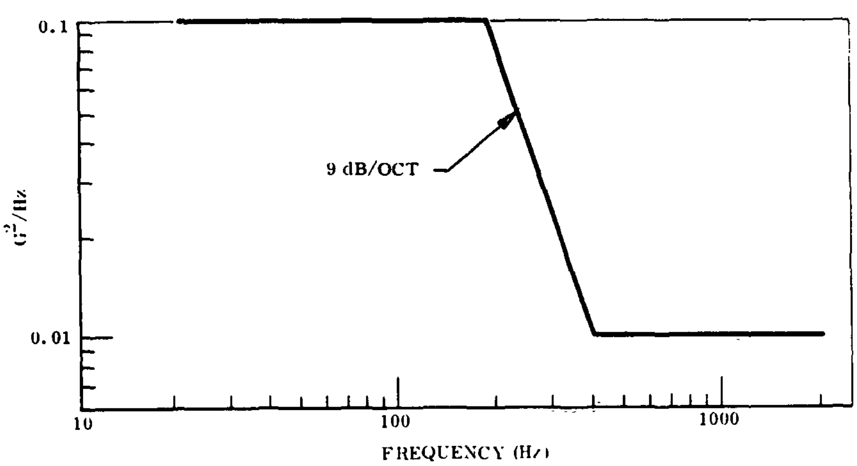


Fig. 5.3-7 Random Vibration Environment

5.3.5 Stress Analysis

The SV stress analysis was primarily concerned with the forward truss structure. The analytical model consisted of rod and bar members. A bar member possesses extensional, torsional, and flexural stiffness; a rod member has extensional and torsional stiffnesses only. The design load conditions considered in the member designs are ascent, burnout, reentry, landing, and crash landing. The load factors associated with these flight conditions are those specified in Table 5.3-3. The loading conditions for ascent, reentry, and landing are considered to be normal operations, and a factor of safety of 1.4 was applied to obtain the ultimate member design loads induced under these normal flight conditions. However, no factor of safety is applied to the crash landing loads. At a crash landing of the STS, the SV is not required to survive the initial impact. However, damage to the Orbiter crew compartment must be avoided by preventing any pieces of the SV from impacting the crew compartment bulkhead.

The results of the static analysis of the design showed that the loads and stresses were reasonable. More importantly, it was discovered that dynamic requirements sized most members; consequently, many stresses in the truss system are well below those needed to satisfy static loads. It should be pointed out, however, that static analysis seldom uncovers the problems of fitting and equipment support design associated with real truss structures used to fulfill the stringent requirements of spacecraft designs. Thermal stress or distortion analysis of the truss was not conducted in this study. Thorough analysis of these aspects will be required.

5.3.6 Thermal Design

The SV thermal design analyses were directed toward three general areas; (1) STS/SV thermal interface requirements, (2) orbital thermal design, and (3) refurbishment requirements. A summary of the results of these analyses is presented below.

5.3.6.1 SV/STS Thermal Interface. The ground air conditioning available through the STS is estimated to be adequate for SV prelaunch thermal conditioning. As indicated in Fig. 5.3-8, payload bay wall temperatures during ascent provide a benign thermal environment resulting in small reductions in external SV temperatures, and have relatively little impact on the SV thermal design. However, after attaining orbit and opening the payload bay doors, extreme thermal environments may be encountered unless STS orientation is restricted. The optimum orientation for the STS with the payload bay doors open appears to be with the payload bay facing the Earth. With this orientation, and with the SV dormant, preliminary analyses indicates that on-orbit temperature requirements can be maintained indefinitely. Further analyses is required to determine the range of STS attitude limits for non-Earth-facing orientations and for SV equipment operation vs dormant operation.

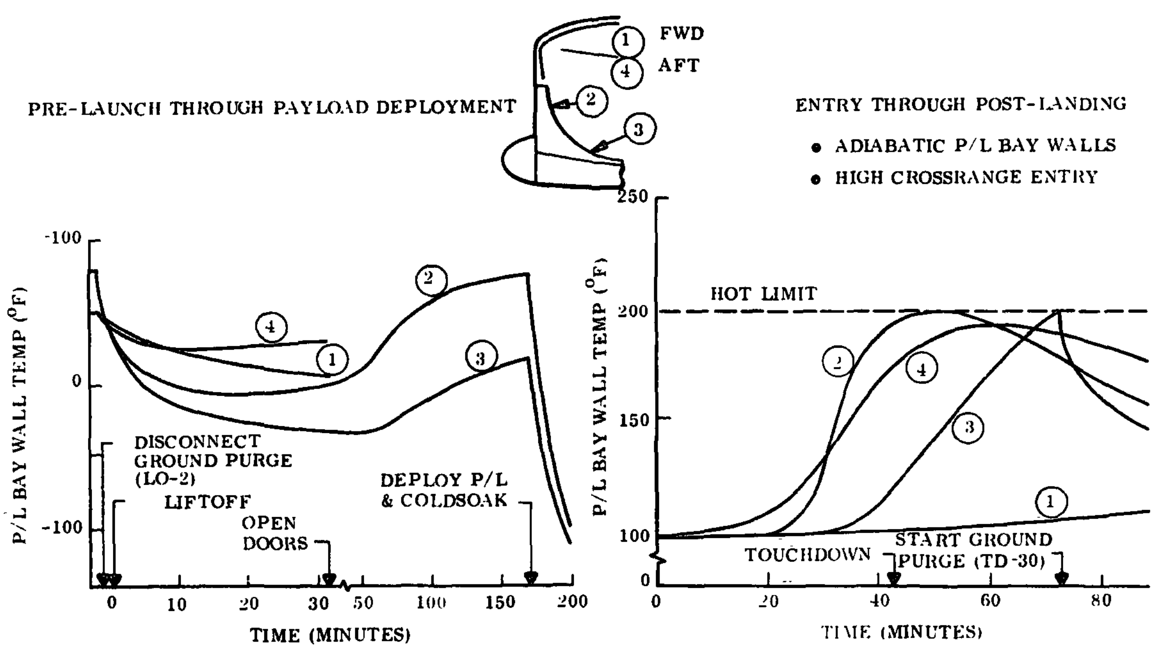


Fig. 5.3-8 Payload Bay Thermal Environment – Typical Mission Phases

The payload bay thermal environment during reentry poses no problems; however, heating soakback after touchdown results in high temperatures on the interior of the payload bay as shown in Fig. 5.3-8. Post-landing temperatures have been computed for SV skin-mounted components, as shown in Fig. 5.3-9. These results show that components in the SV will remain within temperature limits for up to 30 minutes after touchdown. After this time the ground air conditioning system must be activated. A summary of the SV thermal conditioning requirements is presented in Table 5.3-4.

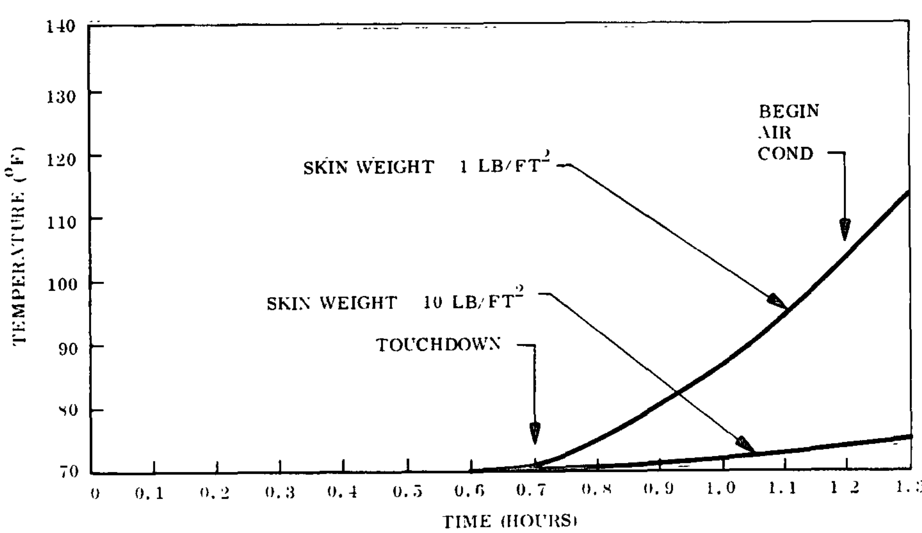


Fig. 5.3-9 Post-Landing Temperature Rise of Skin-Mounted Components

5.3.6.2 Orbital Temperature Control. In general, SV orbital temperature control is simplified and made more reliable by use of the STS. The thermal design concept is shown in Fig. 5.3-10. The large reduction in SV ascent heat loads allows the use of lightweight external shields for component shielding. The STS payload bay environment greatly reduces contaminant fluxes impinging on the SV, virtually eliminating the requirement for large tolerances in optical properties of external thermal control coatings.

Table 5.3-4

THERMAL CONTROL REQUIREMENTS

Mission Phase	Requirement
• Pre-Launch (in Payload Bay)	Air conditioning during checkout and equipment operation, and for propellant temperature conditioning*
• Ascent	Passive thermal control
• Orbit (Before Deployment) STS Doors Open	Orient STS to simulate SV operational thermal environment
• Orbit	Passive thermal control plus heaters for RCS, OAS tank, batteries – same as present
• Reentry (in Payload Bay)	Passive thermal control
• Post-Landing (in Payload Bay)	Air conditioning required no later than 30 minutes after touchdown to maintain equipment below maximum temperature limits (110 – 130°F)

*Existing STS capability estimated to be adequate

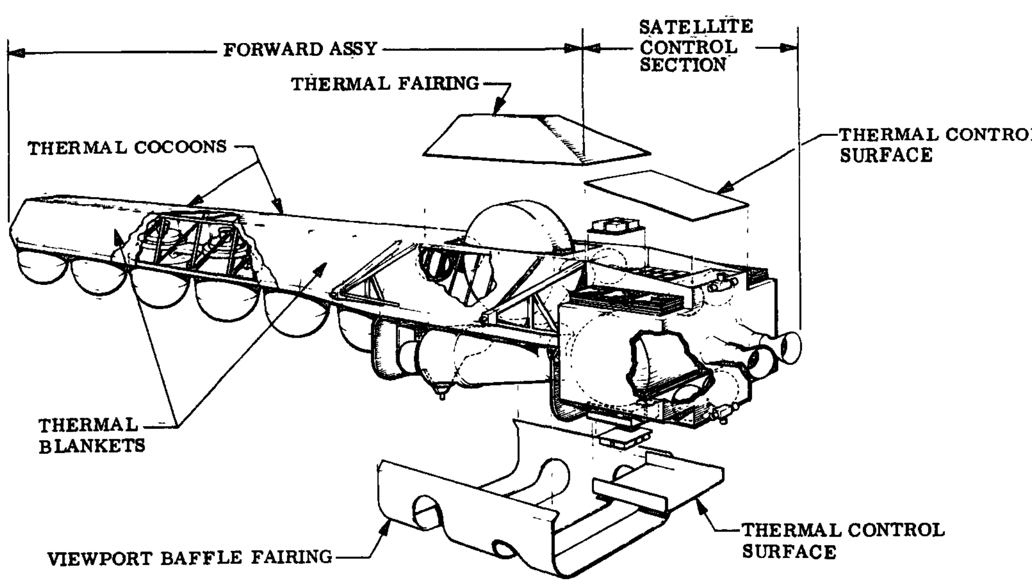


Fig. 5.3-10 Thermal Design Concept

Most SCS equipment is enclosed by lightweight thermal shields to which low α/ϵ thermal control coatings are applied externally.

Preliminary external coating selections include Flexible Optical Solar Reflector (FOSR), Mystik aluminum tape, and bare anodized aluminum. Interior radiative exchange is facilitated through the use of black anodized surface treatment.

Battery temperature control is maintained by mounting batteries on the lower (earth-side) surfaces, with temperature fluctuations controlled by electrical heaters. Heaters are also required for most propulsion system components, as in the present design. Multilayer insulation is used for radiation isolation of the SCS from the Forward Assembly. Further analysis is required to determine conduction isolation requirements.

Temperature control of the Forward Assembly utilizes essentially the same techniques as in the existing system. Multilayer insulation plus electrical heaters are used to maintain required temperature in and along the RVs and film path components. Multilayer insulation also provides adequate protection from aerodynamic heating of the supply unit. Radiation shields between adjacent RV bays and on the base of the Forward Assembly are required to minimize RV heater power requirements.

5.3.6.3 Refurbishment Considerations. Refurbishment of external thermal control surfaces is minimized through the use of space-stable coatings whenever possible. FOSR and anodized aluminum surfaces should require no refurbishment, except to repair damage caused during handling and maintenance operations. Small areas of the SCS require the use of Mystik aluminum tape, which does degrade with long exposure to orbital environments. Replacement of these coatings can be accomplished through the use of prepared tape panels which can be overlaid on the surface to be repaired.

No refurbishment of other coatings or insulation layers is expected to be necessary, other than to repair damage caused during handling and maintenance operations.

5.3.7 Testing

The structural mechanical test program for the SV and the components will be as extensive as the original Hexagon SV test program. New mechanisms will require development and qualification tests, and the structure and structural components will require (besides the usual static and dynamic tests) cyclic loading tests to demonstrate the reusability and predictable repair requirements. Load adapters and dynamic simulators of associated contractors' equipment will be provided by the associate contractors. Major differences from the original structural testing are (1) the simultaneous 3-axis loading instead of 2-axis loading, which complicates the test setup considerably, and (2) the cyclic loading tests. The tests which are required are summarized in Table 5.3-5.

Table 5.3-5

STRUCTURAL AND THERMAL TESTS

Test	Objectives	Description
Coupon and Panel Tests	Establish strength properties and structural stability design data	Apply simulated stress and temperature environments; cyclic loads and ultimate loads
Cargo Bay Support Fittings Tests	Establish structural integrity under crash load environment	Apply crash load levels
Fuel Tank Tests (1) Leak and Proof (2) Burst (3) Slosh	(1) Establish QA acceptance procedure; acceptance test for tank (2) Verify strength and design margins (3) Establish adequacy and integrity of slosh baffling and propellant management systems.	(1) Pressurize with helium to 5 psig for leaks; pressurize to proof pressure (2) Pressurize to burst in hydrostatic tank (3) Apply low-frequency dynamic environment to partially full tank
Vehicle Structure Tests (static) (1) Static Limit Load (2) Cyclic Load (3) Ultimate Load	(1) Verify structural analysis; establish static structural stiffness (2) Demonstrate reusability and predictable repair requirements (3) Establish structural integrity	(1) Apply limit +10% load spectrum (2) Apply operational load spectrum, including ground, STS and space operations (3) Apply ultimate (crash) load envelope
Acoustic Tests (1) With Propellant Simulation (2) Without Propellant Simulation	(1) Establish equipment and structural responses to ascent acoustic environment (2) Establish equipment and structural responses to return acoustic environment.	(1) Apply STS cargo bay acoustic environment to vehicle with simulated equipment and full propellant tank (2) Apply STS cargo bay acoustic environment to vehicle with simulated equipment; empty tank, no RVs
Mechanisms Tests Solar Array	Establish deployment and retraction capability	Perform deployment and retraction (stowing) operations in simulated zero-g environment
Modal Tests	(1) Establish vehicle modes for loads verification (2) Establish vehicle modes for orbit performance analysis verification (controls analyses)	(1) Apply low-level dynamic environments with multiple shakers to vehicle with simulated equip; vehicle with STS simulator (2) Apply low-level dynamic environments with multiple shakers to vehicle with simulated equipments; vehicle in free-free config, with RVs
Thermal Tests	(1) Demonstrate ability of vehicle to meet design and life requirements under temp and pressure conditions	Apply temperature and pressure conditions simulated ground, STS, and orbit environments & operate modes; all-up vehicle

5.4 ELECTRICAL DISTRIBUTION AND POWER (EDAP)

The following paragraphs describe the electrical power requirements, analysis, and concept design for the SV main, Lifeboat, and pyro power systems.

5.4.1 Power Requirements

The power requirements have been established and are tabulated below:

Main Bus

- Voltage 25.5 - 33 VDC
- Surge current 283 amps for 600 msec
- Power (watt-hr/day) 9491
  - SV steady state 8059
  - SV cyclic 1113
  - Camera system cyclic 319
- Orbital life 9 months

Lifeboat Bus

- Voltage 25.5 - 33 VDC
- Orbital life 9 months
- Operate SV critical systems during RV recovery and SV retrieval or deboost

Pyrotechnic Buses

- Voltage 25.5 - 33 VDC
- Orbital life 9 months

The main bus power requirements listed above are less than the Block III Hexagon requirements, primarily because of the deletion of the Mapping Camera Module from this configuration.

5.4.2 Power System Selection and Sizing

5.4.2.1 Main Bus Power. The first consideration in developing the sizing criteria to meet the main bus power requirements was to provide a generating system (solar arrays, fuel cells, or nuclear) capable of providing the total power required plus some contingency for degradation and manufacturing tolerances. Fuel cells and radioactive systems were quickly discarded due to weight, short life, or radiation effects. This eliminated all but the solar array system, which is available and will meet the requirements. The 2 cm x 2 cm violet cells currently under development were selected because their output is approximately 30 percent greater than the cells currently used on the Hexagon program, which will reduce the array size. It is expected that these solar cells will be in use by 1978. The cells are in panels, each of which has a minimum output of 697 watt-hr/day including degradation. Eighteen solar array panels were selected instead of the present 22 panels. This configuration provides adequate margin for degradation and manufacturing tolerances.

The next step was to determine the type and quantity of rechargeable batteries that meet the required life. The type selected for this service was the Ni-Cd Type 29 or a modified version. Evaluation of the power required during the non-charge period indicated that the presently developed Type 29 would provide the necessary power with adequate margin. Other factors considered were the surge current required and the peak charge acceptance capability of each battery. The battery peak charge acceptance capability was the pacing item in selecting the quantity of battieries required. The Type 29 battery has been tested with charge currents as high as 16 amps without degradation. Increasing the charge to a value greater than 16 amps could compromise the battery efficiency and shorten the life cycle due to cadmium migration. The quantity of batteries selected was derived by dividing solar array peak charge current by 16 amps. The surge currents can be met with one less battery, thereby providing a capability to continue the mission with one failed battery. Activity would be reduced but mission termination would not be required.

The battery capacity is controlled by providing a charge current controller for each battery to maintain battery capacity at 90 to 92 percent of full charge. The 90 to 92 percent was selected to minimize the inherent temperature rise during charge and discharge cycles, thereby extending battery life. These charge current controllers are currently used on the Hexagon program.

5.4.2.2 Lifeboat and Pyrotechnic Bus Power. The primary batteries currently used on the Hexagon program cannot be used because of their short life span. Lifeboat power can be derived from a rechargeable and modified Type 29 battery with increased capacity (60 amp-hr instead of 45 amp-hr). This battery will be an off-the-shelf item by 1974. Charge can be maintained by wiring in parallel two of the main bus solar array panels to the Lifeboat battery charge current controller. This method of maintaining the Lifeboat battery capacity will not compromise the main power capability since the capacity-maintenance power is slight.

Each pyro bus will be provided power from a separate main bus battery, with isolation provided by diodes. This eliminates separate batteries that would serve only pyro functions. In addition, backup power to the pyro buses will be provided from the Lifeboat bus. This system has been proposed by LMSC for the Block III Hexagon vehicles, except that the Block III Lifeboat battery is a primary battery.

5.4.3 Power Systems Design

A functional diagram of the power system is shown in Fig. 5.4-1. A system description followed by a detail discussion of the design changes required is presented in the following paragraphs.

5.4.3.1 Systems Description. The main bus design is the same as that currently planned for Block III except that the power transfer switch can now be commanded on or off by secure command. This provides the capability to shut down all vehicle main bus power if desired during an abort sequence.

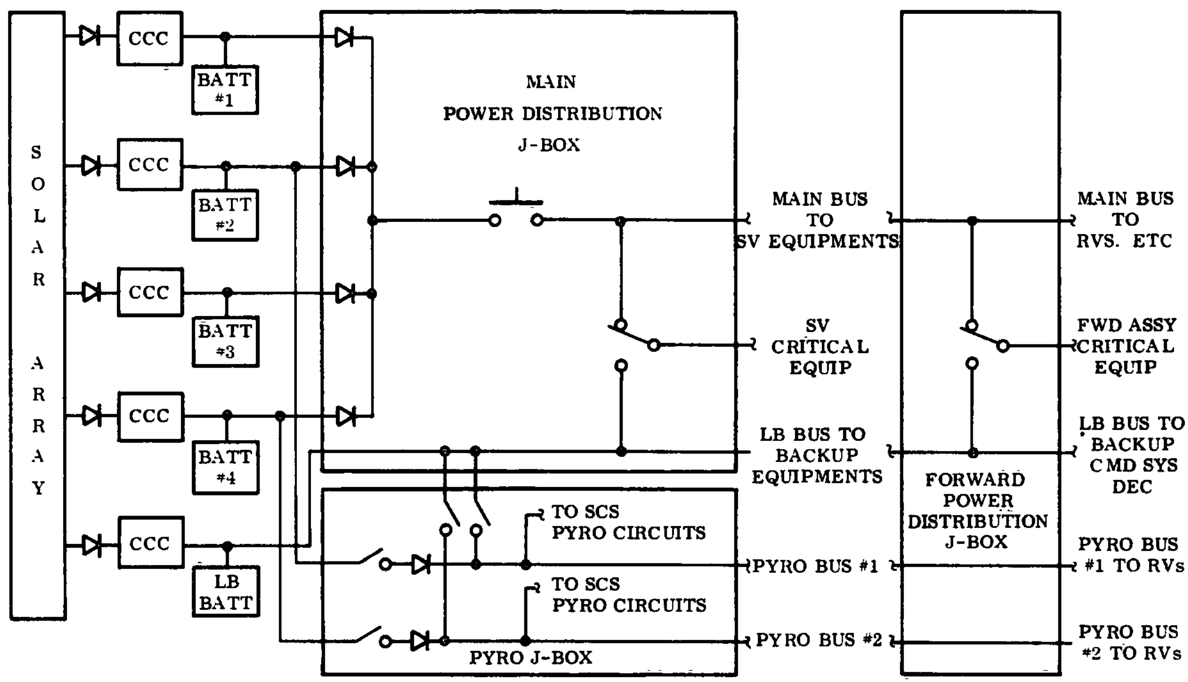


Fig. 5.4-1 Power Systems Functional Diagram

Power is available continuously from the Lifeboat bus to the Lifeboat guidance, backup command system, and Lifeboat receiver. Lifeboat power is switched to other SV equipment that is critical to an SV abort, RV recovery, SV retrieval, or deboost. This equipment includes the redundant telemetry system and pyrotechnic buses. The pyrotechnic buses can be independently switched on or off as required during the mission. This provides additional safety and protects the main bus (or Lifeboat bus) from possible short circuits on either of the pyrotechnic buses.

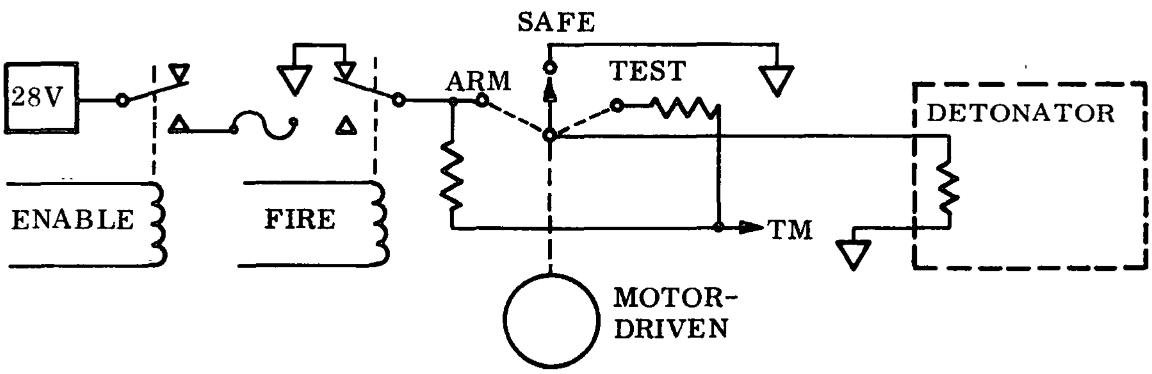
Main bus, Lifeboat bus, and pyrotechnic bus power is provided to the forward power distribution J-box in the Forward Assembly for distribution to the RVs and associated equipment as shown. Modifications are required to provide power to the two additional RVs.

5.4.3.2 Solar Array Design. The solar arrays present a significant problem because the SV cannot be retrieved with the arrays extended. A cost analysis performed during the SV/STS Minimum Modification Study showed that the ability to reuse the arrays for multiple flights could achieve substantial cost savings over the life of the SV/STS program. The use of a retractable array was therefore chosen. In the event the retracting mechanism should fail, a pyrotechnic-activiated separation system must be added to ensure that the solar arrays will not prevent an SV retrieval by the STS.

Implementation of the extension/retraction and blow-off capabilities in the solar array will be by means of the following design changes:

- Redundant DC motors to extend/retract the solar arrays
- Two pyro-operated pinpullers (or equivalent) to allow jettisoning of the solar array wings, if required by failure of retraction mechanism
- Spin-off connectors (pyro-operated) to allow disconnect of the solar array wiring from the SV prior to jettisoning the arrays
- Deletion of the mast erection mechanisms; this can be deleted since the solar array modules are relocated as shown in Fig. 5.1-1.
- Deletion of the present solar array deployment pinpullers pyro functions.

5.4.3.3 Pyrotechnic System Design. To provide additional safety during SV operations in or near the STS, the pyrotechnic safe/arm plugs were replaced by a remotely activated safe/arm switch. A schematic representation of a typical pyro circuit with this safe/arm capability is shown below.



The proposed safe/arm device is a motor-driven switch that can be commanded to the arm position just prior to a pyrotechnic fire event and then immediately returned to the safe position.

A detail design can be implemented where each pyrotechnic circuit is individually armed for an event while all others remain in the safe position. A test position is provided such that continuity of the pyrotechnic circuits can be verified on telemetry, by passing a small current through the entire circuit including the detonator.

Other changes required to the pyrotechnic circuits due to the new SV configuration are:

- Deletion of booster separation, shroud separation, and horizon sensor eject events
- Deletion of the solar array deploy events
- Addition of 21 pyrotechnic-operated valves in the propulsion system
- Addition of solar array jettison, pyrotechnic pinpullers, and spin-off connectors
- Addition of camera system pyrotechnic-operated pneumatic dump valve events
- Addition of pyrotechnic bus power to the two additional RVs.

The safe/arm switching and pyrotechnic circuit changes will be incorporated into the pyrotechnic J-box. The forward power distribution J-box will be modified to provide the pyrotechnic bus power, as well as main bus and Lifeboat bus power to the two additional RVs.

5.4.3.4 Active Thermal Control. The addition of two RVs necessitated a modification to the temperature control electronics assembly (TCEA) and associated harnessing to control heater power to these RVs. The TCEA can be modified by the addition of two additional heater channels to the primary and redundant sections of this box. Other modifications are required to provide heater power to the:

- Redundant orbit adjust engine valve and manifold heaters
- Lifeboat hot gas thruster heaters
- Lifeboat tank heaters

The Block III RCS tank, Lifeboat regulator, and Lifeboat tank heaters will be deleted for the STS-compatible SV design. These changes will be incorporated into the propulsion system J-box. Other propulsion system J-box modifications include relay switching to control the redundant orbit adjust engine, and deletion of the Lifeboat regulator on/off controls.

5.4.3.5 Umbilicals and Harnessing. The majority of the SV wire harnesses will be new because of the harness routing requirements of the new structure. The only exception to this will be some SCS module harnesses. Wiring changes and new harnesses will be required because of the design changes discussed in this and other sections of this report.

The two electrical umbilicals provided for ground test and interfacing with the STS during flight operations can be engaged or disengaged remotely by the STS Mission Specialist Station. Umbilical No. 1 contains hardline control, caution and warning monitors, and battery charging power. Umbilical No. 2 contains coaxial cables for serial-digital command and telemetry information.

During STS flight operations with the SV in the payload bay, battery charging is supplied by the Orbiter power system through the umbilical J-box installed in the STS payload bay. As discussed in Section 5.2, DC-DC converters are required to up-convert the 30.5 VDC Orbiter bus voltage to 34 VDC at the SV charge current controllers, to maintain adequate battery charge.

5.5 PROPULSION SUBSYSTEM

The SV propulsion subsystem includes an integrated Orbit Adjust System (OAS)/Reaction Control System (RCS), and an operationally independent Lifeboat hot gas reaction control system. The subsystem design emphasizes high reliability, ease in refurbishment, and safe SV/STS operations.

5.5.1 Requirements

The propulsion subsystem design satisfies the following requirements:

- Orbit adjust and attitude control capability for 9 months on-orbit life
- Backup attitude control capability for RV recovery, SV retrieval, or SV deboost events
- Propellant dump capability for:
  - Abort (STS ascent, glide, orbit)
  - On-orbit retrieval (attached in STS and/or deployed)
- Propellant tank at standby pressure during ascent
- Refurbishment ease (flush/purge/hardware replacement).

5.5.2 Trade Studies

The following section discusses the results of the tradeoff studies performed on the propulsion system.

Monopropellant vs Bipropellant System. The two main types of propulsion systems studied were bipropellant and monopropellant. A bipropellant system was considered favorable from a weight savings standpoint (because of higher specific impulse) but was dropped as a candidate because of overall disadvantages in terms of development cost, reliability, refurbishability, operations, and safety. The monopropellant hydrazine system offers the benefits of an integrated OAS/RCS system and attendant simplicity of a common propellant tank. The bipropellant system has a disadvantage in reliability for a similar arrangement and has another associated problem in that a 5-lb bipropellant RCS thruster would have to be developed. Table 5.5-1 provides a summary of the advantages and disadvantages of the two systems for the various criteria.

Table 5.5-1

PROPULSION SYSTEM TRADEOFF SUMMARY

Considerations	Monopropellant System	Bipropellant System
Weight	(Combined OAS/RCS with integrated storage of propellant provides some weight saving over current system.)	Advantage: Specific impulse difference (285 vs. 235 sec). However, some additional weight penalty would arise due to systems considerations of thermal control, propellant utilization and structure. Also additional weight increase for bipropellant RCS when compared to integrated OAS/RCS monopropellant system. Designing redundancy into bipropellant system adds weight.
Reliability	Advantage: Fewer components and one propellant. Catalytic decomposition rather than hypergolic combustion process, thus eliminating the criticality of sequencing propellant entry into thrust chamber to avoid destructive starting in "vacuum".	
Cost	Advantage: System simplicity/fewer components. Reduction in design, test, data acquisition, and analysis. Ground support activities and equipment proportionally reduced. Production costs consequently lower. Hardware easier to manufacture. Injector less complex, materials easier to fabricate, less components assembled and checked out, less instrumentation for calibration. Acceptance test is reduced.	
Development Risk	Advantage: Lower development cost related to lower development risk. LMSC has successfully flown monopropellant OAS/RCS (Hexagon Program) and currently developing two other monopropellant systems (design/test) with OAS/RCS capability. Compliance with LMSC design specs, including performance and producibility, are well advanced. Design of monopropellant tank with passive propellant management is uncomplicated as demonstrated by existing LMSC flight experience. Involves significantly less problem than designing for both fuel and oxidizer service as required for bipropellant tanks. Less schedule risk with a monopropellant system, and consequently affords more opportunity to recover in the vent of delays.	
Factory/Pad Operations	Advantage: Overall operational simplicity; only one propellant with excellent storability characteristics. Simplified AGE, propellant transfer, and system checkout equipment. Reduced number of parameters to be monitored for test and flight. Easier to assemble, test, maintain, service, and operate.	

Table 5.5-1 (Cont)

Considerations	Monopropellant System	Bipropellant System
Vehicle Integration	Advantage: Normally easier to accommodate due to compactness and requires less vehicle volume. Vehicle thermal control simplified because of single tank heat transfer characteristics. Interfacing is less complex, reflecting fewer functions to be performed. The lower operating temperature and reduced heat radiation of the chamber and its exhaust presents fewer thermal control problems. Simplified and attractive design feature is utilization of monopropellant hydrazine for both OAS/RCS control with integrated storage of propellant. This feature provides weight savings.	Disadvantage: Two tanks (oxidizer and fuel) may impose volume constraints. Requires rigid tank temperature control so tank pressures do not get excessive due to vehicle temperature changes. This would impact the mixture ratio, which is a critical parameter, and thus would degrade I_{sp}. Additionally, to provide one nested tank for both oxidizer and fuel with common bulkhead impacts safety margin considerations.
Contamination (Solar Array)	Advantage: Lower operating temperature and reduced heat radiation of chamber and its exhaust presents fewer thermal problems than bipropellant system.	(Contamination of solar arrays with bipropellant system during RCS operation requires study. Low temperature combustion in bipropellant with RCS operating mode could result in carbon exhaust products. Requires plume impingement and contamination study.)
Material Compatibility	Advantage: Non-corrosive propellants compared to bipropellant system.	
Safety Margins	Advantage: Lower temperature energy conversion process. (Catalytic vs hypergolic combustion process.)	(Reference comments under Vehicle Integration)

External Repressurization vs. Blowdown Monopropellant System. Evaluation of a blowdown and external repressurization system indicated that, although the blowdown mode offers simplicity and higher reliability due to fewer components, it requires a larger tank size which impacts vehicle integration considerations. The external repressurization system provides minimum tank size and offers an additional safety feature of a low-pressure capability during ascent. On the basis of tank size constraints (for a propellant load of 7100 lb) the external repressurization system was selected for this design. Figure 5.5-1 illustrates the tank size comparison of blowdown (requires a 58-in. cylindrical insert) vs. external repressurization (requires a 38-in. cylindrical insert with approximately 20 percent ullage). This figure also illustrates the propellant load and tank size relationship for spherical and non-spherical tanks.

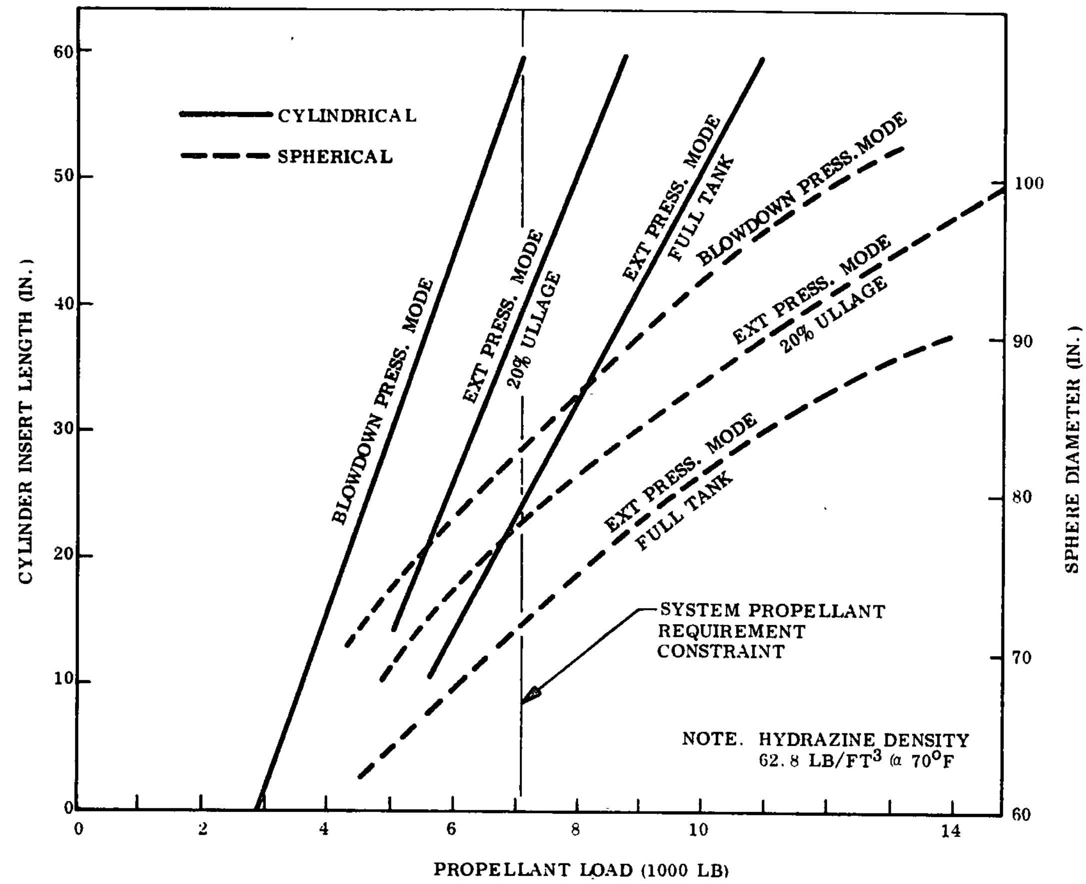


Fig. 5.5-1 Propellant Tank Characteristics and Requirements

Cylindrical Tank vs. Spherical Tank. In order to satisfy the total mission requirement of 7100 lb of hydrazine, three ways of increasing total tank capacity were evaluated:

- For minimum weight, an 83-in.-diameter spherical titanium tank
- For minimum development, three existing Block III 62-in.-diameter spherical aluminum tanks; but lower reliability due to increased plumbing
- For a compromise between weight, development and reliability, an existing Block III 62-in. spherical aluminum tank with a cylindrical insert.

The 83-in. spherical tank requires extensive development costs, and incorporation into the vehicle SCS impacts equipment integration considerations. The present 62-in.-diameter tank with a 38-in.-long cylindrical section inserted between two halves was selected on the basis of weight, development, and reliability (compared to three 62-in.-diameter tanks). This tank design requires redesign of the present propellant management device (PMD) to accommodate the increased fluid head. The weight of the pressure shell for this tank is greater than that of three spherical tanks; however, due to the lower number of welds and their reinforcements, the overall weight can be less. This tank design is optimum from a cost-effectiveness viewpoint.

Monopropellant Lifeboat System vs. Cold Gas Lifeboat System. The monopropellant hydrazine system was selected because it offers the following advantages when compared to the existing cold gas Lifeboat system (nitrogen/Freon system):

- Net weight saving of 300 lb (propellant and tank weight)
- Tank heating not critical as required for a nitrogen/Freon system and does not impose any severe constraints
- Hydrazine tank allows for additional propellant capability of up to 100 lb (50 lb hydrazine required for present mission)
- Lower pressure system (200 psia vs 2800 psia).

The monopropellant hydrazine system can be incorporated with no significant development cost penalty because existing designed tank and thrusters are used.

Multi OA Engines Vs Requalified Single OA Engine. The mission requirements of total impulse (1,472,000 lb-sec) and 200 engine starts impose significant demands on the OA engine design. The critical element in the life of the engine is the catalyst bed. The degradation mode for this design is thrust degradation (reduction in specific impulse). The mechanism for degradation is that the catalyst bed reaction progresses downstream with use and can result in propellant not reacting in the catalyst bed, but in the nozzle and/or propellant exiting the nozzle. This phenomena results from some deactivity of the catalyst with use and catalyst attrition (voids due to catalyst breakup) due to ambient firings. Although the OA engine design has demonstrated in testing a capability of

2,000,000 lb-sec and 242 starts, the many variables associated introduce risk for those mission requirements, especially at the end of the mission when the SV has to adjust into a retrieval orbit. Therefore, rather than project engine development to guarantee these requirements and their attendant high development costs, it becomes more cost-effective to incorporate two of the current Block III OA engines to satisfy this mission requirement.

5.5.3 System Characteristics

The systems design, based upon tradeoff study results, is illustrated in Fig. 5.5-2. This design provides at least a full 2 to 1 factor of safety when loaded and pressurized.

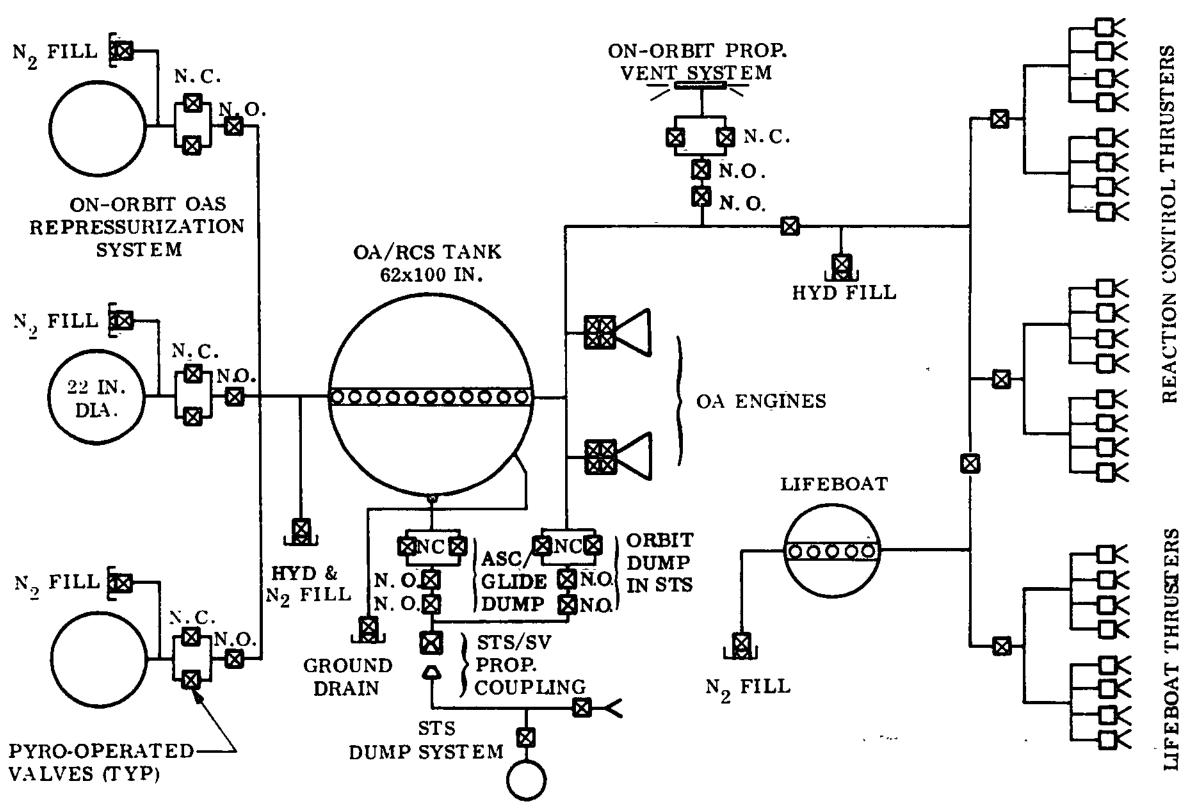


Fig. 5.5-2 Orbit Adjust/Reaction Control System

Table 5.5-2 is a summary of the pressure ratings of the system.

Table 5.5-2

PROPULSION SYSTEM DESIGN SAFETY MARGINS

System	Design Working Pressure (psig)	Proof Pressure (psig)	Design Burst Pressure (psig)
Orbit Adjust			
Propellant Tank	300	450	600
Pressurization Spheres	3000	4500	6000
Isolation Valves	400	600	800
Propellant Fill Valves	400	800	1600
OA Engines	300	450	600
N_2 Fill and Pyro-Operated Valves	2000	3000	4000
Reaction Control			
Fill Valves	400	800	1600
Isolation Valves	400	600	800
Filters	400	900	1200
RCS Thrusters	300	600	1200
Lifeboat			
Tank	350	525	700
Fill Valve	400	800	1600
Thrusters	300	600	1200
Filters	400	900	1200
Isolation Valves	400	600	800

The system characteristics which satisfy mission requirements are discussed below:

Propellant Dump Capability. The propulsion system is designed to provide SV/STS gravity feed and a fast propellant dump capability to vent OAS propellants during an STS launch abort in STS ascent (SV vertical) and Orbiter glide (SV horizontal) flight modes. This feature is provided by an OAS tank dump line (two locations on tank), pyro-operated valves, and an SV/STS propellant coupling system. This feature depends on the gravity field present for propellant orientation and will allow the STS, in case of an abort, to land with an empty, unpressurized SV OAS tank.

The propulsion system is designed to also provide SV on-orbit low-gravity propellant dump capability to vent OAS propellants thru the Orbiter dump system when in the retrieval mode. This is done through the OAS tank dump line, pyro-actuated valves, and the SV/STS propellant coupling system, and/or in the deployed mode it is done through the PMD, pyro-operated valves, and discharge nozzles designed to nullify the impulse during venting operations. This latter feature affords on-orbit vent capability independent of the STS while in a deployed mode. In addition to this capability, OA engine firings could be programmed to burn residual propellants prior to retrieval by the STS. Although the SV will normally be retrieved with most of the on-board propellants consumed except for residuals required for contingency and deboost, safety aspects are improved during SV/STS retrieval/deorbit operations by dumping propellants on-orbit prior to reentry. The OAS tank PMD requires low "g" conditions and the dump time required is a function of the design.

The propulsion system is designed to provide SV post-landing propellant drain capability with the SV in a horizontal orientation. This is accomplished through the OAS manually-operated ground drain line. Any remaining Lifeboat propellants can be transferred to the OAS tank and can thereby also be removed with the OAS propellant.

On-Orbit OAS Repressurization. The on-orbit OAS external repressurization system consists of three 22-in.-diameter spheres (from the existing Lifeboat system). These spheres will have an operating pressure of no greater than 1500 psia. One sphere will provide the pressurant (GN_2) source to fully pressurize the OAS tank to its initial operating pressure on-orbit. This method allows for reduction of OAS tank stresses during the launch and abort operations. Since most of the tank stresses are a direct result of tank pressurization induced loads, this feature takes advantage of the inherent tank strength to withstand high "g" loads when unpressurized.

The two other spheres will provide the pressurant (GN_2) source necessary to maintain the OAS tank feed pressure within the operating range during the mission. These two spheres will be used separately and spaced at intervals to provide an operating pressure range compatible with the hardware and sufficient at the end of the mission to provide satisfactory thrust capability in the event an SV deboost is required. The OAS tank pressure profile during the mission, utilizing external pressurization, is illustrated in Fig. 5.5-3.

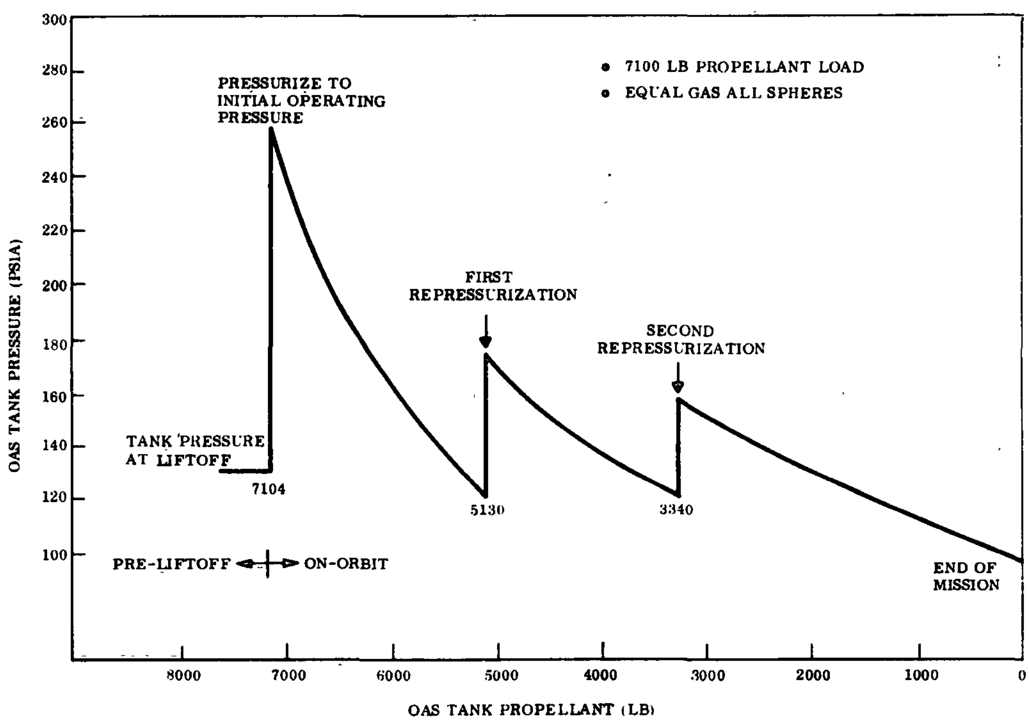


Fig. 5.5-3 OAS Tank Pressure Profile

In additon to these tanks, the on-orbit repressurization system consists of fill valves and pyro-operated valves. The pyro-operated valves provide high reliability and inherently leak-tight features, critically necessary for the long mission duration. The valves are arranged to provide redundancy for the mission-critical (one-time) "open" event. Also provide is a normally-open valve, which will be closed after the repressurization event to prevent hydrazine from migrating into the pressurant spheres.

Refurbishment. The system design emphasizes maximum protection from fluid leakage, using a brazed plumbing system and pyro-operated (one-time use) valves. However, for ease of refurbishment, judicious placement of a minimum number of flange fittings with double "O"-rings will be implemented. This same method is planned and in development on another LMSC program.

The current OAS tank design has a non-removable passive propellant management device (PMD). The allowable pressure drop across the PMD screened components (galleries) is low (< 1 psid), which makes it difficult to reclean and test when installed in the OAS tank. The PMD design will be modified to eliminate the screens in the galleries, using perforated plates/baffles instead. This will permit the tank and PMD to be adequately cleaned. The lower receiver and OAS tank outlet will be modified to permit removal, recleaning, and replacement. This feature will lift the restriction on reuse of the OAS tank, which is currently limited to three propellant loads due to possible contamination buildup in the PMD.

The reaction control thrusters and OA engines will be removed for refurbishment and the filters will be replaced after each flight. To facilitate refurbishment, the type of plumbing connections for the thrusters, filters, and OA engines will be designed to provide separable type connections.

During the refurbishment, the OAS repressurization spheres will be vented through the fill valves, retaining a low positive pressure until the repressurization flight load is required during pre-launch operations. No special refurbishment is required.

Performance. The capabilities of the propulsion system are summarized below:

	OAS	RCS and Lifeboat
Thrust	280−125 lb	5−2 lb
Isp	230 sec	200 sec steady state 120 sec pulse mode
Tank Pressure	300−85 psia	300−85 psia
Starts	200; 100 amb, 100 restarts (Block III qualified for 120, demonstrated ~ 240)	300, 000 pulses (within Block III requirements)
Total Flow	6400 lb	700 lb

The OAS engine is designed to operate over a pressure range of 300−85 psia and provide thrust capability of 280 to 125 lb, while providing a minimum specific impulse of 230 seconds. The propellant temperature range is 70° to 100°F for the current engine design. The engine is required to provide approximately 200 starts (100 ambient and 100 hot restarts) for the mission duration. The current Block III engine will be qualified to 120 starts and this design has demonstrated 240 starts. The total flow required is 6400 lb of hydrazine or approximately 1, 472, 000 lb-sec of impulse. The current Block III will be qualified for 750, 000 lb-sec but the design has demonstrated up to 2, 295, 000 lb-sec of impulse. The engine is capable of impulse bits of 650 to 165, 000 lb-sec. The quad valve offers valve redundancy in that the engine can operate through only one leg of the quad valve and satisfy performance requirements. The impulse predictability for the OAS is 5 percent for impulse bits greater than 10, 000 lb-sec.

The RCS thrusters are designed to operate over a pressure range of 300 to 85 psia and provide thrust capability of 5 to 2 lb. The steady-state specific impulse is at least 200 sec and goes to a minimum of 120 sec for pulse mode operation. The propellant temperature range is 40 to 140°F. The most active thrusters will be required to provide a 300. 000-pulse capability. This requirement is within the Block III thruster demonstrated capability. The total propellant flow requirement is 700 lb of hydrazine, whereas the current Block III requirement is approximately 550 lb.

5.6 ATTITUDE CONTROL SYSTEM

This section discusses the primary and Lifeboat attitude control requirements and design concept.

5.6.1 Primary Attitude Control

The requirements for the primary attitude control system are summarized below:

- Control Attitude errors and rates during camera operations to:

	Attitude (deg)	Rates (deg/sec)	
Roll	±0.70	±0.021	Same as Block III requirements
Pitch	±0.70	±0.014	
Yaw	±0.64	±0.014	

- Control SV during maneuvers, orbit adjust, RV separation, and deboost
- Provide SV stability for retrieval.

A review of these performance requirements and operating conditions has shown that the present Block III system can be used for the STS-compatible SV design. The pertinent items considered in the review and the evaluations are given below:

Required Vehicle Attitudes and Rates

- During camera system operation – same as present
- During deployment/retrieval – current capability acceptable

Disturbances

- RV separations – linear impulse same; torque impulse greater (~20%) only for No. 1 RV but still within control moment capabilities
- Aero torques – reduced to near insignificant values due to the higher altitudes flown versus Block III
- Orbit adjust torques – thrust and c.g. excursion same as present
- Camera system disturbances – individual disturbances larger than present actual values but not expected to exceed Block III Interface Control Document (ICD) values; detail studies may show minor revision of settling times, particularly during mono operation.

Reaction Control Requirements

- Thrusters – design permits moment arms same as present; existing thrusters can be used.
- ACS/RCS compatibility – vehicle inertias and geometry satisfy present arm-to-inertia ratio requirements, therefore rate limit cycling avoided.
- Total control impulse – increased for more and larger camera system disturbances; more RV separation maneuvers and more orbit adjusts are mostly offset by reduction of aero torque disturbances; total estimated at less than 50 percent increase over present requirements; this is within Block III capability.

The booster separation command which activates the ACS will be deleted by a harness modification. This is the only modification required on the ACS system. A functional block diagram of the ACS is shown in Fig. 5.6-1.

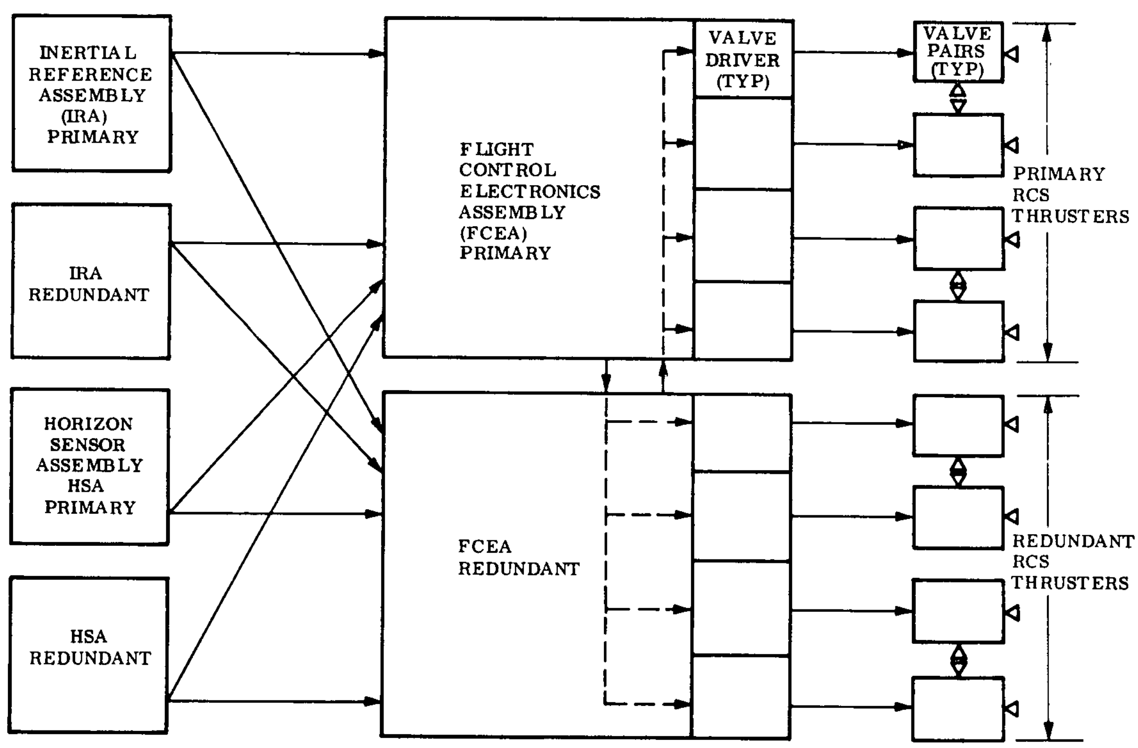


Fig. 5.6-1 Attitude Control System

5.6.2 Lifeboat System

The requirements to be met by the Lifeboat system are the same as the existing requirements, except that the backup capability to provide attitude control for an SV retrieval by the STS is required in the event the primary ACS has failed. Also, the present requirement to recover a second RV is deleted because the MCM is not included in this configuration. The requirements for the Lifeboat system are summarized below:

Operations

- 1 RV recovery
- 1 SV retrieval by STS
- 1 SV deboost

Maximum SV Capture Rates

- 15 deg/sec roll
- 2 deg/sec pitch and yaw

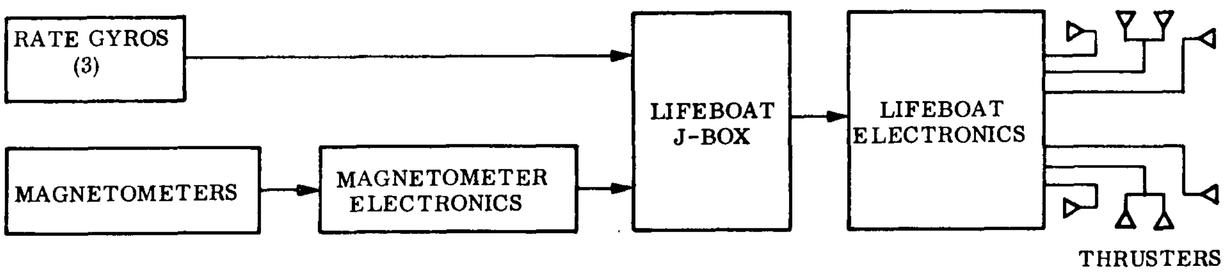
Capture Response

- Within 5 minutes

Control Accuracy

- Attitude: ±5 deg
- Rates: ±0.3 deg/sec

The only box modification required is the addition of relays (with associated commands and instrumentation) to the Lifeboat J-box to enable and disable the magnetometer input to the electronics which is necessary for the inertial mode of operation. In addition, the harnessing from the Lifeboat electronics to the thrusters will be modified to simultaneously operate the pitch control hot gas thrusters. This was necessary because the existing RCS hot gas thrusters are designed in an 8-valve configuration while the cold gas thrusters were in a 6-valve configuration. A block diagram of the Lifeboat attitude control system is shown below:



5.7 TRACKING, TELEMETRY, AND COMMAND (TT&C) SUBSYSTEM

The Block III TT&C system will be expanded for the SV/STS by adding additional command and telemetry capability. Driving factors for this increased capability are:

1. Additional safe and arm provisions
2. Monitoring of vehicle status
3. Provisions for the STS to have control during deploy/tests/retrieval
4. Additional OAS engine, repressurization, and propellant dump system
5. Retractable solar arrays
6. Longer life and refurbishment considerations
7. Two additional RVs

5.7.1 Commands

The added commands necessary to perform these functions are presented in Table 5.7-1.

These new functions require 151 additional extended command system (ECS) commands and 60 additional minimal command system (MCS) commands. Also required are 135 additional telemetry points. In order to provide for STS on-board processing of SV data, two 16 KBPS telemetry formats are required in the PCM system.

Table 5.7-2 lists the present capability of the ECS and MCS and the number of commands required for the new STS-compatible design. As shown, 10 ECS commands in excess of Block III capability are required. This capability is easily obtained by the addition of an ECS remote decoder which will meet the new requirements and provide 54 spare commands for future growth. This change requires only a minor modification of the existing system.

Table 5.7-1
ADDITIONAL COMMANDS

Subsystem	Command Type	Primary Commands	Backup Commands
Tracking Telemetry, and Command	RTC	FM telemeter ON (non-red.)	–
	NSPC	FM calibrate (non-red.)	–
	SPSPC	Remote decoder 2 ON	–
	PSPC	Remote decoder 2 OFF	–
	SRTC (internal)	Clock reset	Clock reset
	SRTC (internal)	Clock hold	Clock hold
	SRTC (internal)	Clock release	Clock release
	SPSPC (internal)	Secure word gen 'A' ON	Secure word gen ON
	SPSPC (internal)	Secure word gen 'B' ON	–
	NSPC	Select SV checkout (16 KBPS) PCM format	Select SV checkout (16 KBPS) PCM format
	NSPC	Select P/L checkout (16 KBPS) PCM format	–
	RTC	C&W inst pwr 1 ON (non-red.)	–
	RTC	C&W inst pwr 1 OFF	–
	RTC	C&W inst pwr 2 ON	Same as primary
	RTC	C&W inst pwr 2 OFF	Same as primary
Electrical Distribution and Power	NSPC	Connect pyro bus 1 to main bus	–
	NSPC	Connect pyro bus 1 to Lifeboat bus	Same as primary
	NSPC	Connect pyro bus 2 to main bus	–
	NSPC	Connect pyro bus 2 to Lifeboat bus	Same as primary
	NSPC	Disconnect pyro buses from main bus	–
	NSPC	Disconnect pyro buses from from Lifeboat bus	Same as primary
	SPSPC	Main transfer sw open	
	PSPC	Main transfer sw closed	
	SPSPC	Pyro safe/arm to arm	
	PSPC	Pyro safe/arm to safe	
	PSPC	Activate solar array 1 spinoff	
	PSPC	Activate solar array 2 spinoff	
	PSPC	Jettison solar array 1	
	PSPC	Jettison solar array 2	
	NSPC	Extend solar array 1	
	NSPC	Extend solar array 2	
	NSPC	Retract solar array 1	
	NSPC	Retract solar array 2	Same as primary

Subsystem	Command Type	Primary Commands	Backup Commands
Propulsion	PSPC	Fire No. 1 tank NC valves	Same as primary
	PSPC	Fire No. 1 tank NO valves	
	PSPC	Fire No. 2 tank NC valves	
	PSPC	Fire No. 2 tank NO valves	
	PSPC	Fire No. 3 tank NC valves	
	PSPC	Fire No. 3 tank NO valves	
	PSPC	Fire OA tank orbit dump NC valves	
	PSPC	Fire OA tank orbit dump NO valves	
	PSPC	Fire OA/STS orbit dump NC valves	
	PSPC	Fire OA/STS orbit dump NO valves	
	PSPC	Fire OA/STS ascent dump NC valves	
	PSPC	Fire OA/STS ascent dump NO valves	
	NSPC	Select OA engine 1	
	NSPC	Select OA engine 2	
	PSPC	OA engine prop valve pri htr ON	
	PSPC	OA engine prop valve red htr ON	
	NSPC	OA engine 2 pri manifold htr ON	
	NSPC	OA engine 2 red. manifold htr ON	
	NSPC	OA engine 2 pri manifold htr OFF	
	NSPC	OA engine 2 red. manifold htr OFF	Same as primary
Guidance	NSPC	Lifeboat to inertial mode	Same as primary
	NSPC	Stop Lifeboat inertial mode	Same as primary
Camera System	NSPC	Six commands per P-E request	Two pneumatic dump commands
Recovery Vehicles	NSPC	RV5 test pwr ON (non-red.)	–
	NSPC	RV6 test pwr ON (non-red.)	–
	NSPC	RV5 orbit power ON	Same as primary
	NSPC	RV5 orbit power OFF	
	NSPC	RV6 orbit power ON	
	NSPC	RV6 orbit power OFF	
	SPSPC	RV5 separate	
	SPSPC	RV6 separate	
	SPSPC	RV5 FOTS	
	SPSPC	RV6 FOTS	
	SPSPC	RV5 battery activate	
	SPSPC	RV6 battery activate	
	SPSPC	RV5 arm	
	SPSPC	RV6 arm	
	SPSPC	RV1 B/U arm	
	SPSPC	RV2 B/U arm	
	SPSPC	RV3 B/U arm	
	SPSPC	RV4 B/U arm	
	SPSPC	RV5 B/U arm	
	SPSPC	RV6 B/U arm	
	SPSPC	RV5 fits	
	SPSPC	RV6 fits	Same as primary

NOTES:

(1) All primary commands are redundant unless otherwise noted.

(2) RV Reset (existing command) safes all RV pyro circuits.

Table 5.7-2

COMMAND REQUIREMENTS

	ECS			MCS		
	Block III	Added	Required	Block III	Added	Required
OAS/RCS	46	40	86	7	20	27
EDAP	32	36	68	4	13	17
ACS/LB	59	4	63	13	2	15
TT&C	34	17	51	13	3	16
Subsatellite	14	–	14	–	–	–
Camera System	128	12	140	–	2	2
RV	54	42	96	28	20	48
MCM	51	–	0	6	–	0
Spare	90	–	–	15	–	–
TOTAL	508	151	518	86	60	125

5.7.1.1 STS Control Method. To ensure STS safety and accomplish the SV mission, it is necessary to control the time of occurrence of the abort and checkout sequences. The occurrence of these events is controlled by the ECS clock, which releases each stored program command (SPC) when its time tag is reached. The alternative methods to control the SPC executions are (1) use of the presently implemented continuous clock with control of SPCs, using a realtime command "SPC Inhibit Release" sent by the STS crew, or (2) addition of new clock controls "Reset," "Hold," and "Release" to be activated by the STS crew at its discretion.

In the "SPC Inhibit Release" method, the ECS clock is running continuously and all stored command sequences are preceded by an SPC Inhibit command. This command prevents any SPCs from being executed at the time designated by their individual time tag unless the realtime command (RTC) designated "SPC Inhibit Release" is sent prior to the time tag of the first command in the stored sequence to be executed.

If this time window is missed for any reason, the entire block of commands for that particular sequence must be erased and reloaded with new time tags. The clock control method requires that the programmer clock be "reset and held" (stopped) and then released when the sequence is to be executed. Upon clock release, the stored commands would be executed according to their predetermined sequence.

Although hardware modification is necessary, the controllable clock method was selected because of its flexibility. This clock control capability is also required on the backup command system (BCS).

5.7.1.2 Command Security. Since the ECS and BCS clock controls the execution of all events in the mission, it was considered mandatory to assign the clock control commands as secure. The security for these commands prevents any inadvertent activation of the clock controls by any source except the STS crew or SCF. This concept requires the implementation of a secure realtime command capability in the command systems. This change is considered a minor modification.

The secure realtime capability and the increased length of the mission requires the use of a greater number of secure words in the ECS. The increased capability may be obtained by (1) increasing the size of the ECS memory to hold the necessary secure words, or (2) modifying the ECS software so that the ECS would generate its own secure words without limit. The secure word generator was selected for this configuration, because the software, which has already been developed on another program, is less expensive than new hardware, and the need is eliminated for ECS removal from the vehicle for secure word loading during refurbishment. This capability is also required in the backup command system.

5.7.1.3 Backup Command System. As shown on Table 5.7-2, 39 additional minimal command system (MCS) commands in excess of Block III capability are required for the new STS-compatible design. This fact, coupled with the realtime secure command requirement as described above, requires the incorporation of one of the following:

1. A larger MCS
2. More MCSs
3. One-half of an ECS
4. A general-purpose computer

Use of more minimal command systems was the approach taken in the previous study (Minimum Modification). While this approach is probably the least costly, it adds to system complexity. It is also costly in terms of weight, and there is no improvement in reliability. While adding one-half of an ECS meets the capability requirements, it is more costly, has a greater systems design impact, and results in an increase in weight. A larger MCS seems like a reasonable approach from the standpoint of cost and design impact. However, an investment for scaling-up a 10-year-old design (and technology) with no reliability improvement seems rather inappropriate for this study. A general-purpose computer can be incorporated into the system with minimum design impact. Such computers are in existence today; they were built using state of the art technology and offer high reliability and low weight. A typical computer (such as the CDC 469) could be programmed to emulate the existing MCS and would reduce the design impact on the rest of the system. The fact that the development cost of incorporating a general-purpose computer may be higher than the cost of the other options is outweighed by the above-stated reasons, and the GPC was selected to serve as a backup command system in this configuration. As with the MCS, this system will utilize a remote decoder. The backup command system will be configured with a capability of 256 commands, which provides 131 spares for future growth.

5.7.2 Instrumentation

Table 5.7-3 presents the functions identified for the caution and warning instrumentation system. These functions will be independently instrumented and redundant to the subsystem monitors. The system will be powered by an independent instrumentation power supply (redundant) in the caution and warning instrumentation J-box. Also in the J-box will be the signal conditioning necessary to drive the displays at the Mission Specialist Station. Approximately 80 wires (70 instrumentation points and 10 returns) will be required. These instrumentation points will be monitored on the SV PCM telemetry. Increased monitoring (as shown in Table 5.7-3), is provided on the PCM for additional diagnostic information.

Table 5.7-3

CAUTION AND WARNING SYSTEM FUNCTIONS

	Hardwired	PCM
Solar Array		
Caged/uncaged	2	4
EDAP		
Pyros safe/arm	2	4
Main bus voltage and current	2	2
Pyro buses 1 and 2 current	2	2
Main transfer sw ON/OFF	1	2
Pyro buses 1 and 2 ON/OFF	2	4
Lifeboat bus voltage and current	2	2
TT&C		
ECS/BCS clock status	2	4
Caution and warning inst pwr	2	2
Propulsion		
Repressurization tanks temp and press.	6	6
Orbit adjust tank temp and press.	2	2
Lifeboat tank temp and press.	2	2
Propellant pyro valve status (NC)	12	42
Thruster isolation valves	3	6
Camera System		
Pneumatic tanks temp and press.	8	8
Pneumatic valve status	4	8
Recovery Vehicles		
Pyros safe/arm	6	12
Pyros safe/arm (backup)	6	12
Payload Bay		
Umbilicals IN/OUT	3	–
Propellant leak detector	1	–
	70	124

The telemetry points required for the STS-compatible SV are listed in Table 5.7-4. An additional 135 points are necessary over Block III. To provide this additional capability in the required SV locations, one thermocouple slice and six analog/discrete slices must be added to the PCM telemetry system remote units. These remote units are presently designed to accept additional slices without modification.

Compatibility with the STS for SV checkout, deployment, and retrieval requires a telemetry format of 16 KBPS. This requirement necessitates a minor change to the existing PCM master units. As shown in Table 5.7-5, two 16 KBPS formats are necessary to process the data required for both vehicle and camera system checkout.

Table 5.7-4

REQUIRED TELEMETRY POINTS

	Block III	Changes	Required
Structures	112	-24	88
Propulsion	81	+48	129
EDAP/SA	165	+8	173
ACS/LB	140	+2	142
TT&C	327	+28	355
Subsatellite	51	0	51
Camera System	212	+10	222
Recovery Vehicles	111	+52	163
MCM	113	-113	0
Caution and Warning	0	+124	124
TOTAL	1312		1447

Table 5.7-5

DATA FORMATS FOR VEHICLE AND CAMERA SYSTEM

	Checkout, Deployment, Retrieval		Camera System Checkout	
	Data Points	BPS	Data Points	BPS
Subsystems	548	14,537	114	8,204
Camera System	–	–	60	6,400
Recovery Vehicles	162	314	–	–
Caution and Warning	70	124	70	124
TOTAL	780	14,975	244	14,728

5.7.3 System Design

The system as designed is presented in Fig. 5.7-1. As discussed in par. 5.2, the interface with the STS is through the umbilical J-box, which provides signal routing and conditioning.

The SGLS-compatible telemetry system provides PCM realtime data at 48 KBPS, engineering evaluation data in realtime at 128 KBPS, vehicle health and status in realtime at 64 KBPS, PCM tape-recorded data (48 KBPS played back at 256 KBPS), and two formats at 16 KBPS for SV monitoring and checkout by the STS. The PCM telemeter provides status for STS operations, ground test operations, normal mission operations, and postflight evaluation. Each tape recorder provides a maximum of 60 minutes of continuous recording of data, including the storage of payload status information and SV operational, attitude, and attitude rate data. STS compatibility requires that the telemetry data be hardwired to the STS for on-board processing or relay to the ground. The hardwired functions will require line drivers to transmit the signals over the coaxial cables connecting the SV and STS. All telemetry system equipment is existing Block III equipment with minor or no modifications.

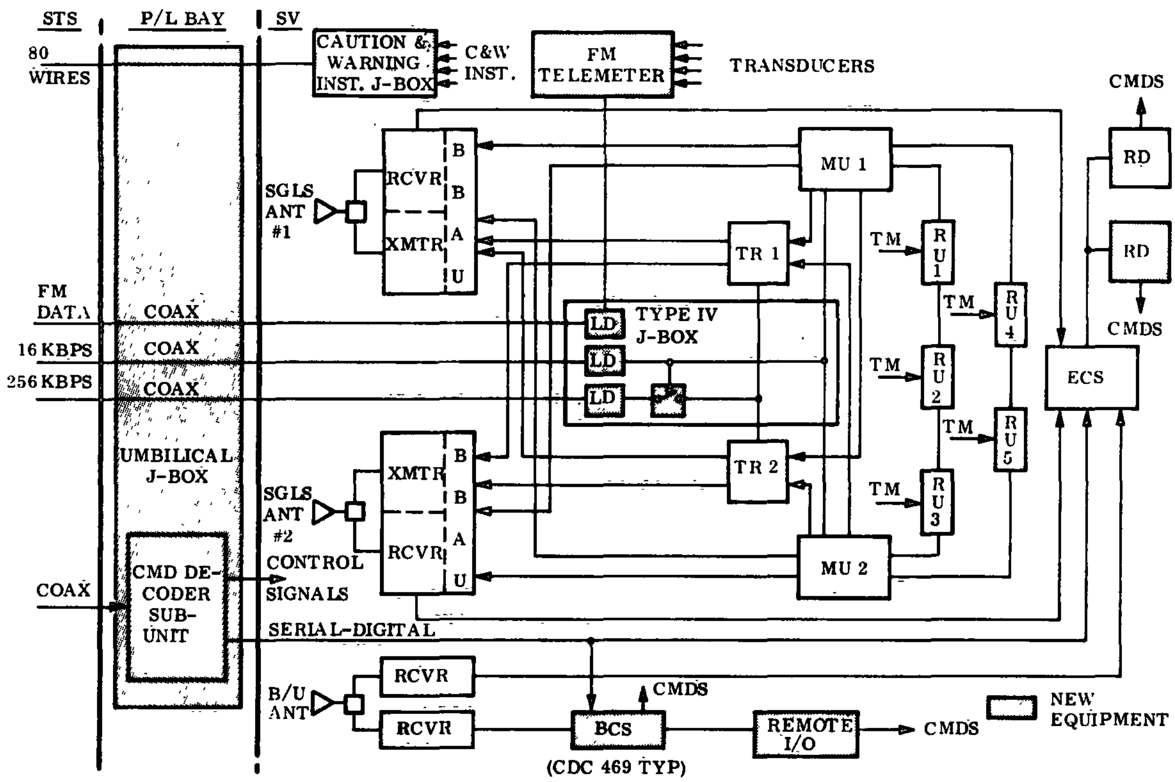


Fig. 5.7-1 TT&C Functional Diagram

An FM/FM data system will be used to record acceleration and vibration data during ascent and reentry. This system will be essentially the same as that used on the first two SVs of Block I, except that a transmitter and an antenna are not required. A line driver will be used to transmit the composite signal to an analog recorder at the Mission Specialist Station in the STS.

The extended command system (ECS) is compatible with both the SGLS and the backup system receivers, and it is completely redundant, with two dual remote decoders. The ECS provides commands for performing the mission, security provisions for critical functions, and vehicle system time for the PCM and the camera. In addition to the two SGLS inputs, a backup command receiver demodulator is provided as a tertiary command input to the ECS. It is assumed that the present 375 MHz backup command receiver will be converted to an S-band frequency by 1980. While mated to the STS, a hardwire command capability exists in which the STS will provide command inputs in the form of serial-digital data to the ECS for test, checkout, and abort.

An emergency or backup (Lifeboat) command system is included, which provides independent capability for recovering any one of the six RVs, and/or initiating SV retrieval or deboost. Command inputs to the backup command system are through an independent backup command receiver demodulator, which idles in a "sleep" mode until "awakened" by S-bits on the RF carrier. While connected to the STS, input signals will be hardwired serial-digital in the same manner as in the ECS. All backup command system equipment is independently powered by the Lifeboat power bus to provide complete backup capability.

5.8 RELIABILITY

The probability of an SV successfully accomplishing its defined mission over a given time period is the reliability of the SV for that same time period. The success of a multi-flight spacecraft program depends on a high reliability for the operating SV and the capability to refurbish previous flight vehicles to original condition. Refurbishment necessarily involves replacing failed equipment and repairing equipment which is unsuitable for additional flights because of life constraints. This section describes the SV configuration and reliability model that was employed in the probability studies along with some comments on life testing.

5.8.1 Satellite Vehicle Configuration

The present or existing configuration of the Block III Hexagon SV (excluding camera system and RVs) was used as the design baseline. Since this study deals with vehicle orbit operations in the 1980s, an updated or improved Block III configuration was generated. This improved configuration was based upon the following considerations:

- Present hardware design problems resolved by 1980 (e.g., gyros, RCS thrusters)
- Piece part failure rates specified today will be attainable by 1980
- Utilize MSI in certain high-density boxes (e.g., PCM system, flight control electronics)

The improved Block III model also included a second orbit adjust engine and one additional remote decoder associated with the ECS. The Lifeboat system, which operates in a tertiary mode to effect an RV separation or control the SV for retrieval or deboost, was not included in the SV reliability model.

5.8.2 Reliability Analysis

The SV (excluding camera system and RVs) reliability estimate was obtained from an LMSC computer program called "SYEFF", which was derived from an Aerospace Corp. program entitled, "Redundancy Allocation Subject to Constraints". This program was also employed in the portion of the study which investigated the increase in vehicle weight due to additional redundant equipment and the resulting changes in reliability. The SV reliability consisted of the combined reliabilities of the LMSC hardware, ECS, RV separation mechanism (RV reliability following separation was not included), and the camera system. Camera system reliability data was provided by P-E.

The component or equipment failure rates, or probability of survival (where it applies) used in the reliability model of the LMSC hardware (including ECS), are listed in Table 5.8-1. Failures may occur in either the active component or in its redundant standby unit. A dormant failure rate of 10 percent of the active failure rate was employed for components in standby redundancy. The failure rates were assumed to be constant for the orbital lifetime of the SV.

The probability of a successful RV separation was taken to be 0.998 and this was used as a one-shot reliability for each RV. The RVs were assumed to be released at the rate of one approximately every 45 days, or six during nine months. The reliability of the SV (including the camera and RV separation events) to accomplish its assigned mission versus orbit time is shown in Table 5.8-2.

As noted in paragraph 5.8.1 the Lifeboat system was not included in the above calculation. The reliability of the SV (including the Lifeboat system) to accomplish the one-time events of an RV separation, SV deboost, or retrieval by the STS has been calculated to be 0.9992.

The SV mean life, which defines the time during which no catastrophic failure will occur, is the area under the reliability curve, which could be plotted from the data in Table 5.8-2. Analysis of the SV mean life and resulting impact on the program schedules, hardware, and costs were not conducted during this study.

Table 5.8-1

COMPONENT FAILURE RATES AND REDUNDANCY

Equipment or Component Group	Failure Rate (1 x 10^{-6})		Redundancy(1)
	Active	Dormant	
Solar Array Deployment	0.9990(2)	–	N
Battery plus Charge Current Controller	2.2	2.2	A (3 of 4 units)
Power Dist J-Box	2.1	2.1	A
1/2 Quad Valve and J-Box	10.0	10.0	A
Tanks OAS/RCS + Repress. Sys	0.9890(2)	–	N
OA Engine	0.6	0.06	S
SGLS Transponder	17.6	1.76	S
Tape Recorder	18.4	1.84	S
Data Interface Unit	7.2	7.2	A
PCM Master Unit	17.7	1.77	S
PCM Remote Unit 1	5.25	0.525	S
PCM Remote Unit 2	3.78	0.378	S
PCM Remote Unit 3	7.99	0.799	S
PCM Remote Unit 4	5.86	0.586	S
PCM Remote Unit 5	6.74	0.674	S
Forward J-Box	2.0	2.0	A
ECS plus 2 Remote Decoders	46.0	46.0	A
Structures	0.9963(2)	–	N
Pyros and J-Box	1.8	1.8	A
Temp Control Elec Assy	4.0	0.4	S
Thermal Insulation	0.9950(2)	–	N
RCS Thruster Assy 1	5.0	0.5	S
RCS Thruster Assy 2	5.0	0.5	S
RCS Thruster Assy 3	5.0	0.5	S
RCS Thruster Assy 4	5.0	0.5	S
Inertial Reference Assy	26.0	2.6	S
Horizon Sensor Assy	9.97	0.997	S
Flight Control Elec Assy	12.0	1.2	S

(1) A = Active; S = Standby; N = Non-redundant (2) One-shot reliability

Table 5.8-2

SV RELIABILITY

Orbit Time (Days)	Reliability			
	RV (Excluding Camera System and RVs)	RV Separation	Camera System	SV
30	0.9777	0.9980	0.9533	0.9301
60	0.9726	0.9980	0.9087	0.8820
90	0.9645	0.9960	0.8662	0.8321
120	0.9536	0.9960	0.8258	0.7843
150	0.9402	0.9940	0.7872	0.7356
180	0.9245	0.9920	0.7504	0.6882
210	0.9067	0.9920	0.7153	0.6434
240	0.8871	0.9900	0.6819	0.5988
270	0.8660	0.9880	0.6500	0.5561
300	0.8435	0.9880	0.6196	0.5164
330	0.8198	0.9880	0.5907	0.4784
360	0.7953	0.9880	0.5630	0.4424

5.8.3 Predicted Equipment Failures

Equipment failures were predicted for specific components by calculating the expected number of failures. Considering a 10-year program in which a single SV spends 5 years in orbit and 5 years on the ground in refurbishment and testing, expected number of failures were calculated for the orbital period and estimated for the ground period. The results of these calculations are listed in Table 5.8-3, which shows the number of failures expected per vehicle for the critical equipment during the 10-year program.

Table 5.8-3

EXPECTED NUMBER OF FAILURES PER SBA

Equipment	Expected Number of Failures		
	Orbital Period	Ground Period	Total Program (10 years)
SGLS Transponder	3/4	1/4	1
Tape Recorder	3/4	1/4	1
PCM Master Unit	3/4	1/4	1
ECS	2-1/2	1/2	3
Temp Control Elec Assy	1/4	1/12	1/3
Inertial Reference Assy	1	1/4	1-1/4
Horizon Sensor Assy	1/2	1/6	2/3
Flight Control Elec Assy	1/2	1/6	2/3

5.8.4 Life Testing

Life testing for this program can be considered to consist of three phases: operational, orbital, and calendar. Operational life refers to the total operating time, including ground and flight time, with duty cycle considerations. Orbit life refers to the total time in orbit. Calendar life reflects the total useful lifetime, which includes storage. Specific equipment and materials employed in this program will require life testing in one or more of the three phases because of the possibility of mechanical wear and chemical change.

Certain equipment experiences degradation in performance due to extended times of operation. A selection of such equipment should be subjected to both operational and orbit life testing. Suggested components are: RCS isolation valves, RCS thrusters, inertial reference assembly, and horizon sensor assembly. It is probable that with minor design changes and modifications, this testing will demonstrate that most equipment can survive the program life. The one possible exception is the tape recorder, which presently exhibits limited operating life. It is expected that this equipment will require periodic replacement during the refurbishment cycle.

Operational life testing of electronic equipment could be accomplished by performing accelerated life tests on perhaps two representative units. The flight control electronics assembly and SGLS transponder would fit this category.

A representative sample of the non-metallic materials to be used (e.g., plastics, silicones, potting compounds, wire insulations, reaction engine catalyst, etc.) should be subjected to calendar life tests.

Life tests necessarily require long time periods because acceptable accelerated life testing is difficult to formulate. In view of this situation, it is suggested that selective life testing be implemented as soon as possible.

5.9 ASSOCIATE INTERFACES

In this study, only the camera system interface is radically altered. The changes are mainly in the mechanical area to accommodate the larger optical components and film supply. Detailed structural and dynamic analysis of interface loads was not possible, but the limited analysis performed and engineering judgment shows that structural environments are in the same range as at present and do not obviously disqualify any existing equipment. Individual items require detailed review to determine whether existing design and test limits are adequate. New equipment would be designed and tested to the STS-required levels. Operating environments and environmental requirements are essentially unchanged for all contractors, with the exception of the safety requirements. Electrical interface functional changes are limited to providing minimal increases in commands, monitor points, power for new safety hardware, and the additional RVs. Additional changes would be required to define and accommodate the new mechanical layout arrangement. More detailed descriptions are given in the following paragraphs.

5.9.1 RV Contractor

Because of the estimated similarity of loads, the existing RVs, incorporating weight reductions now proposed for Block III, are used unmodified except for the addition of electromechanical safe/arm devices, a reusable umbilical, and modifications for refurbishment. However, full dynamic analysis may show the need to stiffen the retro motor truss as was done in the previous Hexagon SV/STS study.

Additional design studies may show advantages in revising the RV/Forward Assembly attachment interface to be more adaptable to the truss structure proposed. Further interface changes might be required as a result of separation disturbance investigations.

5.9.2 Subsatellite Contractors

No specific subsatellite was prescribed for this study so only general comments are possible. The greater clearance of the SV from the payload bay walls than within the existing shroud could permit much larger subsatellites. SV mounts for the subsatellite payloads will be designed to the new loads. If any existing design subsatellite were considered, its attachments and equipment would have to be reviewed for adequacy with the newly defined loads and safety criteria. Launch of the subsatellite from the SV could be from either the SV deployment orbit or the higher operational orbit.

5.9.3 Camera System Contractor

The major change to this interface is in the mechanical area to accommodate the larger optical components and increased film capacity. The supply unit (SU) was relocated above the optical bars. The SU itself was enlarged and a new mounting concept developed, which uses a statically determinate arrangement as opposed to the present 6 points in a vertical X-Z plane. Induced deformation problems are greatly reduced and the final definition of this interface should be much easier than the present Block III.

A new concept of optical bar mounting was also developed to minimize weight and volume. In the new design the optical bar mounts are attached directly to the SV structure or, for assembly and test, to a similarly configured ground handling fixture. The separate optical bar load-carrying peripheral frame in the present design is eliminated.

Preliminary analysis indicated that deformation of the structure and subsequent misalignment of the optical axis would not be detrimental nor would deformations during ground operations, launch, and landing produce significantly different optical bar loadings. A lightweight platform, attached to the mounts, is incorporated to support and locate the other film path elements and all electronic equipment on one unit for ease of test and refurbishment. Concepts for mounting the enlarged pneumatics supply were not developed beyond the point of providing suitable volume in the structure.

No major functional changes in the existing electrical interface are required beyond extra commands and telemetry for the added two RVs, a possible peak power increase for the larger film spools, pyrotechnic events for pneumatic dump during abort, and a modest growth allowance on total functions. Special diagnostic information for ascent and reentry will require an FM/FM interface.

Section 6
TEST OPERATIONS AND REFURBISHMENT

This section discusses the SV hardware assembly and test operations. Included are the hardware qualification and acceptance test requirements, as well as the pre-launch, post-landing, and refurbishment operations. All levels of test from the piece-part and component to the systems level are discussed in this section.

6.1 COMPONENTS

6.1.1 Development and Qualification

All new Satellite Vehicle components (mechanisms or electronic boxes) and SV-provided STS-mounted components will receive full qualification tests under simulated operational environments. The level of these environments and duration of these tests will normally exceed the expected flight levels and duration in order to demonstrate adequate design margins. These qualification tests will normally be performed on two units of each new component. For new complex components, a development unit will also be manufactured. This unit will normally receive ambient functional tests only, to verify performance against design requirements. However, if one or more environments are critical to the final design, tests will be conducted at those environments on the development units. Software programs will also be developed during the development test program of the backup command computer.

Existing components that have received significant modifications will be subjected to a requalification test. Requalification tests will be similar to those for new components, except that only one unit will be subjected to test.

Other existing components which have not been modified or have received only minor modifications will be subjected to Product Reliability Assessment Tests (PRAT). PRAT tests are designed to detect, early in the production cycle, any weaknesses that may have developed in processes and controls since termination of previous hardware production. Whenever production is stopped for several weeks or more; personnel turnover, changes in equipment, methods, and sources of parts often adversely affect end item reliability. Also, design changes originally judged to be minor may generate unforeseen problems compromising hardware reliability. These tests may include one or more environmental tests at qualification levels.

Component life tests, both operational and calendar, will be required on representative boxes and materials to ensure that equipment wearout or material degradation will not occur during the program life, and/or to identify equipments that may require periodic replacement during the refurbishment cycle. These tests are discussed in detail in Section 5.8.

The components that have been subjected to qualification and PRAT tests can be refurbished, retested, and used as flight spares.

Testing of structural and mechanical components is discussed in Section 5.3

6.1.2 Acceptance Tests

The improved LMSC piece parts program currently in effect for the Block III Hexagon SV will remain in effect for the Hexagon vehicles flown on the STS. This program consists primarily of closer control and examination of parts during manufacture and assembly processes. This program should reduce the number of failures that are currently experienced during tests of higher levels of assembly and should reduce unscheduled refurbishment operations.

Acceptance tests to verify quality, workmanship, and performance of new vehicle components will be essentially the same as that planned for Block III. These tests include room-ambient functional, temperature/vacuum, vibration, and burn-in tests.

Functional tests are performed before, during, and after each environmental test. Mechanical components receive specific acceptance tests, such as leak tests of pressure vessels and hot-fire tests of thrusters. Wire harnesses receive continuity and dielectric withstanding voltage tests. Coaxial cables receive VSWR and insertion loss tests. Structures, in general, receive only inspection for dimension, weight, workmanship, materials, and processes.

Burn-in tests of new components, subassemblies, and electronic piece parts used as replacement hardware during refurbishment will be increased over those currently planned for Block III. The additional burn-in time will result in all replacement hardware having approximately the same operating time at launch as that of new hardware at the time of its initial launch. This should ensure that infant mortality failures of replacement hardware are detected, and that additional operating time on reused components and parts will be held to a minimum.

6.2 MODULES AND SUBSYSTEMS

6.2.1 Development and Qualification

Modules and subsystems which have been significantly modified will require special development and/or qualification tests. Included in this category are the:

- Propulsion subsystem
- Solar array module
- Backup command system module

A development/qualification propulsion subsystem will be assembled and will receive acoustic, electrical functional, and high-pressure leak checks at the factory. This system will then be tested at a hot-fire facility to verify functional performance of the thrusters, engines, repressurization system, and propellant dump system. Tests will also be performed to develop fill, drain, flushing, and purging procedures, and to verify that the system design is proper for adequate flushing and purging. Refurbishment procedures will also be developed during these tests. This propulsion system will subsequently be transported to WTR to proof the fill, drain, flushing, and purging procedures and verify AGE readiness to support the flight vehicle propulsion system.

A development/qualification solar array module will receive electrical/mechanical functional tests and acoustic tests. These tests will be performed primarily to develop and qualify the deployment/retraction and emergency ejection mechanisms. The solar array cells and panels will have previously been qualified.

A development/qualification backup command system module will be assembled and will receive acoustic and thermal/vacuum qualification tests, as well as room-ambient functional tests before and after each environment. The backup command system computer software programs will also be certified during these tests.

6.2.2 Acceptance Tests

The following modules will require module-level acceptance tests.

- Electrical distribution and power #1
- Electrical distribution and power #2
- Attitude reference
- Tracking and telemetry #1
- Tracking and telemetry #2
- Primary command
- Lifeboat guidance
- Backup command
- Solar array

New modules (except solar array) will receive acoustic and thermal/vacuum environmental tests with room-ambient functionals before and after each environment. These tests are similar to those currently planned for Block III SV modules.

New solar array modules will receive an ambient electrical/mechanical functional test, including deployment, retraction, and illumination tests. The solar array modules will be acoustically tested only at the systems level then removed for post-acoustic deployment, retraction, and illumination tests. The solar array tests are similar in nature to those planned for Block III, but the detailed tests are modified to check the retracting mechanisms.

The propulsion subsystem is integral to the SCS and will be tested during the SCS assembly operations. These tests will include an acoustic test, a room-ambient functional, and a high-pressure leak check.

6.3 SATELLITE VEHICLE ASSEMBLY AND TEST

6.3.1 Approach

The approach to the SV assembly and test sequence on new vehicles is basically the same as that planned for Block III Hexagon vehicles, except that the RVs, film path components, and camera system are installed at the systems level of assembly rather than the section level, and the acoustic testing is conducted without a shroud installed. These changes in assembly sequence are due to the vehicle design concept where the Block III Mid and Forward section structure is now an integral assembly, the camera system has been modularized for ease of removal and installation during refurbishment, and the shroud is no longer required to protect the SV during ascent.

The SV functional design changes are not significant enough to warrant special systems-level development tests using a development vehicle. Any development tests required can be conducted at the module or subsystem level, as described earlier, or during the early phases of the systems qualification test program.

Requalification of the SV is required due to the structural redesign, relocation of components and modules, and a new thermal design. The approach to the systems qualification test program is to assemble the first vehicle of flight-qualified hardware that has received the normal acceptance tests. This vehicle will receive systems qualification tests and will then be shipped to WTR, where a series of tests will be performed to verify compatibility of the AGE, facilities, and STS, as well as to proof the procedures to be used on the first flight vehicle. The SV will then be refurbished after qualification, and flown as flight vehicle 2. Adequate instrumentation will be installed during the test program to verify that component qualification levels have not been exceeded.

The second vehicle assembled will receive the normal acceptance tests, shipped to WTR, and flown as flight vehicle 1. This approach is shown in the Program Schedule Summary (ref. Fig. 2-10).

To support the systems-level test program and refurbishment schedules, three interchangeable camera systems are required, as depicted in Fig. 2-10. To support these camera systems during test and refurbishment at P-E, three support structures (consisting of a portion of the SV Forward Assembly structure) will be supplied to P-E. The camera system and supply unit will be shipped to LMSC as separate units. This allows use of the existing Mid Section transporter, with minor modifications, for shipment of the camera system. The supply unit will be shipped in a P-E-supplied shipping container.

6.3.2 SV Assembly and Test Description

The SV assembly and test program for new vehicles is shown in Fig. 6-1. This flow is representative of both the qualification and acceptance test programs. Special tests (pyrotechnic shock and EMC) required on the first vehicle for qualification are identified. The following paragraphs provide a brief description of these activities.

6.3.2.1 SV Mating. The propulsion system components will be installed into the SCS structure and tested as described in paragraph 6.2.2. The SCS will then be assembled with harnesses, components, and modules which have completed acceptance tests. It will then be moved to the Vertical Integration Stand (VIS) in the clean room area.

The Forward Assembly structure will be assembled with LMSC and P-E harnesses and electrical components which have completed acceptance tests. Mating to the SCS will be accomplished in the VIS.

6.3.2.2 Forward Assembly Buildup. The SV will be moved to the horizontal position and placed on the SV dolly for Forward Assembly buildup activities. The film path articulators and the RV, which have completed Receiving Inspection tests following shipment from the associate contractors, will be installed. After installation of these components a complete functional, including takeup operation, film tracking, film transfer,

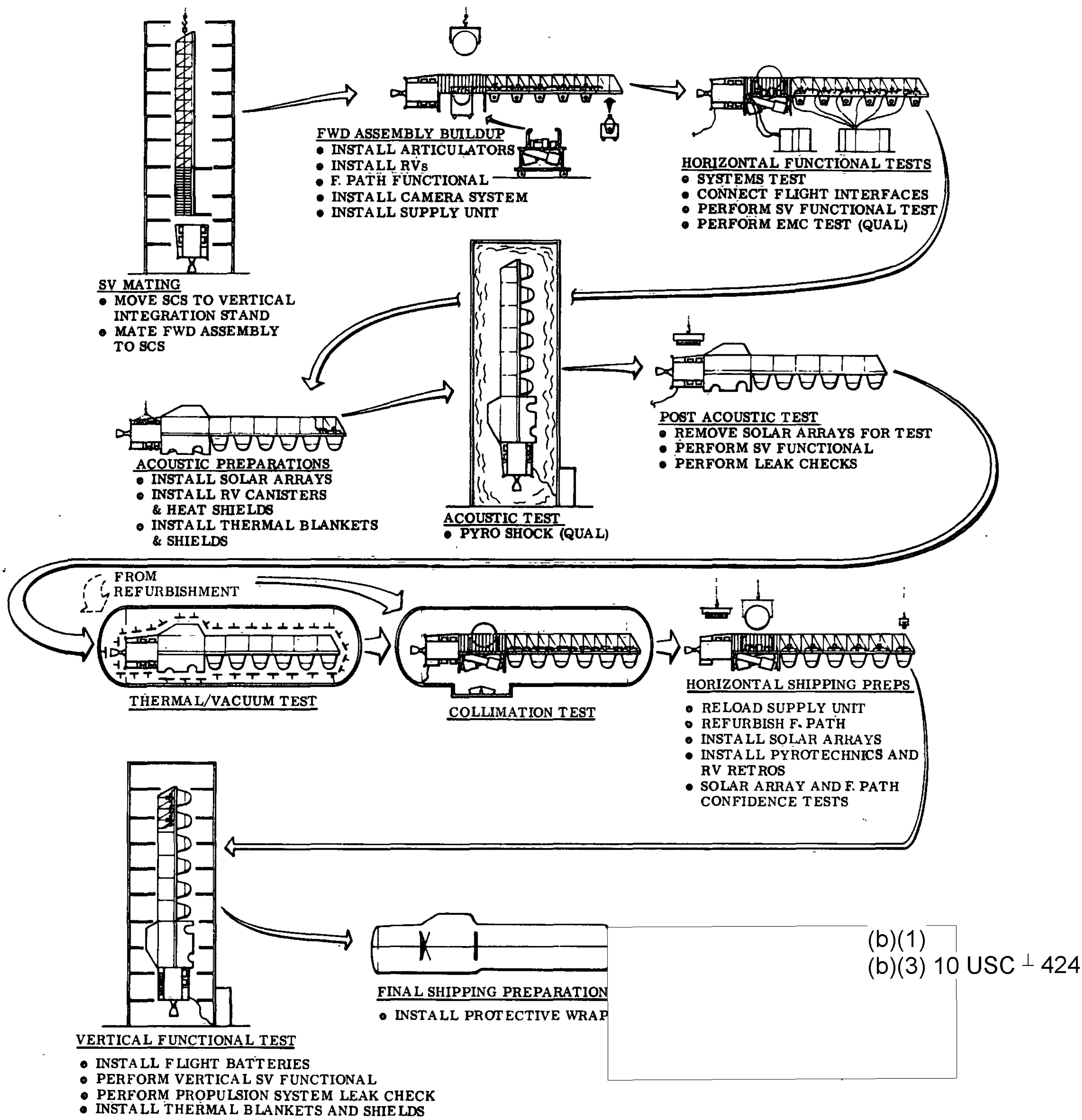


Fig. 6-1 SV Assembly and Test Flow

and film cut and wrap operations will be conducted, using a supply unit simulator with test film. The camera system and supply unit will then be mechanically installed and the film paths mated. Alignment of the camera system and attitude reference module will be verified.

6.3.2.3 Horizontal Functional Tests. Prior to connecting the electrical interfaces between the LMSC hardware and the associate contractor hardware, an electrical functional test will be performed using simulators. This test is designed to verify that all interface functions are correct before connecting the hardware. After the interfaces are connected, a complete electrical/mechanical functional of the SV system will be conducted to verify functional integrity and film tracking prior to environmental exposure. A special test will be conducted on the first vehicle to establish electromagnetic compatibility of the SV assembly.

6.3.2.4 Acoustic Preparations. Following the horozontial functional tests, solar array modules, RV canisters and heat shields, and thermal blankets and shields will be installed to configure the vehicle to an ascent condition for acoustic testing. Special instrumentation will be added on the first vehicle to record vibration and shock levels at the components and modules.

6.3.2.5 Acoustic Test. An acoustic test of new SVs will be conducted to verify that the vehicle will perform its required functions during and after exposure to the ascent environment. Acoustic excitation levels for both qualification and acceptance will have been previously established during the dynamic test program. The duration and levels of the acoustic exposure for qualification testing on the first vehicle will be greater. During acoustic exposure, the vehicle is in a power-on ascent configuration, with the vehicle health continuously monitored.

Following the acoustic tests on the first vehicle, pyrotechnics which may present a critical pyrotechnic shock environment will be installed and detonated to verify that no hardware is susceptible to this environment.

6.3.2.6 Post-Acoustic Functional. The vehicle will be transferred from the acoustic cell to the functional test area, where it will receive a complete electrical/mechanical functional test, including a propulsion system leak check. This will verify whether hardware failures have occurred or leaks developed during the acoustic or pyrotechnic shock tests. The solar array modules will then be removed and returned to the solar array laboratory for deployment, retraction, and illumination tests. The SV will be instrumented with thermocouples and calorimeters to prepare it for the thermal/vacuum test.

6.3.2.7 Thermal/Vacuum Test. The thermal/vacuum test is conducted to verify that the vehicle performs its required functions during exposure to the orbital environment. This test on the first new vehicle will be of extended duration and will include a thermal design verification test. The acceptance test duration for the second vehicle will nominally be ten days at vacuum conditions. Vehicle power will be on, with simulated orbital sequences and comprehensive functionals conducted throughout the test.

6.3.2.8 Collimation Test. A collimation test of the camera system will be performed on all new vehicles to verify that optical performance is satisfactory after exposure to the acoustic, thermal/vacuum, and ground handling environments. The test will be conducted in the A-2 collimation chamber at vacuum conditions. Prior to the collimation test, the thermal blankets and shields are removed to provide access. Film will be retrieved and analyzed following the test sequences.

6.3.2.9 Horizontal Shipping Preparations. The horizontal shipping preparation activities include:

- Hardwire tests of the RVs
- Retrieval and analysis of the film exposed during previous testing
- Flight film loading of the supply unit
- Installation of the RV internal pyrotechnics, retro motors, canisters, and heat shields
- Installation of the solar array modules
- Installation of SV pyrotechnics

Vehicle tests are performed during these activities to verify proper operation of the camera system after cleaning and flight film loading operations, and to verify the solar array module electrical interfaces with the vehicle. A surefire test of the SV pyrotechnic circuits is also performed prior to pyrotechnic installation.

6.3.2.10 Vertical Functional Test. A final ambient functional test is performed to verify readiness of the vehicle for shipment to the launch base. This test will also serve as a baseline for the launch base systems testing. The test includes a comprehensive functional test of all vehicle subsystems and a limited functional test of the camera system.

Following completion of this test, the vehicle will be readied for shipment to the launch base. These pre-shipment activities include:

- Propulsion system final leak check
- Installation of thermal blankets and shields
- SV weight and center of gravity determination
- OA engine alignments

6.3.2.11 Final Shipping Preparations. Following the activities described above, a protective wrap is installed on the SV for environmental and cleanliness protection during shipment to the launch base. The vehicle is then transported to the launch base [redacted]

(b)(1)
(b)(3) 10 USC 424

6.4 LAUNCH OPERATIONS

The WTR launch site, as described in Section 7, consists of an STS launch pad with a Missile Service Tower (MST) and appropriate supporting AGE. A clean room, which includes an SV test area and an SV/STS mating area with interconnecting doors, is located in the MST at the STS payload bay level. Most of the SV pre-launch tests and preparations will be conducted in the test area independent of the STS. At the conclusion of these pre-launch activities the SV is installed into the vertical STS for final preparations, countdown, and launch.

The vertical SV/STS mate concept was chosen to minimize impact and dependence on the STS during the prelaunch phase, and to reduce the impact on the STS of possible pre-launch SV aborts. It also reduces security problems and avoids environments associated with STS ground handling. The SV pre-launch activities are essentially the same as those planned for the Block III SV/SLV program. The following subsection discusses the SV operations at the launch base under the vertical mate concept.

6.4.1 SV Launch Operations

The launch base flow, shown in Fig. 6-2, begins with SV transporter arrival at the launch site. The SV, with the protective wrap installed, is hoisted into the MST clean room and positioned on a test stand. The air conditioning is attached to the SV to maintain SV cleanliness and temperature until the clean room environment is established, after which the air conditioning umbilical is disconnected and the protective wrap removed. The SV is then ready for the pre-launch systems test phase.

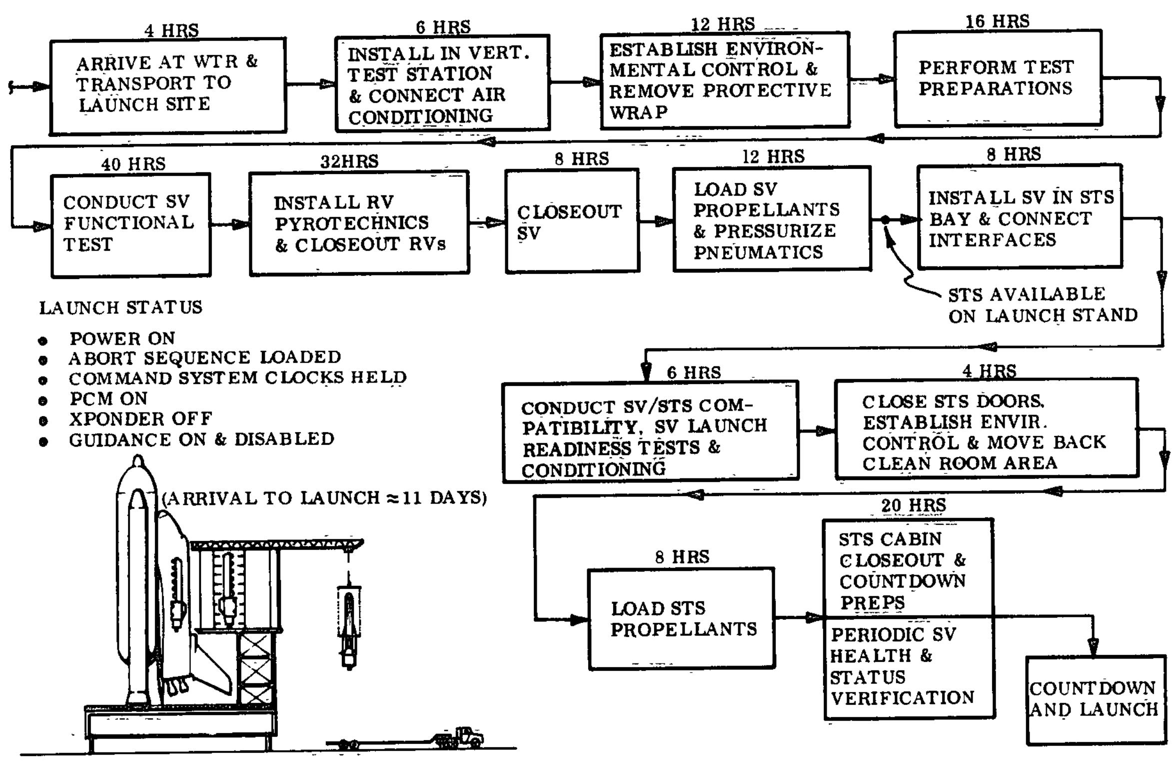


Fig. 6-2 Launch Base Flow

Test preparations include the connection of the umbilicals, test AGE, and charging of the batteries. The SV functional test will be the same as the last test performed at the LMSC factory. This includes a comprehensive functional test of all SV subsystems and a limited functional test of the camera system. Included within this test is a dry run of the SV/STS compatibility, SV readiness, and countdown sequences to ensure that these test sequences (to be used after SV/STS mate) are free of procedural errors.

Upon satisfactory completion of these tests, the SV pre-launch preparations are performed. These include the following:

- RV pyrotechnic installation, checkout, and arming. This entails installation of those high hazard pyrotechnic devices that were not installed at the factory (such as retro motor initiators) and arming of all RV pyrotechnic circuits.
- Reinstallation of thermal blankets and shields.
- SV propellant loading and pneumatic pressurization. This includes the OAS/RCS, Lifeboat, and camera system. To reduce the safety hazards, the OAS tank is not pressurized to flight pressures during ground operations. An on-orbit repressurization system was incorporated as described in Section 5.5.

When these activities have been completed the SV is ready for installation into the STS. Under the vertical mate concept, STS arrival at the launch site is not required until this point in the SV launch base sequence.

Based on the current launch base timelines, the SV pre-launch activities, up to SV/STS mate, will take approximately 8-1/2 days. The time from SV/STS mate to launch is approximately 46 hours, based on the information supplied by Aerospace Corporation during the study, and on estimates of the final SV pre-launch activities.

During STS refurbishment and pre-launch activities prior to transportation of the STS to the launch pad, the special payload-provided umbilical mechanisms, harnessing, and Mission Specialist equipment will be installed. The installation of this equipment will be validated, using the payload-provided STS interface test unit described in paragraph 7.1.3.

When the STS is ready to receive the SV, the mating area is secured around the STS payload doors, this area is cleaned, and proper environment is established. The doors between this area and the SV test area are opened, then the payload bay doors are opened and the SV is installed into the bay. The mechanical attachments are secured, umbilicals connected, and the vehicle installation equipment removed. The system is now ready for SV/STS compatibility and final readiness tests before launch.

The SV/STS compatibility test will verify all interfaces with the SV and will verify the STS operational capability to control and monitor the SV. The readiness test will consist of a limited confidence test and will verify final health of each SV subsystem. A dry run of the on-orbit pre-deployment test will be conducted to verify procedures and software. This test will be similar in scope to the countdown test currently planned for Block III Hexagon vehicles.

At the conclusion of these tests, the STS payload bay doors will be closed and the internal bay environment established by payload bay air conditioning. The MST will then be moved back clear of the STS. The STS propellants are then loaded and STS launch preparations completed. During these launch preparations periodic health and status checks are conducted on the SV. At the conclusion of these activities, the SV command system is loaded with the stored commands necessary to effect an abort during ascent. The STS skin umbilical, carrying SV functions, can be removed at this time. Final systems health is then verified and the STS is launched. The SV battery water cooling may be terminated a short time (less than 1 hour) before liftoff.

The qualification test vehicle/launch base compatibility tests will be similar to those described above for the first flight vehicle except that: (1) pyrotechnic simulators rather than live pyrotechnics will be used, (2) propellant loading and pneumatic pressurization will not be conducted (this will have been demonstrated earlier using the development/qualification propulsion system), and (3) a special SV/STS electromagnetic compatibility test will be performed.

6.4.2 SV Post-Flight Launch Base Operations

Following retrieval of the SV, reentry of the STS, and landing of the STS at the launch base, certain operations are required to ready the SV for shipment to the factory. These operations place the SV in a safe mode for transportation and condition it for receipt into the factory clean room facilities. Figure 6-3 depicts the post-landing flow.

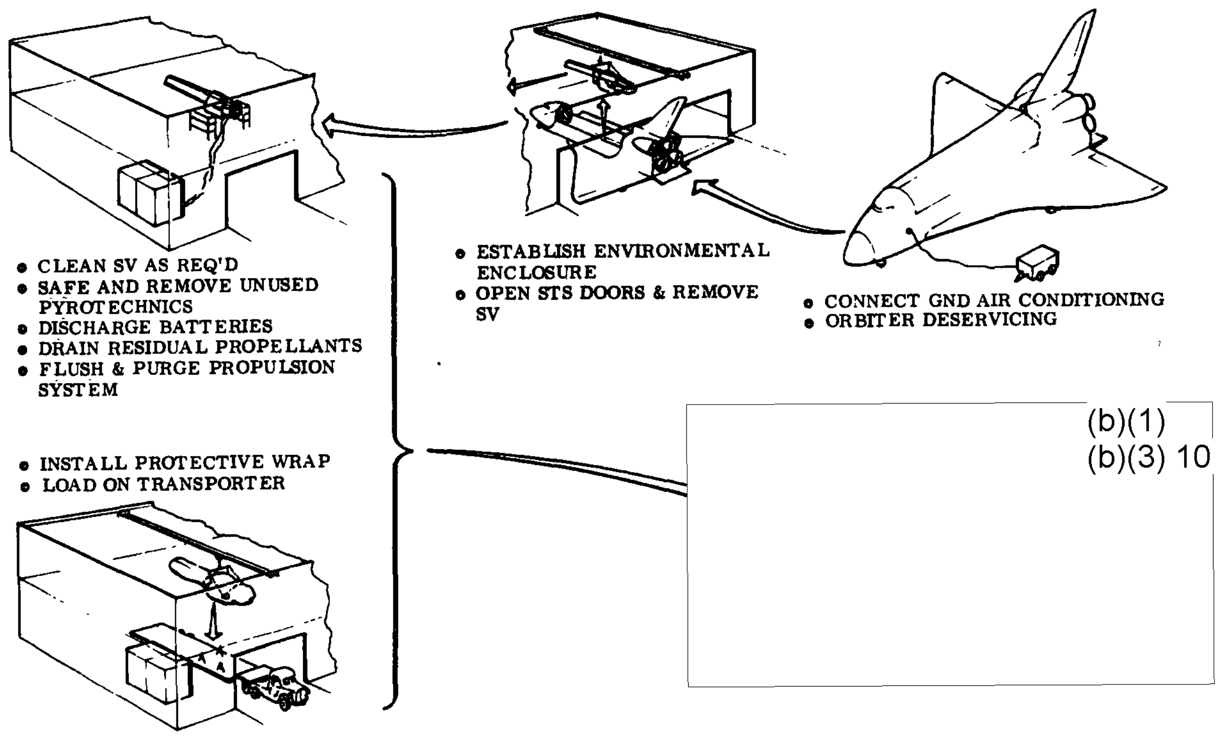


Fig. 6-3 Post-Landing Flow

As soon as possible after landing, the ground air conditioning must be connected to the Orbiter payload bay to prevent SV equipment temperatures from exceeding specification values. After Orbiter deservicing operations, the Orbiter is towed to the Orbiter maintenance and checkout facility for SV removal. Once inside the demating area, the external surfaces of the payload bay doors are cleaned and a flexible enclosure is lowered from the overhead clean room. The enclosure is attached and sealed around the edges of the payload bay doors. The environment is then established to the required cleanliness levels and the payload bay doors are opened.

The SV horizontal handling AGE is attached to the SV and the SV is released from the STS attach points and umbilicals, then removed from the payload bay. The SV is then positioned on a horizontal workstand located in the clean room where the shipping preparations are performed. These shipping preparations include:

- Cleaning the SV external surfaces of orbital contaminants to an acceptable level for entering the factory facilities.
- Removing any unexpended pyrotechnics and arm plugs that are normally installed at the launch base. These may include RV retro motor initiators if one or more RVs have not been recovered.
- Discharging the secondary batteries to maintain safety. These batteries will be returned to the factory with the SV for use as test batteries during refurbishment.
- Draining of any residual propellants that have not been dumped and a complete flushing and purging of the propulsion system. (The thrusters may be removed at this time to prevent contamination of unused redundant or Lifeboat thrusters and to facilitate flushing and purging operations.)

Following these activities, the protective wrap is installed on the SV, and the SV is shipped to the factory by aircraft. These post-landing activities, from landing to shipment to the factory, are estimated to take 8 days.

6.5 REFURBISHMENT OPERATIONS

6.5.1 Approach

Experience on the existing Hexagon program has indicated that a large number of test failures occur following activities where significant personnel work and hardware removal/replacements have been performed on the vehicle. To reduce these test problems and to provide maximum confidence in the flight readiness of a refurbished SV, the following objectives were established for the refurbishment concept and SV design.

- Minimize disassembly of the vehicle
- Minimize hardware removals from the vehicle
- Provide the best possible access for work that is required on the vehicle

With these objectives, maximum confidence in maintaining the initial SV reliability goals is realized. In addition, systems-level environmental retest of a refurbished vehicle can be reduced if vehicle disassembly is held to a minimum. This reduces the hardware environmental exposure and operating time during ground testing, which is a prime consideration in long-life vehicles. Additionally, overall refurbishment costs are reduced.

The significant design features that were incorporated to meet these objectives are as follows:

- Truss design of the SV structure to improve personnel access
- Modular design of the camera system to permit easy removal/reinstallation
- Relocation of SCS components to external locations, permitting easy removal/reinstallation if required
- Redesign of the propulsion system to permit flushing, purging, and thruster replacement at the systems level of assembly.

Figure 6-4 depicts the significant operations that are required during the refurbishment cycle. As shown. the camera system requires removal for major refurbishment operations at P-E. This is required primarily by contaminants and wearout of mechanical components. The supply unit is removed for similar refurbishment operations and reloading of flight film. The orbit adjust engines and RCS thrusters require removal and replacement of the catalyst beds. Solar array modules are removed to allow the deployment necessary to inspect and replace solar cells (if required), and to test the deploy/retract mechanisms.

The batteries that were retrieved with the SV will be used as test batteries throughout the refurbishment operations and retests, then replaced with new or refurbished batteries prior to the next flight. New Rvs are required, and the thermal shields and blankets are removed for cleaning and repair and to provide hardware access.

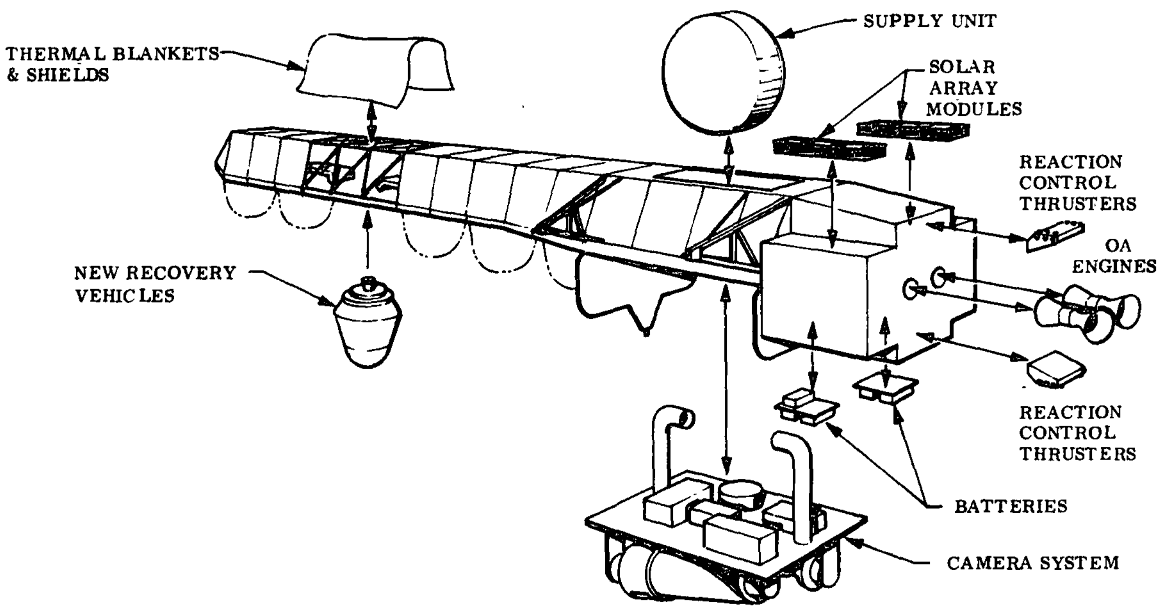


Fig. 6-4 SV Refurbishment Concept

Other hardware removals required during the refurbishment program include:

- Hardware that has failed or is determined to be out of specification during flight or post-flight systems testing
- Hardware that has or will exceed its operating or calendar life by the end of the next flight
- Unused pyrotechnics, pyrotechnic-operated valves, and pinpullers.

If these hardware replacements are minimal, retest of the system can be limited to an SV ambient functional and collimation test of the camera system, then the normal horizontal shipping preparations and vertical functional test. However, time is available for a thermal/vacuum retest of the SV system if extensive hardware replacement has occurred. The collimation test of the camera system can be augmented to exercise the SV systems, thus providing some additional confidence as this test is performed at vacuum conditions.

6.5.2 Refurbishment Description

The SV refurbishment flow is shown in Fig. 6-5. Included is an overall timeline of the major activities to turn around the SV from landing to launch. The unshaded portion represents the timelines for the refurbishment operations at the factory. The following paragraphs provide a brief description of these activities.

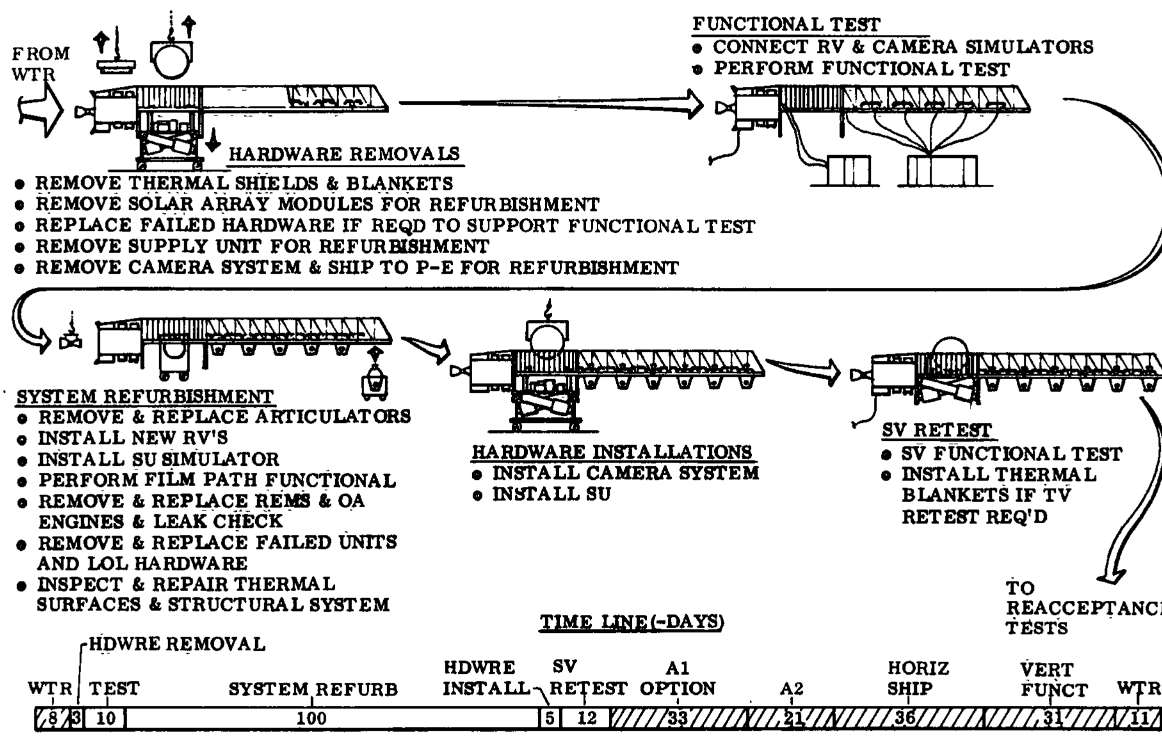


Fig. 6-5 SV Refurbishment Flow

6.5.2.1 Hardware Removals. Following the flushing and purging of the propellant system at WTR (described in paragraph 6.4.2) and shipment of the SV to the factory, major hardware requiring refurbishment at the subassembly level will be removed. These include the camera system, supply unit, and solar array modules.

The camera system is shipped to P-E for refurbishment and retest, then returned to LMSC for installation into the next vehicle to be refurbished. The schedule does not allow sufficient time for return of the camera system to the same vehicle.

The solar array modules are returned to the module area for refurbishment, consisting of a complete electrical/mechanical functional test, including deployment and retraction operations. An illumination test is also performed to verify that solar panel output performance is within specification requirements. These data can be compared each time with prior illumination test data. These modules will be returned to the vehicle during the horizontal shipping preparations. Also at this time, failed hardware (known from orbital operations) that will prevent conducting an SV functional test is removed and replaced.

6.5.2.2 Functional Test. The functional test is performed to identify and troubleshoot any SV hardware failures or out-of-specification conditions that will require repair or replacement during the subsequent refurbishment activities. This test is performed using electrical simulators for the camera system and RVs.

6.5.2.3 Systems Refurbishment. New RVs will be installed to ready the SV for the next flight. Before this can be accomplished the residual RV hardware remaining with the SV after RV separation will be removed and returned to the RV contractor for possible refurbishment. The camera system articulators will also be replaced or refurbished by P-E. Following these activities the SV is ready for RV installation. The RVs are installed, the film path optically aligned, and an electrical/mechanical functional test of the film path is then performed to verify takeup performance, film tracking, and film transfer functions. To accomplish these tests at the SV level, a supply simulator will be connected to the aft articulator to transport film through the system. Hardwire test of each RV will then be required to verify RV performance following installation.

In parallel with the above activity, the orbit adjust engines and reaction control thrusters will be removed and replaced with new or refurbished units and the propulsion system leak checked. The old units will be shipped to the subcontractors for refurbishment. During refurbishment, each unit will be disassembled and rebuilt with new catalyst, injector, valves, and instrumentation as required. A normal hot-fire acceptance test will be performed before shipment back to LMSC.

The hardware on which failures have been identified during the functional test are removed and returned to LMSC manufacturing or to the subcontractor for repair. Any limited operating or calendar life components are also removed, if these limits will be reached by the end of the next flight, and are returned to manufacturing or the subcontractor for refurbishment; however, implementation of the life test program described in paragraph 5.8 should minimize these instances. Components that are repaired or refurbished will receive an acceptance test before reinstallation into the SV. This test will generally consist of an ambient functional, vibration, and temperature/vacuum test.

A detailed physical inspection will be performed on the SV hardware, including thermal surfaces, critical structures, harnesses, and components. All discrepant hardware will either be repaired in place or replaced with new hardware. Structural losses such as nutplates, screws, and fiberglass components are expected. Structural damage due to fatigue and ground handling environments is expected to be minimal. Minor structural damage can be generally repaired in place.

6.5.2.4 Hardware Installations. After completion of the above refurbishment operations, the supply unit, loaded with test film, and the camera system (refurbished from previous vehicle) will be reinstalled, film paths mated, and alignments verified.

6.5.2.5 SV Retest. After reinstallation of the refurbished hardware, a comprehensive electrical/mechanical functional test will be performed to verify functional integrity of the SV including the camera and film path components.

6.5.2.6 Reacceptance Tests. As discussed in the refurbishment approach, reacceptance tests will normally consist of a collimation test of the camera system, horizontal shipping preparations, and vertical functional test. The option is available to perform a thermal/vacuum retest if hardware replacements have been significant. These activities are shown in Fig. 6-1 and described in paragraphs 6.3.2.8 through 6.3.2.10. The SV is then shipped to WTR and processed through the normal launch base operations.

Section 7
AGE AND FACILITIES

7.1 AEROSPACE GROUND EQUIPMENT (AGE)

The AGE consists of the handling, servicing, electrical power, and test equipment required to prepare, test, and launch the SV in the STS. This AGE includes the factory equipment required to handle, service, and fully test the assembled vehicles and SCS modules. It also includes the WTR blockhouse and launch and retrieval site equipment.

The changes are limited to those required to comply with the factory and launch site constraints associated with the modified configuration and operational procedures for launching and retrieving the SV with the STS. The factory changes are generally related to SV configuration differences and principally affect handling equipment. Some factory test equipment changes are required but there is little effect on factory service equipment. The launch site changes are significant because a complete new facility is involved. Consideration has been given to multiple program use of this facility. This is reflected in the mobile concept of launch base AGE, using trailers for testing and service equipment, that can be removed from the area when other programs are using the facility. A remote computer system is also recommended, to minimize the facility requirements. Launch and retrieval handling equipment is based upon a facilities concept that provides a Mobile Service Tower (MST) for vertical pre-launch operations, and a retrieval facility for post-landing operations in the horizontal mode. This concept includes adequate bridge cranes for vehicle handling requirements.

7.1.1 Handling Equipment

Handling equipment is used at the factory and at WTR; some of the same equipment is used at both places. The factory handling equipment changes are due to the new vehicle configuration. The SV structural changes interfere with the use of some of the existing handling equipment. These SV configurational changes, however, provide improved structure for handling the SV. The WTR handling equipment must provide for the SV installation into the STS and for the post-landing operations.

The following paragraphs summarize the new and modified handling equipment:

Vertical/Horizontal Lift Sling. The present vertical lift sling has two cables that attach to the SV handling equipment to lift the vehicle vertically. The new sling configuration will add a cable that picks up the aft end of the SV when it is in a horizontal attitude, and enables moving the vehicle horizontally using the overhead crane.

The vehicle will be tilted from vertical to horizontal by a secondary hoist that lifts the back end while the vehicle is held at the forward STS interface attach points by the vertical/horizontal lift sling.

Horizontal Handling Dolly. The significant reconfiguration of the vehicle, to take maximum advantage of the STS, will obsolete the configuration of the present horizontal handling dolly. The structural integrity of the vehicle, designed for STS compatibility, enables the use of a simple 4-post, air pad dolly for horizontal handling.

Vehicle Holding/STS Installation Fixture. The diminished structural requirement for the SCS resulted in a weight-sensitive design that does not support the vehicle on its aft end. To support the vehicle in a vertical attitude and to hold it securely during installation and removal into and out of the STS, a vehicle holding/STS installation fixture, is required. This fixture will be used in conjunction with a suitable vertical stand in the factory and will interface with the universally applicable installation/removal fixture in the Mobile Service Tower.

Vehicle Protective Enclosure. This enclosure will serve principally as a cleanliness and security cover. It will have sufficient structural integrity to support vertical and horizontal installation and removal. It will be used in conjunction with the horizontal handling dolly to transport the SV in the aircraft and on the transporter.

Vehicle Road Transport System. This existing equipment will be modified to be compatible with the new vehicle configuration and handling requirements.

Forward Assembly Support Structure. The camera system and RVs will be supported by the new Forward Assembly structure. This supporting and handling structure, which serves as a strong back, will be required for handling the assembly in the factory.

Forward Assembly Dolly. The Forward Assembly will be moved and transported on a simple height-adjustable box frame with removable air pads for mobility.

Vehicle Forward Assembly Horizontal Work Access Platforms. These platforms will provide access to upper levels of the Forward Assembly and the fully assembled SV, and will include rolling bridges over the vehicle.

Camera System Dolly. Three dollies will be supplied to P-E. They will be fabricated to represent the portions of the SV Forward Assembly that contain the camera system.

Camera System Transporter. The existing Mid Section transporter can be modified to accommodate the camera system within the LMSC-provided camera system dolly

Camera System Handling Sling. The existing sling will be modified to accommodate the camera system within the camera system dolly.

SCS Handling Sling. The existing sling will be modified to accommodate the SCS.

Vertical Horizontal Removal Sling. The SV vertical/horizontal lift sling will be fitted with a shorter cable installation for use in the Orbiter maintenance and checkout building.

Vehicle Horizontal Removal Access Platforms. Specifically tailored platforms will be required in the Orbiter maintenance and checkout building to enable access to the vehicle in the payload bay, for horizontal removal operations.

Vehicle Vertical Integration Stand – Factory. The existing stand will require platform modifications to accommodate the new SV configuration and the vehicle holding/ STS installation fixture clearances.

7.1.2 Service Equipment

Service equipment is used at the factory and at WTR. The factory equipment changes are minimal, but the WTR equipment changes are significant. The service equipment required at the launch site (Fig. 7-1) consists of a propellant handling system, a pressurization system, and a battery cooling system.

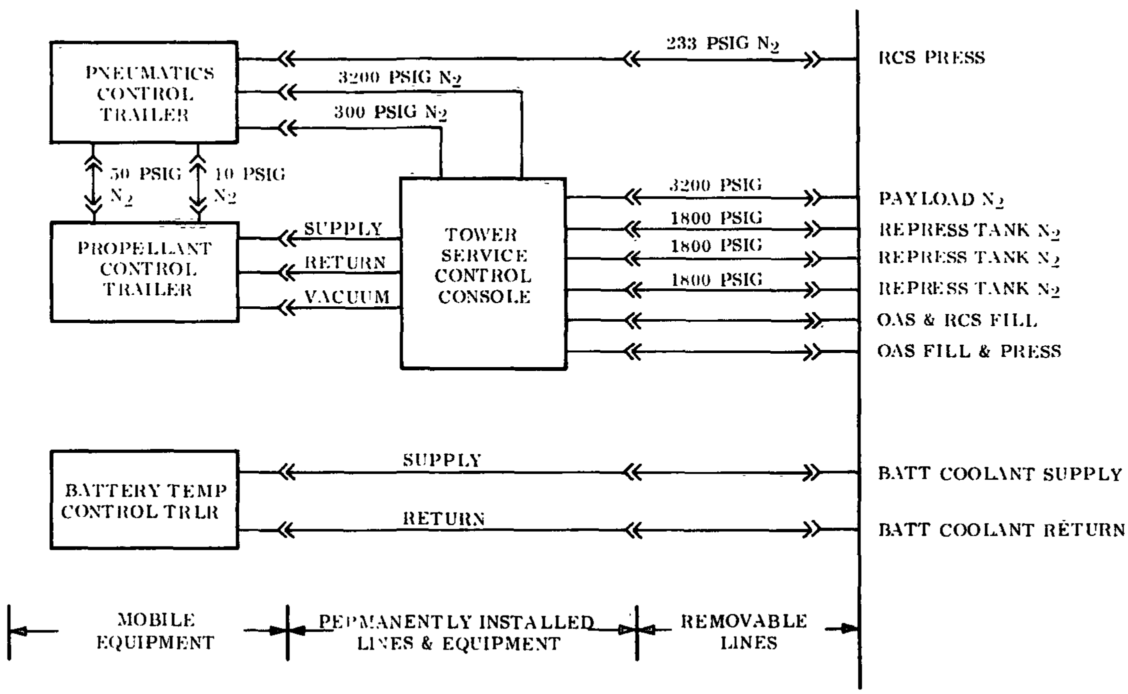


Fig. 7-1 Service Equipment – Block Diagram

The existing launch base service equipment is made mobile by mounting it on semi-trailer units. The service lines and piping are permanently installed on the MST. Electrical power lines supplying the semi-trailer equipment will be permanently installed. Disconnects, similar to those used to connect the lines to the SV, will be provided for attaching mobile equipment. The equipment augmentation and changes are summarized as:

- The propellant transfer unit will be mounted on a suitable semi-trailer assembly.
- The pressurization (pneumatics control) equipment will be mounted on a suitable semi-trailer assembly.
- The battery cooling system will require little if any change because it is already trailerized.
- A new tower service control console will be required at the MST.

A description of the new and modified service equipment is presented below.

Thermal/Vacuum Chamber Heat Flux Simulator. Modifications are required to accommodate the new SV configuration.

Factory High-Pressure Test Equipment. Modifications are required to accommodate the subsystem changes.

Vehicle Removal Area-Flushing and Purging System. A new propellant flushing and purging system will be required at the location where the vehicle is removed from the STS, to ensure a non-hazardous condition during subsequent operations.

Transport Air Conditioning System. A new air conditioning system will be required to provide air at temperature, humidity, and cleanliness that accommodates vehicle needs. This system will move with the vehicle during transportation.

7.1.3 Electrical Power and Test Equipment

Electrical power and test equipment is used at the factory and at WTR. Changes necessary relate to the refurbishment requirements, to the new SV configuration, and to the requirement for new trailerized test equipment. The current retrieval facility operational concept does not indicate a requirement for vehicle electrical power and test equipment. The factory electrical power and test equipment changes and new requirements are:

- Modify the existing module test area equipment to accommodate the new module configurations.
- Modify existing test station and control center equipment to accommodate new umbilical functions.
- Modify the existing test station and control center equipment to accommodate RF equipment differences.
- Provide new terminal equipment to support the telemetry and command data link (leased or GFE) that transmits these data between the launch site equipment and the existing computer at the factory. A computer at the launch site will not be required. Existing computers at the factory will be used to support launch activities.

The launch facility electrical power and test equipment changes and requirements are:

- Electrical power and test equipment of similar existing design will be installed in semi-trailers. This requires four semi-trailers for all contractor's equipment. The RV test equipment will be installed in one semi-trailer and the P-E test equipment will be installed in another. All of the semi-trailers will be removed from the operational locations and used or stored elsewhere when the Hexagon system is not being flown by the STS. This equipment installation is shown in Fig. 7-2; the STS interface test unit and the factory-to-WTR data link interfaces are also shown.
- New permanent cabling and proper terminal consoles will be provided on the MST and between the MST area and the blockhouse area.
- A mobile STS interface test unit will be provided to validate the installation of the SV control and monitor interface equipment in the STS. This console will simulate those SV and AGE functions (umbilical, telemetry transmitters, command receivers, etc.) that interact with the equipment added to the STS to accommodate the Hexagon system.

7.1.4 AGE Testing

In the case of new and modified electrical power and test equipment, and service equipment, validation is accomplished during its first use in flight-quality equipment testing. Handling equipment will be subjected to the normal validation practice of

~~TOP SECRET/H~~/10016

7-7

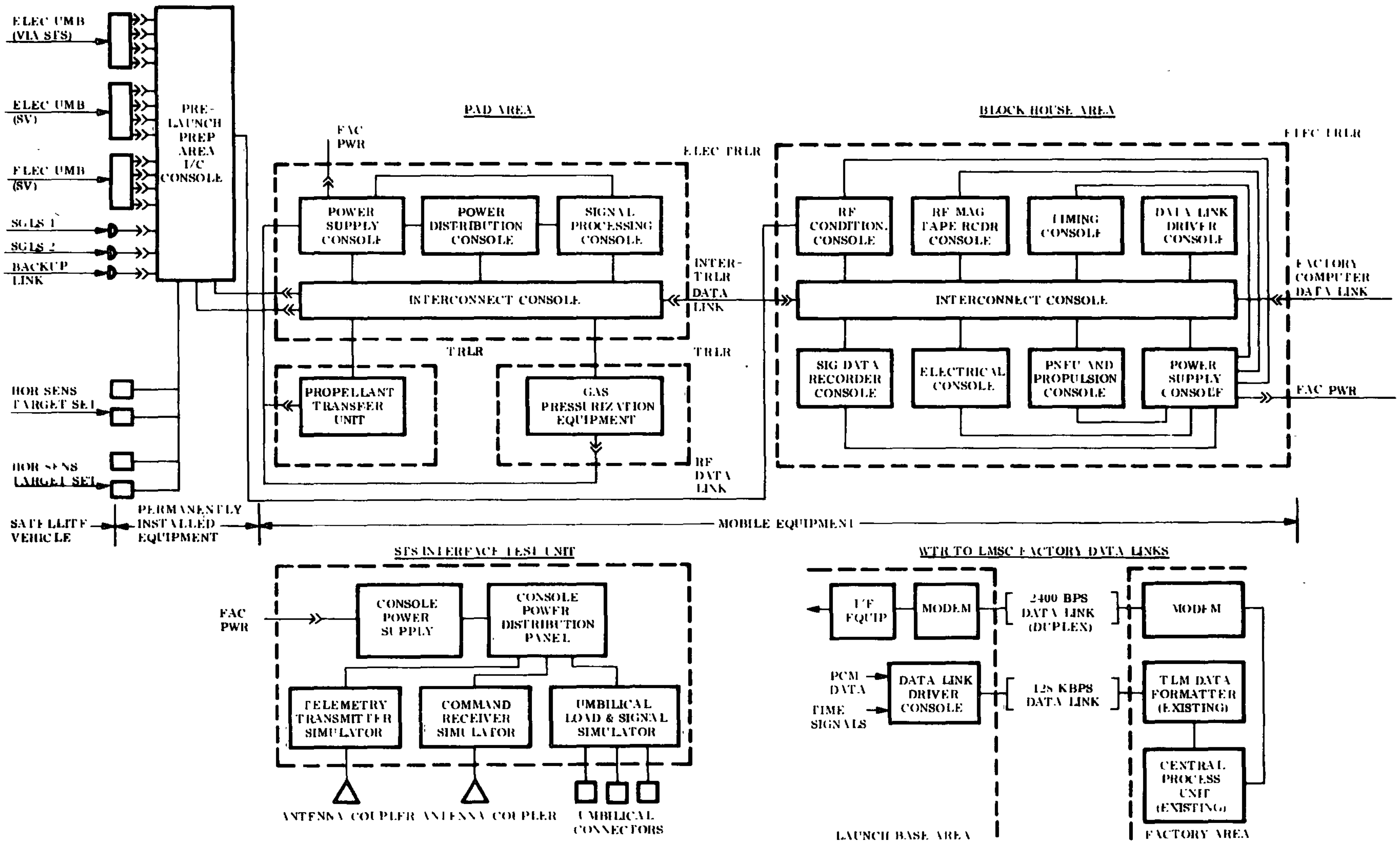


Fig. 7-2 WTR Test Equipment — Block Diagram

~~TOP SECRET/H~~/10016

BIF003W/2-069331-73

"fit checks" and "proof loading" prior to its application in handling operations. WTR AGE validation will require the availability of the launch and retrieval facility for validation tests prior to use of the facility for first flight hardware operations.

7.2 FACILITIES

The study of the factory and WTR facilities was limited to an analysis of impact on the existing factory facilities and a determination of general requirements at WTR. The facilities for WTR are shown in Fig. 7-3 and the LMSC facilities are shown in Fig. 7-4.

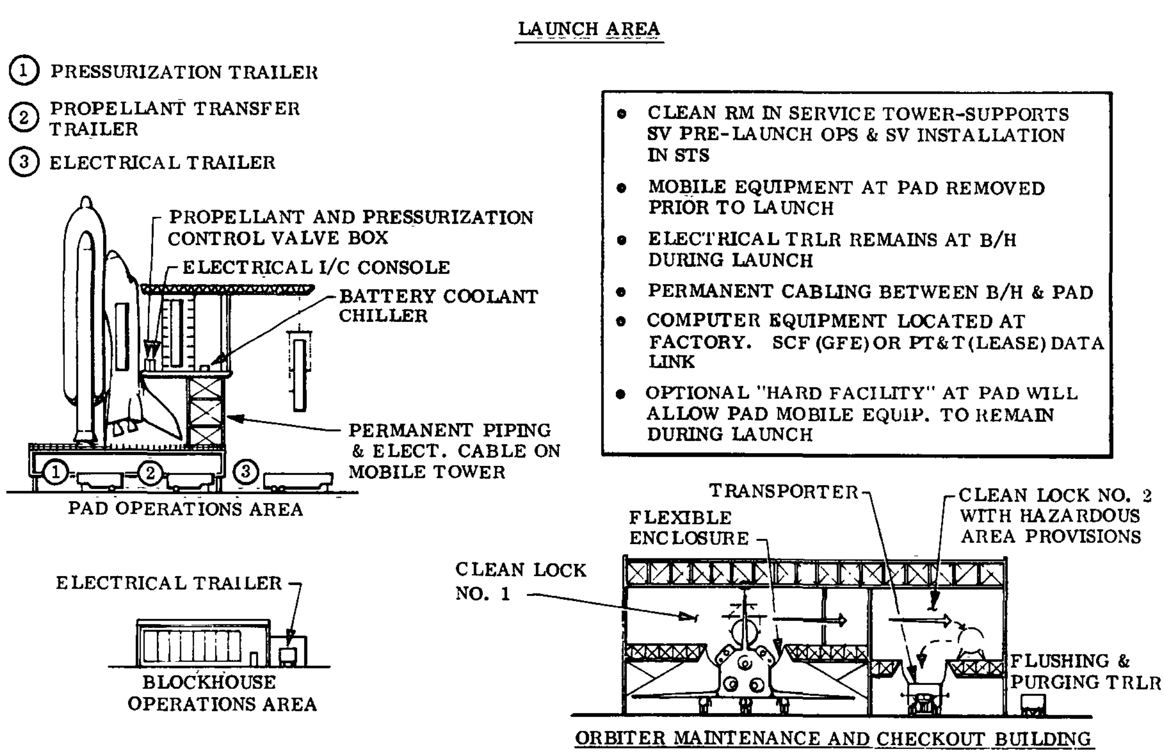


Fig. 7-3 WTR Facilities Requirements

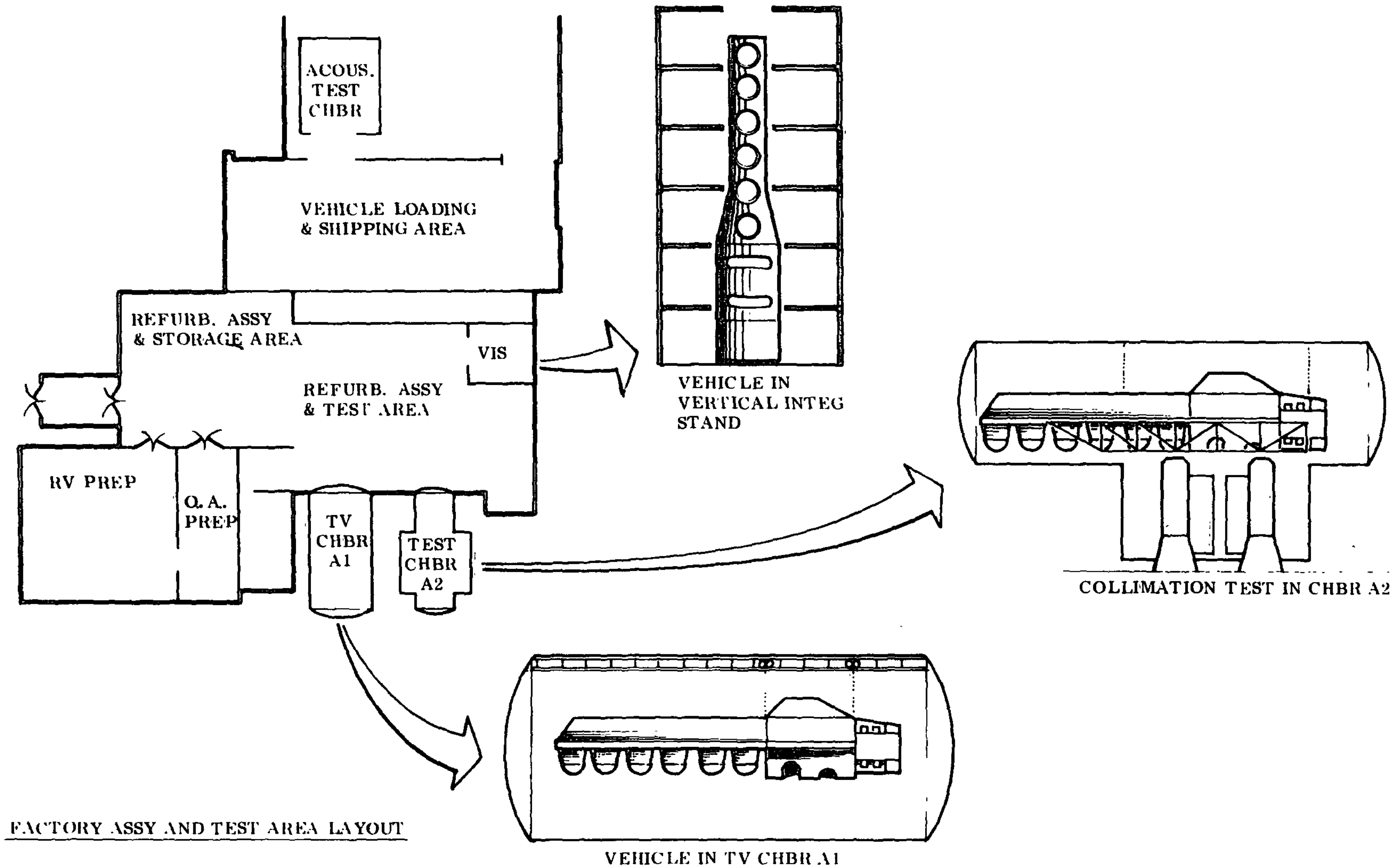


Fig. 7-4 Building 156 Facilities

BIF003W/2-069331-73

Factory Facilities. There are no significant factory facility changes required. In all cases the existing facilities have built-in growth potential that can accommodate the indicated additional traffic.

WTR Facilities. For this study, certain assumptions have been made concerning WTR facilities. The assumptions are generally as follows:

Mobile Service Tower. The MST will consist of a structure that provides an elevated room at the same level as the STS payload bay when the STS is in the vertical (pre-launch) position. It also provides the facilities required to support all pre-launch operations. The room will provide adequate length, width, and height to accommodate the SV, and will be complete with work platforms, utility stands, and storage areas. It will have clean room provisions in keeping with the STS and user cleanliness requirements. This will include enclosure and door arrangements that will enable the SV to be operated in a clean status when the STS area is isolated from it by doors. It will also allow the STS area to be operated in clean status when the SV area is isolated by doors. These doors will be opened when the SV is moved into the payload bay.

SV installation and removal capability will be required for all handling operations. This will include bridge cranes, vehicle installation equipment, and auxiliary monorails as required. A bridge crane is required to support the SV removal from the transporter and installation on a work stand in the pre-launch preparation area. A 5-ton bridge crane or monorail will be required for removal of the protective enclosure and to support other pre-launch operations. The SV/STS installation unit will be required to move the vehicle from the pre-launch preparation area and install it into the STS. Because the configuration of this unit must be universally adaptable to all users, no further details are defined by this study.

A temperature and humidity environment is required that is similar to the current Hexagon factory environment.

Suitable raceways will be required to accommodate the propellant and pressurization piping, electrical power, radio frequency, and control cables that are permanently installed in the MST.

Semi-Trailer Facilities. The concept of using semi-trailers at the launch site for housing the AGE requires that facilities be provided for their use. These facilities will include paved parking areas at the pad and blockhouse, necessary hardening for blast protection, electrical power, communication, and other normal related facilities.

Blockhouse Facilities. Because the testing concept establishes the control computer installation at the factory only, computer terminal equipment will be required at the blockhouse. A minimum number of personnel will require office space and pre-launch and STS launch monitoring accommodations. The actual blockhouse space during STS launch will probably be for a two- or three-man console.

Post-Landing Facilities. The post-landing operations concept requires an integrated facility that will enable the SV and the transporter to be parked in such a position that, without violating the cleanliness requirements, the SV can be removed from the STS, be worked on in a horizontal position, and placed in the transporter. A bridge crane will be required to accommodate the horizontal handling of the SV in the sequence described. Work platforms are required to allow access to the SV in the STS and to enable attachment of the handling equipment required to remove it.

Section 8
SCHEDULES

The schedules presented in this section depict the development of the SV from design through launch, retrieval, and refurbishment cycles. Included are schedules for hardware development and qualification at all levels of assembly. Also included is a schedule showing the LMSC facility loading during transition from the SLV mode of operation.

This schedule shows the major activities and time span allocated to develop, qualify and deliver the Hexagon SVs to accomplish the subsequent refurbishment cycles. Timelines and experience gained from the current and previous blocks of Hexagon systems have been utilized in developing this and succeeding schedules.

The schedule depicts the qualification vehicle being refurbished and launched as the second flight vehicle. The refurbishment and reacceptance spans require that three camera systems be furnished to support the two Hexagon SVs.

The time required for the integrating contractor to develop, qualify, and deliver the first Hexagon SV is 45 months; however, P-E would require up to 68 months and an earlier go-ahead.

~~TOP SECRET/H~~/10116

Approved for Release: 2017/02/16 C05094784

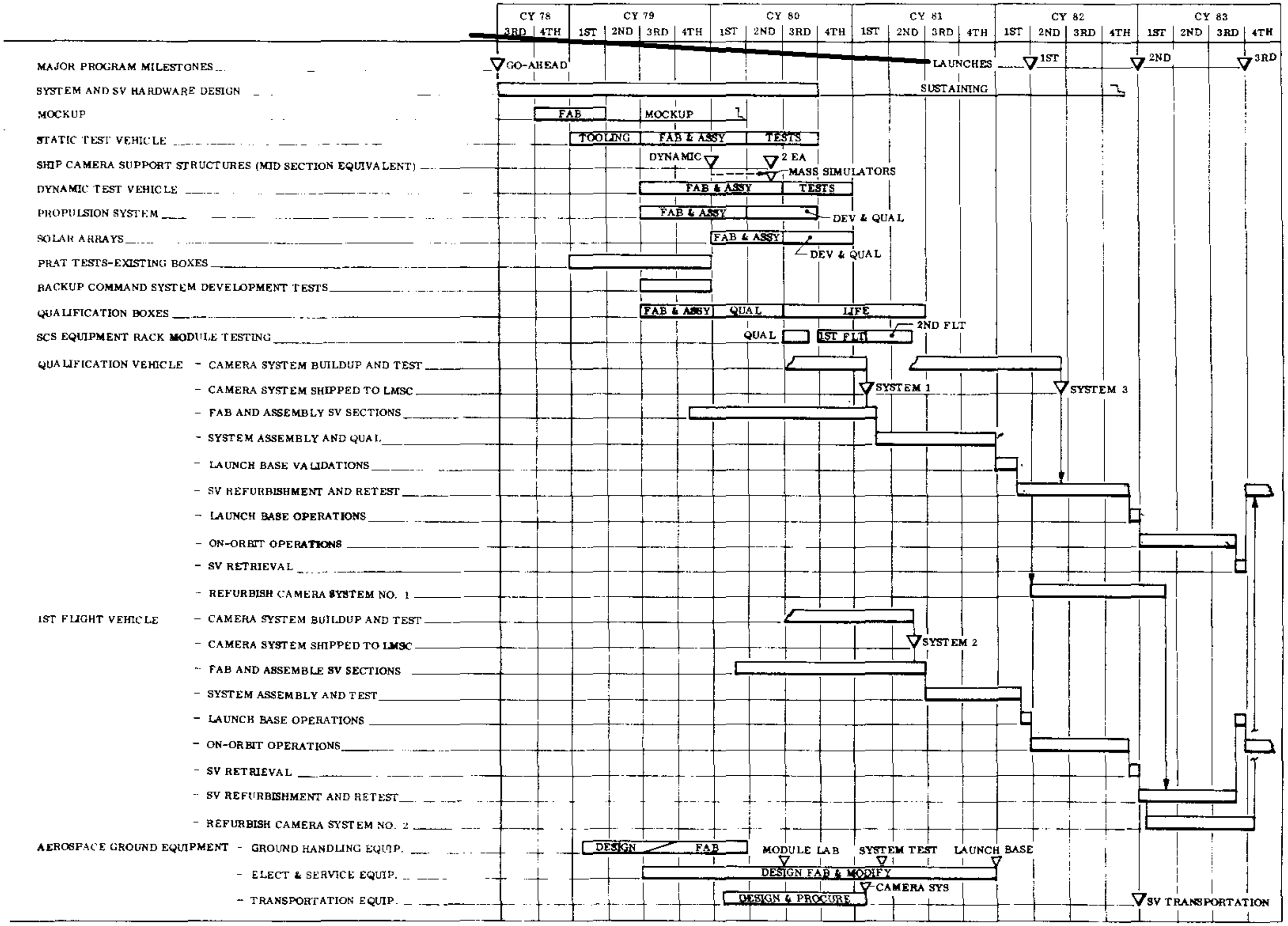


Fig. 8-1 Program Development Schedule

~~TOP SECRET/H/10116~~

8-3

~~TOP SECRET/H/10116~~

BIF003W/2-069331-73

This schedule portrays the sequence of events for a typical R-Day schedule. The STS system arrives at the launch site on R-3 Day and the SV is mated to the STS on R-2 Day.

~~TOP SECRET/H/~~1016

Approved for Release: 2017/02/16 C05094784

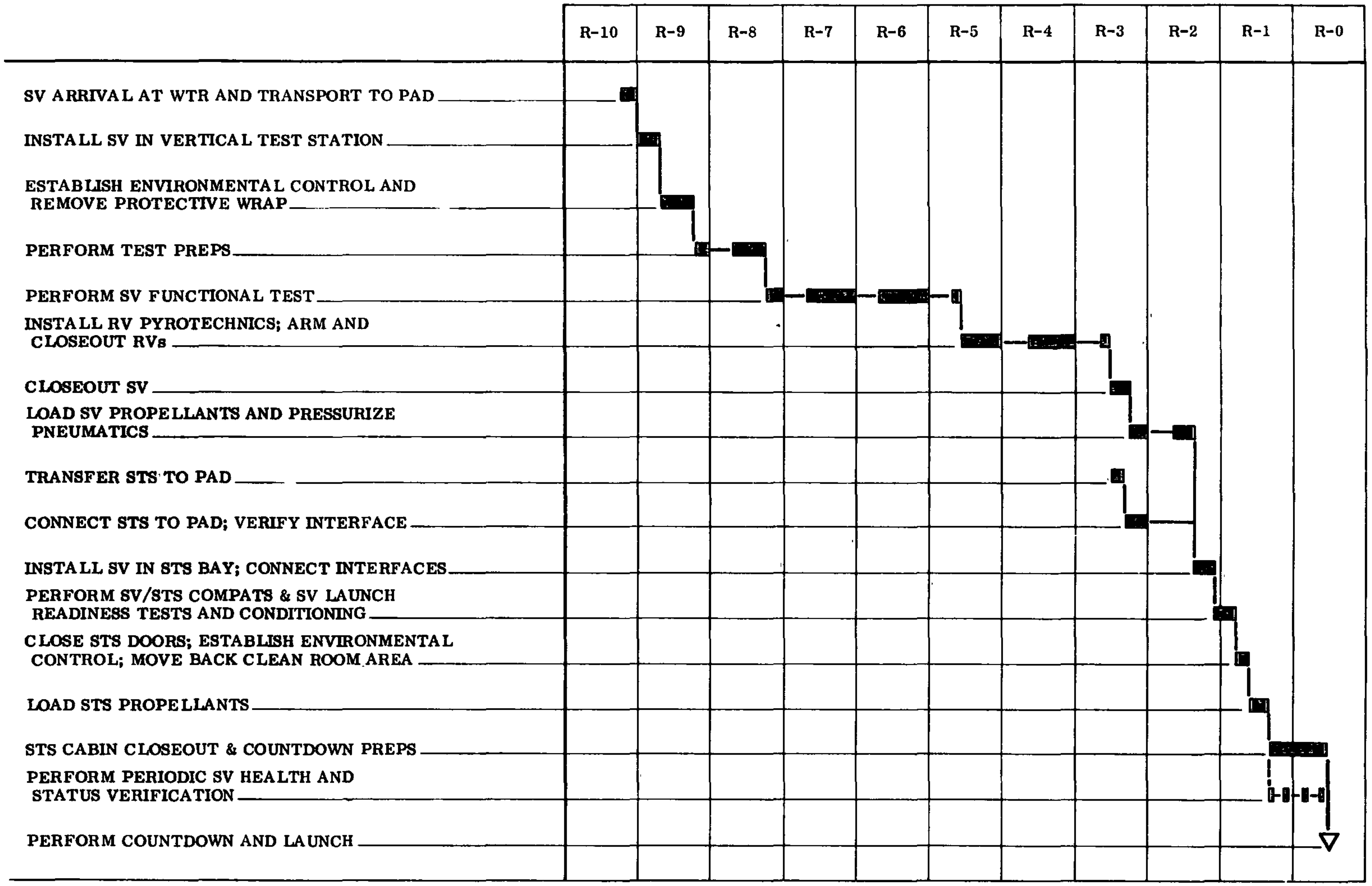


Fig. 8-2 Normal Pad R-Day Schedule

BIF003W/2-069331-73

~~TOP SECRET/H/10116~~

This is basically the same as the normal pad operations except that additional time is allowed for special electromagnetic compatibility tests with the STS. The propellants are not loaded into the SV for these launch base compatibility tests because propellant system compatibility will be previously verified by using the propulsion system development unit.

As shown, the STS is required for a 10-day span for these operations.

~~TOP SECRET//H//~~10116

BIF003W/2-069331-73

Approved for Release: 2017/02/16 C05094784

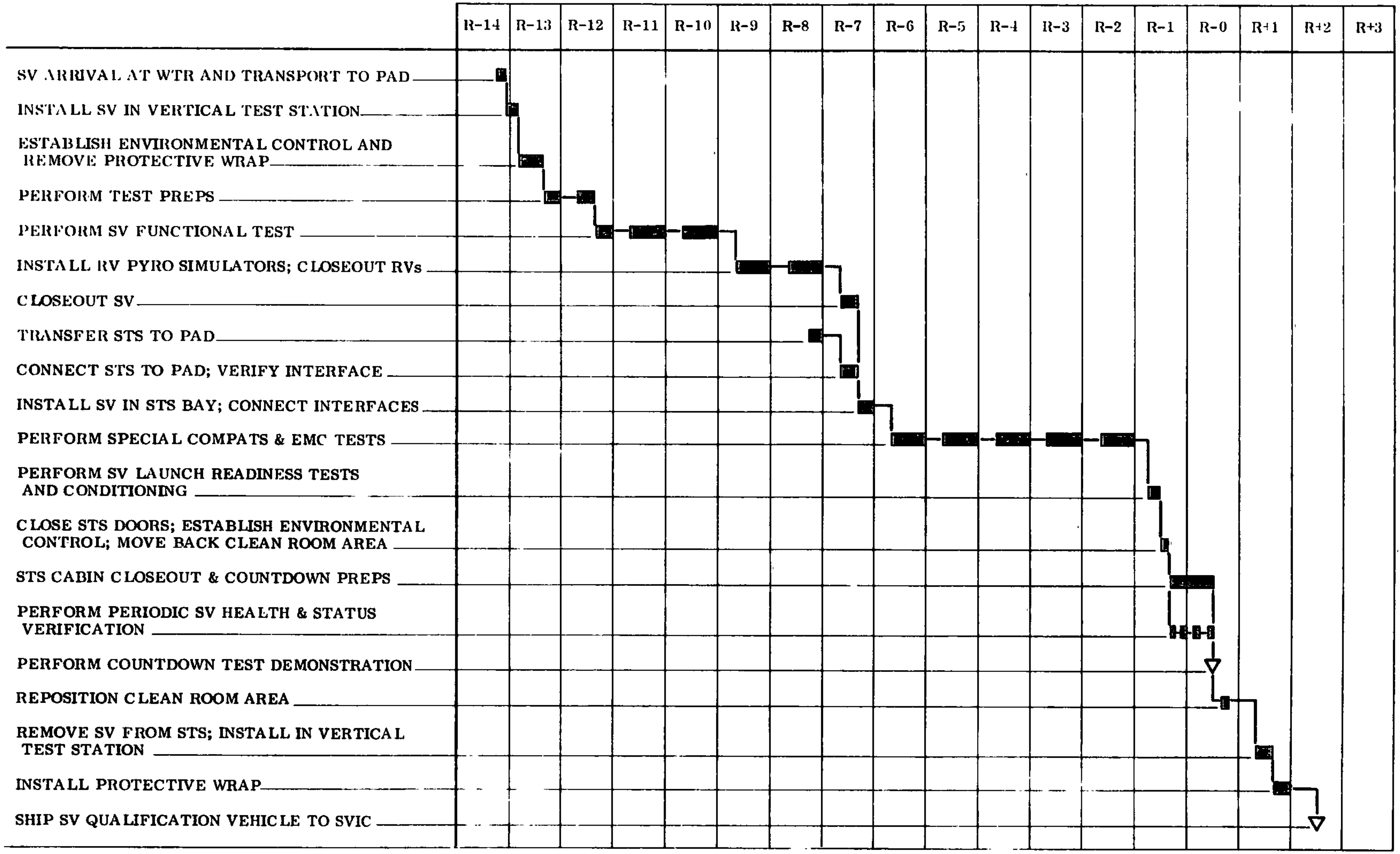


Fig. 8-3 Qualification Vehicle R-Day Schedule

BIF003W/2-069331-73

This schedule depicts the activities necessary to refurbish and reaccept the Hexagon SV. The camera system removed from a given SV will be recycled to the next vehicle. Time is available to perform a thermal/vacuum test on the refurbished vehicle if required.

~~TOP SECRET/H~~/10116

BIF003W/2-069331-73

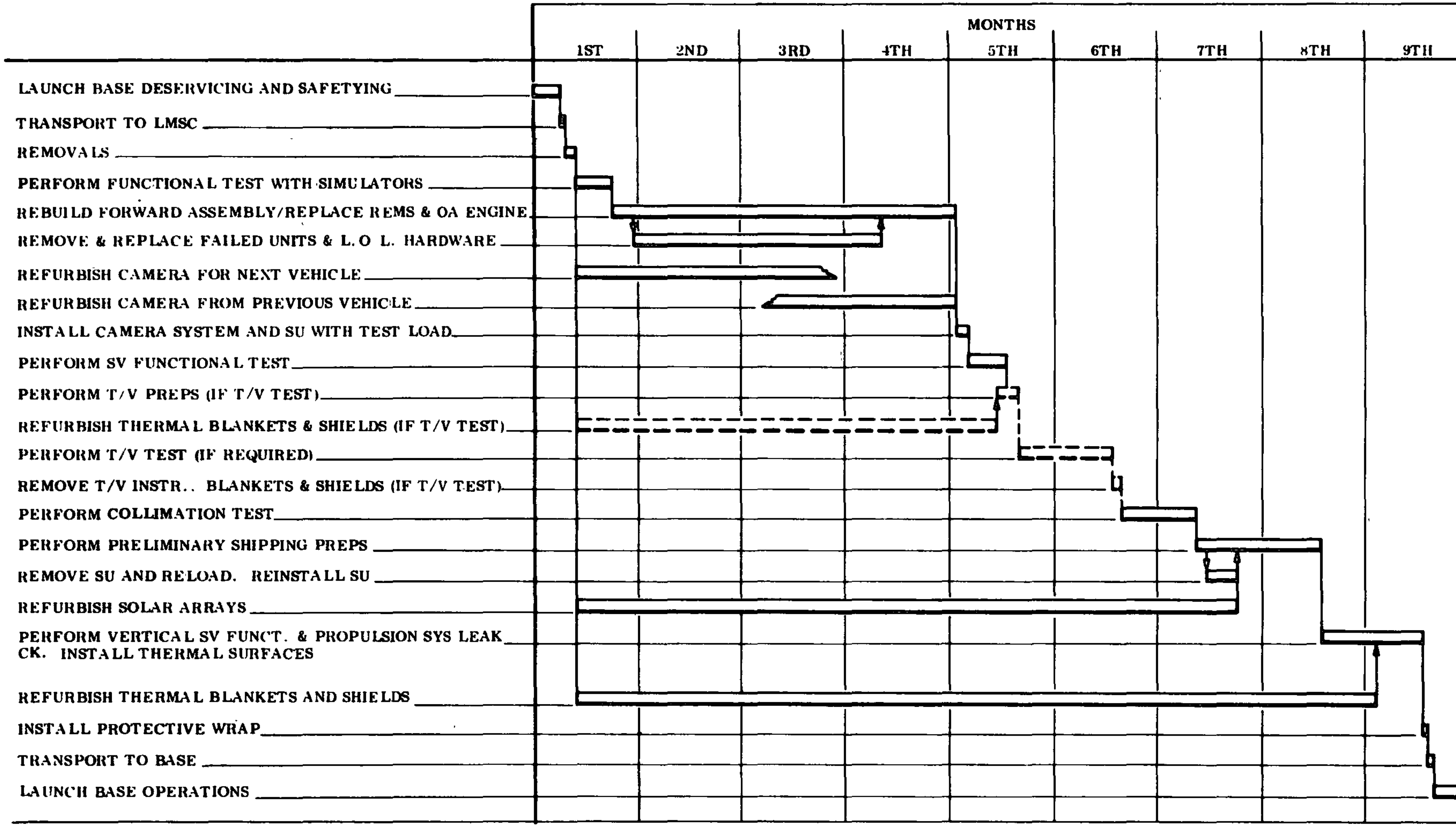


Fig. 8-4 SV Refurbishment Schedule

TOP SECRET/H/10116

TOP SECRET/H/10116

BIF003W/2-069331-73

This schedule displays the phase-in of the new Hexagon SV/STS program as the current SV/SLV program is phased out. The capacity of the existing facilities for processing the SV is adequate and no usage conflicts exist.

The SLV launch program phase-out assumed a three-per-year launch rate, which may not be the case by 1982. A more probable rate will be two per year, in which case the competition between SLV and STS programs for facilities is lessened.

~~TOP SECRET//H/~~10116

BIF003W/2-069331-73

Approved for Release: 2017/02/16 C05094784

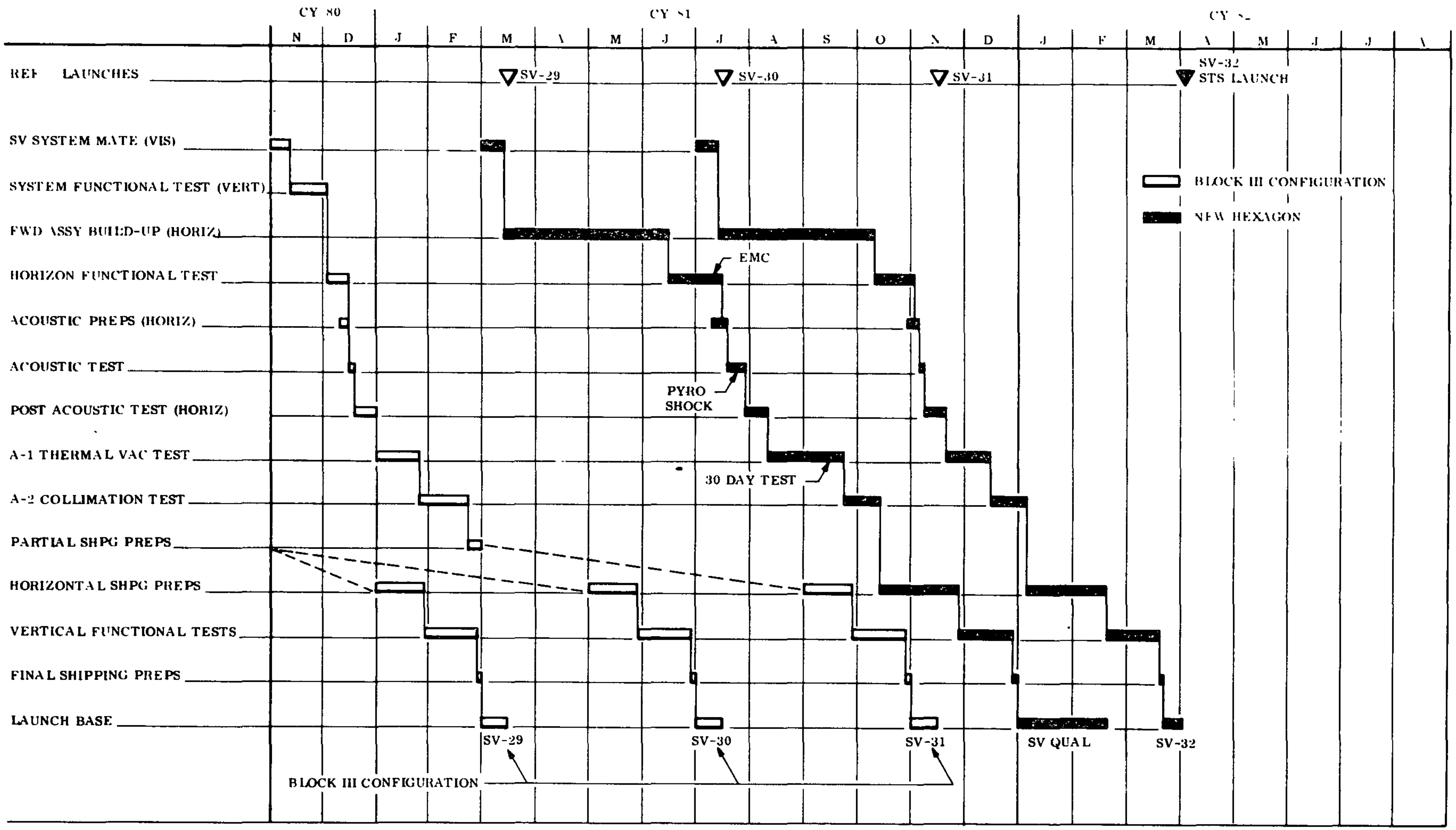


Fig. 8-5 System Test Transition Schedule

8-11

~~TOP SECRET/H/10116~~

~~TOP SECRET/H/10116~~

BIF003W/2-069331-73

Section 9
COST SUMMARY

This section presents the cost summary for only LMSC-provided equipment for the Hexagon SV/STS program. All cost information is based on fiscal year 1973 dollars and does not include a provision for economic escalation in future years. A phase-in from the existing Hexagon program, and a continuing program past 1992, has been assumed.

9.1 PRICING APPROACH

A bottom-up estimating technique was used to develop the component/black box non-recurring costs. A combination of manloading and task estimating techniques was used to develop the recurring costs. The LMSC Hexagon cost profile and history was used as the primary basis for these estimates. The history from Block I, escalated to 1973 dollars, was used generally as a base for areas of new development. Block III costs, adjusted to 1973 dollars, were employed as a base for similar hardware, and for recurring costs such as test, sustaining engineering, etc.

The only major development and thus high cost risk in the non-recurring costs is in the structures area. The costs presented for the structures design, development, test, and production were based upon the actual Block I costs for the existing Hexagon program. This approach is considered conservative because the Hexagon SV was the first satellite vehicle of its size developed at LMSC. Excluding the structures development, more than 80 percent of the remaining non-recurring costs are related to existing hardware or are minor modifications to existing designs, thus providing low cost risk.

All recurring costs except refurbishment are projections from the existing Hexagon levels. To compensate for the launch frequency and stability of design anticipated for a reusable SV, reductions approaching 50 percent have been forecast in all areas except STC support. The refurbishment costs constitute the highest cost risk area. However, total program cost estimates are considered to be well within the ±20 percent accuracy range since the refurbishment costs represent less than 13 percent of the total program costs.

9.2 GROUND RULES

It was assumed that no major design changes would be made after SV development and qualification. Minor changes such as design deficiencies and parts substitution have been included in the estimates.

Costs for STS launch, WTR facilities, camera system, RVs, [redacted] (b)(1) (b)(3) 10 USC 424 tracking network are not included in these estimates; they are assumed to be GFE.

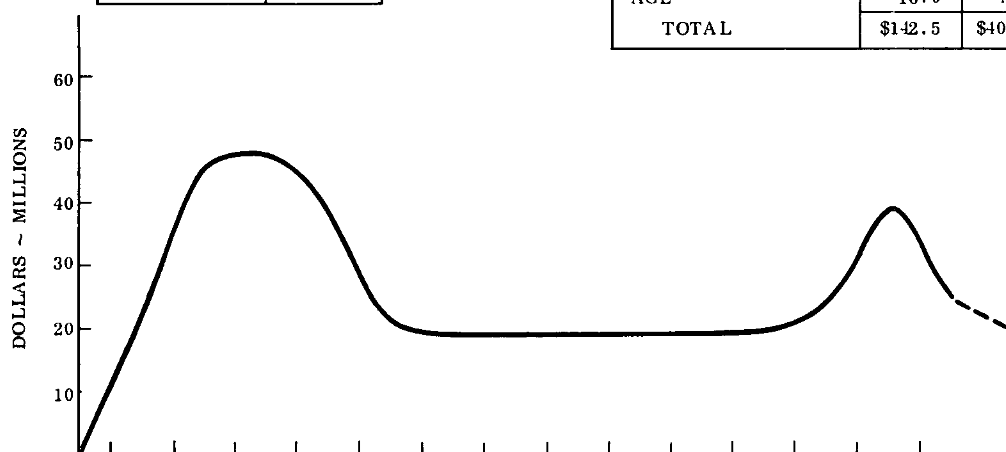
9.3 COST ESTIMATE

The LMSC costs for the Hexagon program are shown in Fig. 9-1, by fiscal year. Also included is a breakdown of non-recurring/ recurring costs by subsystem or function. The Block A non-recurring costs are total costs required to plan, develop, qualify, and deliver two flight SVs and one set of spares. Block A recurring costs begin at completion of systems-level qualification with preparations for the first launch, and continue throughout the operational phase (1982-1992).

Block B non-recurring costs are delta cost above the recurring costs required to replace the two flight units after the initial ten years of operation.

RECURRING	BLK A
SUSTAINING	$ 66.2
STC SUPPORT	10.2
LAUNCH OPS	19.3
TEST OPS	43.3
REFURB	45.7
TOTAL	$184.7

NON-RECURRING	BLK A	BLK B
STRUCTURES	$ 35.0	$ 7.5
PROPULSION	9.2	4.5
EDAP	10.5	4.5
ATTITUDE CONTROL	11.3	8.0
TT&C	30.5	12.0
VEHICLE	30.0	-
AGE	16.0	4.3
TOTAL	$142.5	$40.8



FISCAL YEAR	79	80	81	82	83	84	85	86	87	88	89	90	91	92	TOTAL
	23.0	46.0	48.0	37.0	20.0	18.0	18.0	18.0	18.0	18.0	18.0	24.0	38.0	24.0	$368.0 M

Fig. 9-1 LMSC SV/STS Cost Estimate

Section 10
NON-OVERFLIGHT OPTION

An alternate STS operational concept, the "non-overflight option" was investigated to determine impact upon the selected SV design and operations. No design or costing activities were accomplished. The alternate involves launch of an SV into, or retrieving one from, a 104 deg inclination orbit with a one-rev flight of the STS. The selected orbit inclination avoids an STS overflight of the Sino-Soviet area. NASA has designated the deployment mission as 3A and the retrieval as 3B.

Two approaches can be taken in analysis of the impact on this option. One is to assume that the SV is designed for the non-overflight option. The other makes the assumption that a Hexagon SV/STS system of the type described in this report is operational, and a sudden shift in international relations generates a need to use the non-overflight option. Both possibilities are examined.

10.1 DESIGN

Payload capability of the STS into a 104 deg inclination, 50 x 100 nm orbit is 30,000 lb, per "Space Shuttle Systems Baseline Reference Missions, Volume III — Mission 3A and Mission 3B" JSC Internal Note No. 73-FM-47; 26 March 1973. However, a 50 x 100 nm orbit is not acceptable for SV deployment. If a Hohmann transfer is employed from that orbit to an 80 x 100 nm SV deployment orbit, 1250 lb of OMS propellant would be expended, leaving a 28,750-lb effective payload weight. However, a transfer burn cannot be used for the 3A mission; instead, a direct ascent of the STS to 80 x 100 nm orbit is required. Effective payload weight will be something less than 28,750 lb. An unattractive option is deployment of the SV while the STS is in a 50 x 100 nm orbit; this requires an almost immediate burn of the SV orbit adjust engine. The SV will not survive one rev in a 50 x 100 nm orbit.

A key factor in selecting the 96.4 deg operational orbit for the Hexagon program was its sun-synchronous characteristic, which provides nearly constant beta angles. The 104 deg inclination yields a 1 deg per day beta angle shift, which requires that the thermal design of the SV accommodate large variations in beta angles. A 120-day mission would yield a corresponding 120 deg beta shift. The current Hexagon SV program had a ±60 deg beta angle range as a design goal, which is yet to be achieved. Adjusting the SV orbital plane from the 104 deg inclination to 96.4 deg is not a practical solution. The SV would require a delta velocity of approximately 3500 ft/sec to effect this change. Approximately 15,000 lb of SV orbit adjust propellant would be required for the maneuver.

Coupled with the thermal affect of beta angle shift is the shift of sun angle on the earth's surface, and its impact upon target area illumination and resulting photographic resolution.

Trading off these areas of impact, (1) STS effective payload weight limits and (2) beta angle shift effect on thermal design, and photographic GRD, LMSC established a 120-day maximum mission life. If the SV is a 4-RV design variant of the 6-RV configuration presented in this study, it would weigh approximately 26,850 lb, which is 6500 lb less than the 6-RV version. Most of the weight reduction is in elimination of expendables associated with 2 less RVs and 5 months less orbital life. However, if an existing 6-RV SV/STS configuration were off-loaded, its weight would be 27,475 lb. LMSC does not have STS performance data that identifies whether either case could be handled by a direct ascent to 80 x 100 nm orbit. If not, additional RVs could be removed, along with their expendables.

10.2 OPERATIONS

The normal mission profile for the program in this operational mode would require two 120-day missions per year. Three SVs are probably required. If only two SVs are available, the time span for refurbishment is reduced to eight months. Four STS launches per year would be needed in lieu of one every nine months for a program operating in a 96.4 deg orbit.

Other operational impacts are; (1) sufficient time is not allowed for a pre-deployment test of the SV, (2) for retrieval, the SV must be transferred into the rendezvous orbit prior to STS launch, and (3) once retrieved, time does not permit dump of residual SV propellants prior to SV reentry.

10.3 PROGRAM COST

The obvious program cost increase due to the non-overflight mode of operation is due to an added SV and camera system, and 2-2/3 equivalent additional STS launches per year. In addition, a greater probability exists for losing an SV due to deployment or retrieval problems, generating added SV replacement cost.

Another cost not necessarily measurable in dollars is the loss of 4 months coverage per year (2-month gaps between flights). The 6-RV SV concept operating at 96.4 deg inclination provides continuous coverage at less cost.

10.4 SUMMARY

Further study is required to match an SV configuration against the STS payload capability for direct ascent to an 80 x 100 nm orbit.

Section 11
SUGGESTED AREAS FOR ADDITIONAL STUDY

This study, which optimized the Hexagon SV for STS operations, and the previous 10116 study for minimum modification of the Hexagon SV for STS, have broadly delineated the range of Hexagon SV design changes required for STS compatibility.

It is suggested that additional studies be made of the methodology and required SV design evolution related to the problem of transitioning from an SLV-launched to an STS-launched program. Spares provisioning, backup concepts, and logistics for both the transition period and the STS era should be part of the study.

Firm and detailed SV/STS interface requirements should be generated. Although STS development paces the design of any SV configuration for the STS, the STS must also be designed to interface with the Hexagon SVs as well as other existing vehicles. Definition of detailed Hexagon SV interface requirements will facilitate early STS design guidance.

Lifetime requirements for equipment and materials become more demanding in the STS mode of operation; the SVs proposed in this study will spend a total of five years on orbit and will have a total lifetime of 10 years. Testing of materials and equipment should begin as soon as possible, to verify their capability to survive these lifetime requirements.

The SV reuse and lifetime requirements should also accommodate foreseeable changes in technology and mission requirements. Such changes could affect useful SV calendar lifetime, and at some point in time make it economically advantageous to build new SVs rather than update existing configurations. Methodology should also be established for determining the useful life span of a block of SVs.