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PERSONNEL LAUNCH SYSTEM (PLS) STUDY FINAL REPORT (DRD 12)

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Carl F. Ehrlich, Jr. et al

ROCKWELL INTERNATIONAL CORPORATION
Downey, California

Contract NAS1-18975
October 25, 1991

National Aeronautics and
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FOREWORD

This report has been prepared in response to DRD12 of contract NAS1-18975, Advanced Manned Launch System (AMLS) Study. The principal objective of this study is to conduct a detailed analysis to determine whether the lifting body concept can really achieve simpler operations with lower cost per flight at a low life cycle cost (LCC). This vehicle system was designed for crew safety, simple operations, and high operational utilization. An extensive series of trade studies and supporting analyses were performed on a reference system concept to refine the system design in preparation for defining the final, or "preferred", system concept. The results of these trades, scored with respect to their relative merits, provide significant insight in the design of the preferred system concept and are documented in the Volumes of this report.

This report was originally published as Rockwell Report SSD90D0090. Companion reports providing additional technical detail include:

Subsystems and Vehicle Design (DRD10 STS90D0357-1)
- Trade Studies and Supporting Analyses

Manufacturing and Verification (DRD10 STS90D0357-2)
- Trade Studies and Supporting Analyses

Operations and Support (DRD10 STS90D0357-3)
- Trade Studies and Supporting Analyses

Hardware/Software Design Description (DRD 3 SSD90D0091)

Acquisition Phase Definition (DRD 4 SSD90D0092)

Operations and Support Analysis (DRD 5 SSD90D0093)

Reliability/Maintainability Analysis (DRD 6 SSD90D0094)

Life Cycle Cost Analysis (DRD 7 SSD90D0095)

Technology Development Plan (DRD 8 SSD90D0096)

ACKNOWLEDGMENTS

The performance of this study would not have been possible without the contributions of a highly experienced and dedicated contract team and additional support from many other organizations and functional areas, both government and industry. These contributions have focussed a wealth of knowledge and talents to the definition of a next generation cost-effective manned spacecraft system.

We and our counterparts at NASA/Langley have established a far closer relationship than the normal contractor/customer relationship. Our people have been working together on almost a daily basis with each group providing direct support to the study. Special acknowledgments are given to:

Howard Stone	Technical Monitor
Doug Morris	Operations
Arlene Moore	Life Cycle Costing
Ed Dean	Life Cycle Costing

The study team that contributed to the depth and breadth of this study is composed of a number of Rockwell organizations supported by highly respected subcontractors. The Rockwell Space Systems Division (SSD) in Downey was the lead organization for the study and provided the major engineering, manufacturing, and operations support for the study. Rockwell's KSC function provided launch site operations support while our Space Operations Center provided mission planning and flight support. Pan Am World Services provided critical support in two major areas: airline operations concepts and KSC ground operations. ECON, Inc., performed the cost analysis of the system. The key personnel contributing to the study included:

Rockwell:

Carl Ehrlich	Program Manager
Jack Potts	Deputy PM and Systems Engineering
Steve Kent	Subsystem Design
Ken Schell	Manufacturing and Verification
Ron Caldwell	Operations and Support
Richard Earhart	Technology Assessment
Denver Gore	Life Cycle Costing
Chuck Urrutia	Launch Site Operations at KSC
Ray Randolph	Mission Operations at JSC

Pan Am World Services:

Jim Morris	Airline/Aircraft Operations
Will Linger	Ground Operations at KSC

ECON:

John Skratt	Costing Analysis

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1.0 INTRODUCTION AND SUMMARY

To assure national leadership in space operations and exploration in the future, NASA must be able to provide cost effective and operationally efficient space transportation. Several NASA studies and the joint NASA/DoD Space Transportation Architecture Studies (STAS) have shown the need for a multi-vehicle space transportation system with designs driven by enhanced operations and low costs. The NASA is currently studying a personnel launch system (PLS) approach to help satisfy the crew rotation requirements for the Space Station Freedom. Several concepts from low L/D capsules to lifting body vehicles are being examined in a series of studies as a potential augmentation to the Space Shuttle launch system. Rockwell International Corporation, under contract to the NASA Langley Research Center, has analyzed a lifting body concept to determine whether the lifting body class of vehicles is appropriate for the PLS function. This report discusses the results of this pre-phase A study.

1.1 BACKGROUND

The principal objective of this study is to conduct a detailed analysis to determine whether the lifting body concept can really achieve simpler operations with lower cost per flight at a low life cycle cost (LCC). This vehicle system was designed for crew safety, simple operations, and high operational utilization. A concurrent engineering process was used to ensure coordination of all functional disciplines to establish a total system approach. Producibility, operability, and maintainability requirements were continuously integrated into the system design process. Costing tasks focussed on extracting cost effective design concepts and costing trends and drivers. The study employed the combined experience of several study participants including Pan Am World Services, Inc., ECON, Inc., and other Rockwell divisions including RSOC.

This study focussed on the definition of a single vehicle configuration, the HL-20, without modifications and a single mission, Space Station crew rotation (Figure 1-1). The present Rockwell task was to conceptually define the structure, subsystem complement, and operational concepts for the PLS system. Later studies will explore the changes required to accommodate alternate missions. For nominal missions, the vehicle is designed to land horizontally on a runway at the launch site (KSC). It may also be landed at any suitably-equipped airport by providing portable GSE and any required landing aids.

An extensive series of trade studies were performed on a "reference" system concept, developed at the outset of the study, to refine the system design in preparation for defining the final, or "preferred", system concept. The results of these trades, scored with respect to their relative merits, provide significant insight to the definition of the preferred system concept. Figures of merit for this study include life cycle cost, operational

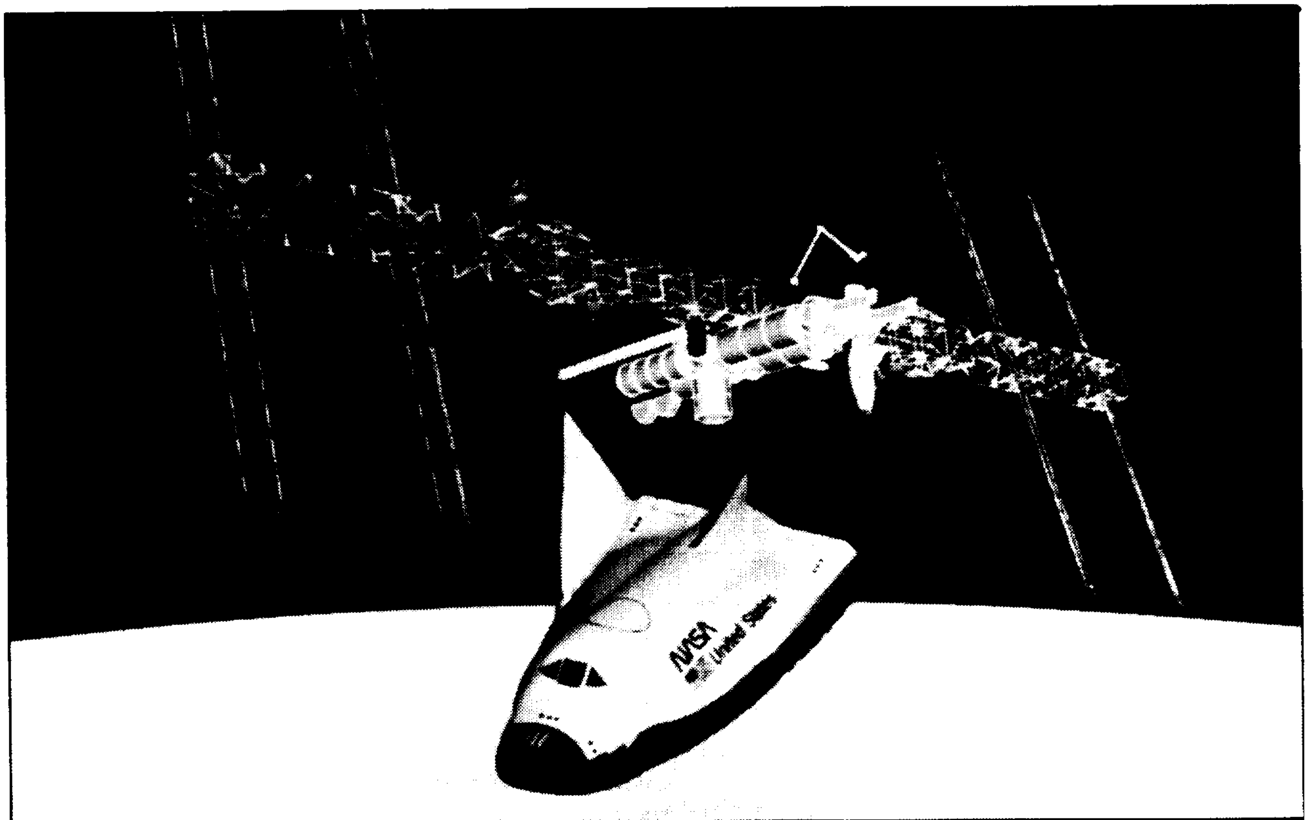


Figure 1-1. The Lifting Body Spacecraft Provides Crew Rotation Services to Space Station Freedom.

utilization, maintainability, manufacturability, crew safety, and crew habitability.

The outputs of the study presented in this report include quantified data that support the utility of the lifting body concept as a cost-effective system for the Personnel Launch System mission. Comprehensive data highlight the potentially significant benefits which may be realized if the projected "airline" approach to reliability, maintainability, and operations is rigorously followed.

1.2 CONCEPT DEVELOPMENT PHILOSOPHY

The primary role of the Personnel Launch System is to provide assured manned access to space. In meeting this objective, we have incorporated features into the design which provide for crew safety, cost-effective operations, and high operational utilization. The key to the overall system design is our adoption of airline/aircraft approaches to certification and flightworthiness: we do not decertify the spacecraft after each flight as is effectively done in the Shuttle program, but rather maintain the system in a flightworthy status. The entire system, spacecraft and support systems, is designed for maintainability and producibility.

The basic program has the principal objectives of achieving high levels of operational efficiency at affordable life cycle costs while maintaining high operational utilization and crew safety. These goals (Table 1-1) have driven the design of the Rockwell PLS concept. The system design reflects the operational goals through design features that have been incorporated into the flight vehicle design concept. It also provides features which facilitate manufacturing, maintenance, and inspection and overhaul.

Table 1-1. Program Requirements Drive The System Design.

PROGRAM REQUIREMENT	PLS FEATURES
CREW SAFETY	PAD ESCAPE SYSTEM CREW MODULE INTEGRITY (WATER LANDINGS) MULTIPLE INGRESS/EGRESS HATCHES ANY RUNWAY
SIMPLE OPERATIONS	STANDARD MISSIONS & PROCEDURES CREW FLIGHT PROFICIENCY MAINTENANCE COMMON DATA BASES HIGH LEVEL OF AUTONOMY
HIGH OPERATIONAL UTILIZATION	MINIMUM TURNAROUND TIME USE OF AIRLINE MAINTENANCE PROCEDURES MAINTENANCE SCHEDULING
LOW COST PER FLIGHT & LOW LIFE CYCLE COSTS	SUBSYSTEMS DESIGNED FOR MINIMUM MAINTENANCE INSPECTABILITY & ACCESSIBILITY TO SUBSYSTEMS HIGH-RELIABILITY SUBSYSTEMS COST-OPTIMIZED BUILD RATE
OPERATIONS & SUPPORT EFFICIENCY	DESIGNED FOR ACCESSIBILITY & MAINTAINABILITY TRANSPORTABILITY BUILT-IN TEST AUTONOMOUS OPERATIONS
ECONOMICALLY PRODUCIBLE	MANUFACTURING ACCESS EXTERNAL SYSTEMS INSTALLATION LESS COMPLEX WELDMENTS HEAT SHIELD INSTALLATION/REMOVAL

LARGE DESIGN MARGINS & SYSTEMS ROBUSTNESS ASSURE OPERATIONAL EFFICIENCY AT MINIMUM DESIGN RISK

The system concept developed for the PLS reflects an integrated approach to the design of the system. No single area (subsystems, design layout, manufacturing, nor operations) dominated the design effort but rather all program requirements were addressed concurrently in conducting the design activity.

Note that the PLS glider is considerably smaller than the Shuttle orbiter (Figure 1-2). Its landed weight is less, and there are substantially fewer maintenance-significant LRU's and thus proportionally less maintenance time. The design process employed inherently drives the system to lower operating costs, and therefore the design reflects features required by operations and maintenance to minimize costs.

In this study, the aircraft/airline approach to aircraft certification and flightworthiness was used as a reference. In this approach, the vehicle and vehicle subsystems are certified one time and regular maintenance is scheduled to maintain that

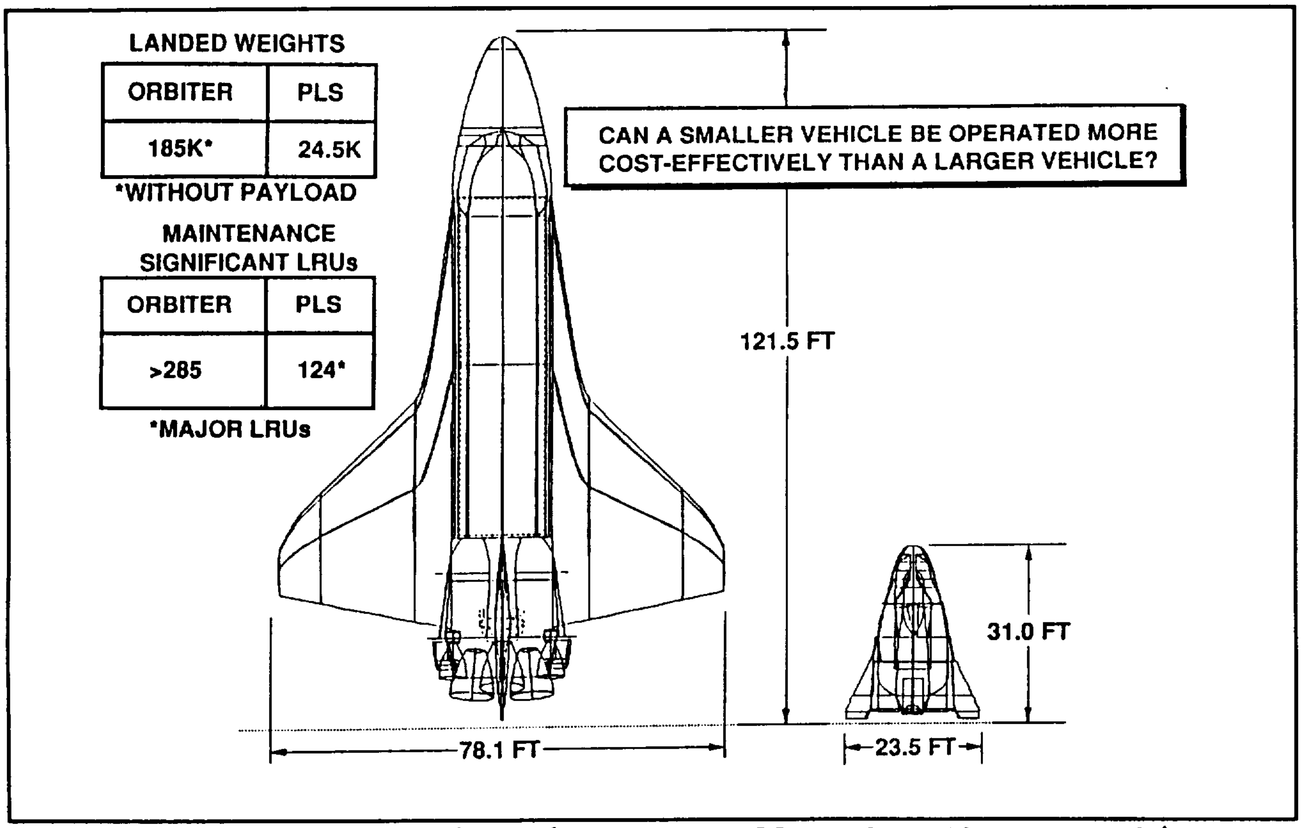


Figure 1-2. The PLS Glider is Much Smaller Than the STS Orbiter.

airworthiness. The subsystems, missions, and operations approaches are designed to reflect this basic philosophy - recognizing that this is a major cultural change from the way NASA does business today.

1.3 CONCEPT DEVELOPMENT

A key feature of the design which results from this philosophy is the provision for easy accessibility for performing turnaround maintenance. Wherever possible, the subsystems are installed outside of a pressurized crew module for direct accessibility during ground support operations. Leeward surface access panels are designed to be removed during the maintenance process.

1.3.1 Design Features

The most obvious feature resulting from our design philosophy are the means we have provided to enable easy accessibility for performing turnaround maintenance. Wherever possible the subsystems are installed outside of the pressurized crew module. This feature provides two major benefits: 1) it greatly enhances accessibility for maintenance during turnaround while minimizing opportunities for incurring collateral damage, and 2) it provides for greater efficiency in manufacturing since more installers can work at the same time (avoiding the Apollo scheduling "crunch" in making internal installations).

Removable panels are provided for this purpose (Figure 1-3); they are removed for access and reinstalled when maintenance is complete - they benefit from intensive seal research performed for the NASP program. Access to the aft compartment is provided by folding the fins along imaginary axes along the upper body inboard of the fins; one-g hinges are provided which are powered externally (internal motors drive the folding mechanism at zero-g). Access to the crew compartment during maintenance operations is primarily through a tunnel extending to the upper hatch (the pad ingress/egress port) from a clean room in the maintenance facility.

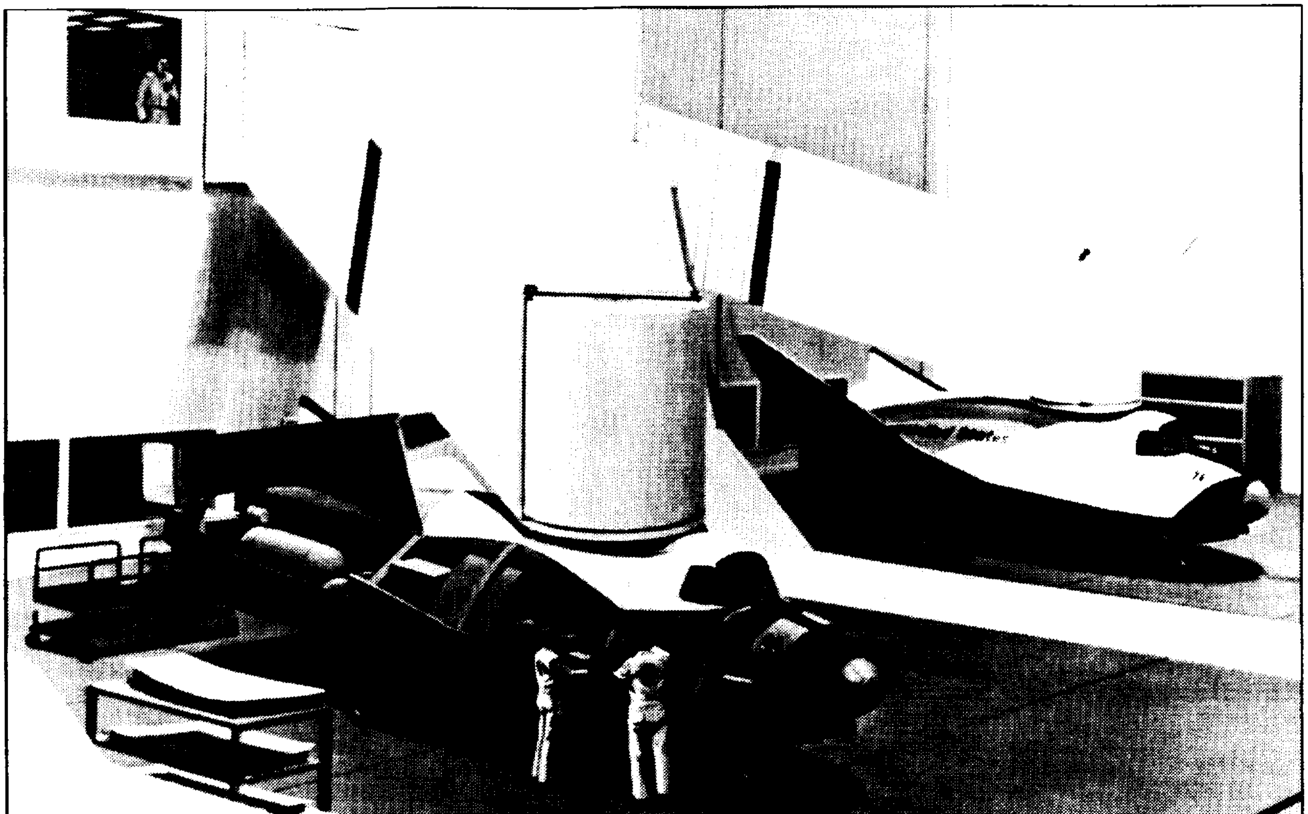


Figure 1-3. Accessibility During Manufacturing and Maintenance Is a Driving Design Feature.

The flight vehicle is made up of a number of major component parts, each of which may be manufactured offsite and mated during the final assembly process (Figure 1-4). The crew module is standard 2219 aluminum made up from sheets rolled in one direction (no compound surfaces with the exception of the windshield area) and welded. Internal manufacturing access is provided at the aft tunnel interface - this feature was adopted from the Shuttle orbiter crew module manufacturing process to ease assembly and installation of internal systems. The crew module is the primary structure of the vehicle; outboard frames carry airloads from the external surfaces to the module and provide support for subsystem installation and the landing gear.

The lower heat shield concept has a legacy extending back to 1983 and the CASTS program sponsored by Langley during 1982-3. In

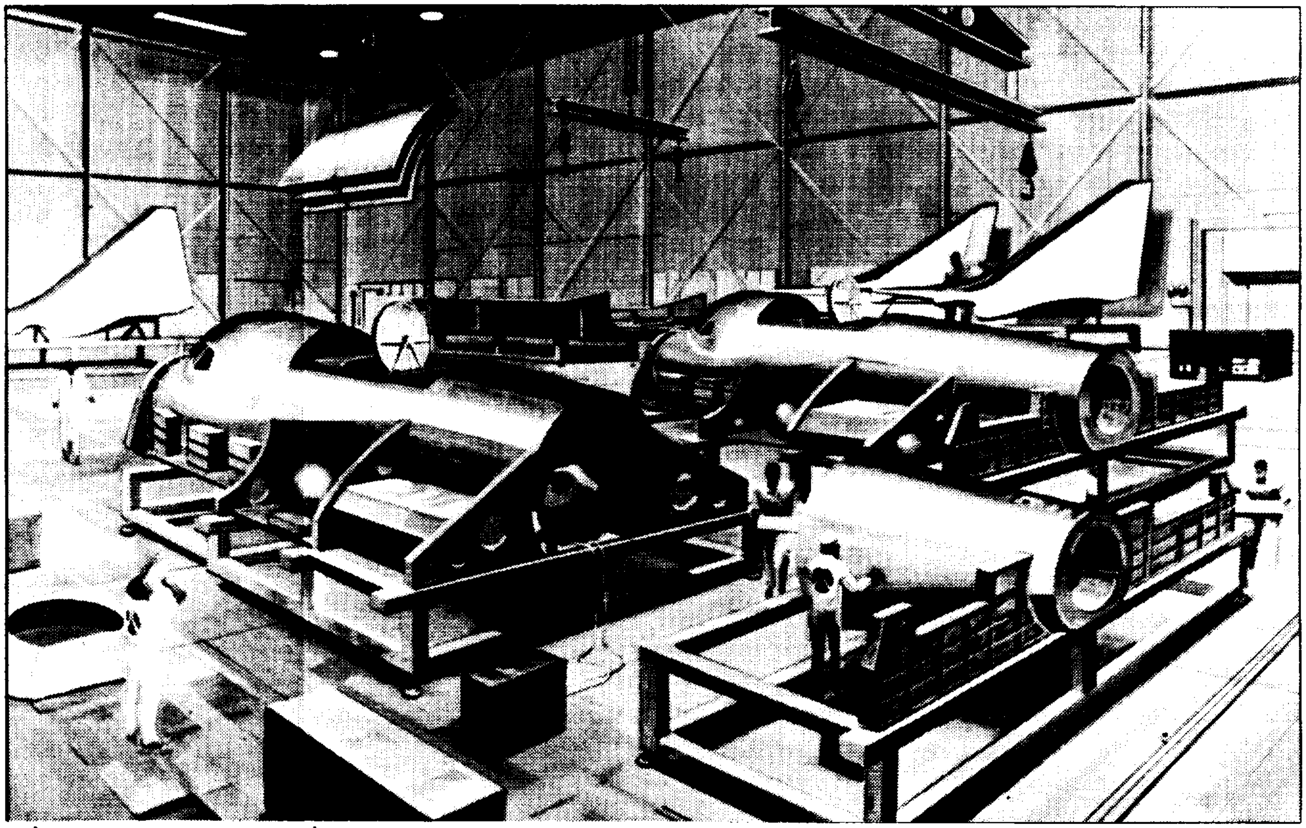


Figure 1-4. Design Features Reflect the Influence of Maintenance and Manufacturing.

that program, Rockwell designed and fabricated a graphite-polyimide composite shuttle Orbitor body flap with tiles directly bonded to the material (no SIP). The present heatshield is to be manufactured in a similar manner (an alternate could be an all-ACC heatshield similar to the Shuttle orbiter nose cap and wing leading edges). It is attached to the carry-through frames through a series of thermal standoffs.

Our development of the lifting body vehicle concept into the PLS concept has benefitted from the combined experience of our Rockwell and subcontractor participants. Given the broad objectives on operational efficiency, low life cycle costs, and crew safety, we have provided features enhancing accessibility for maintenance, enabling easy access for installation of subsystems during manufacturing, simple welds on conventional material, crew ingress and egress hatches that accommodate deconditioned SSF personnel, transportability in aircraft and the Shuttle orbiter, and subsystems enhancing the ability for efficient maintenance operations and turnaround (Figure 1-5).

1.3.2 Launch System

The existing Titan IV launch system was the initial choice for the PLS booster system (Figure 1-6). It provides the launch capability for the PLS from KSC to SSF. The launch escape system (LES) is incorporated into the launch vehicle adaptor, providing a

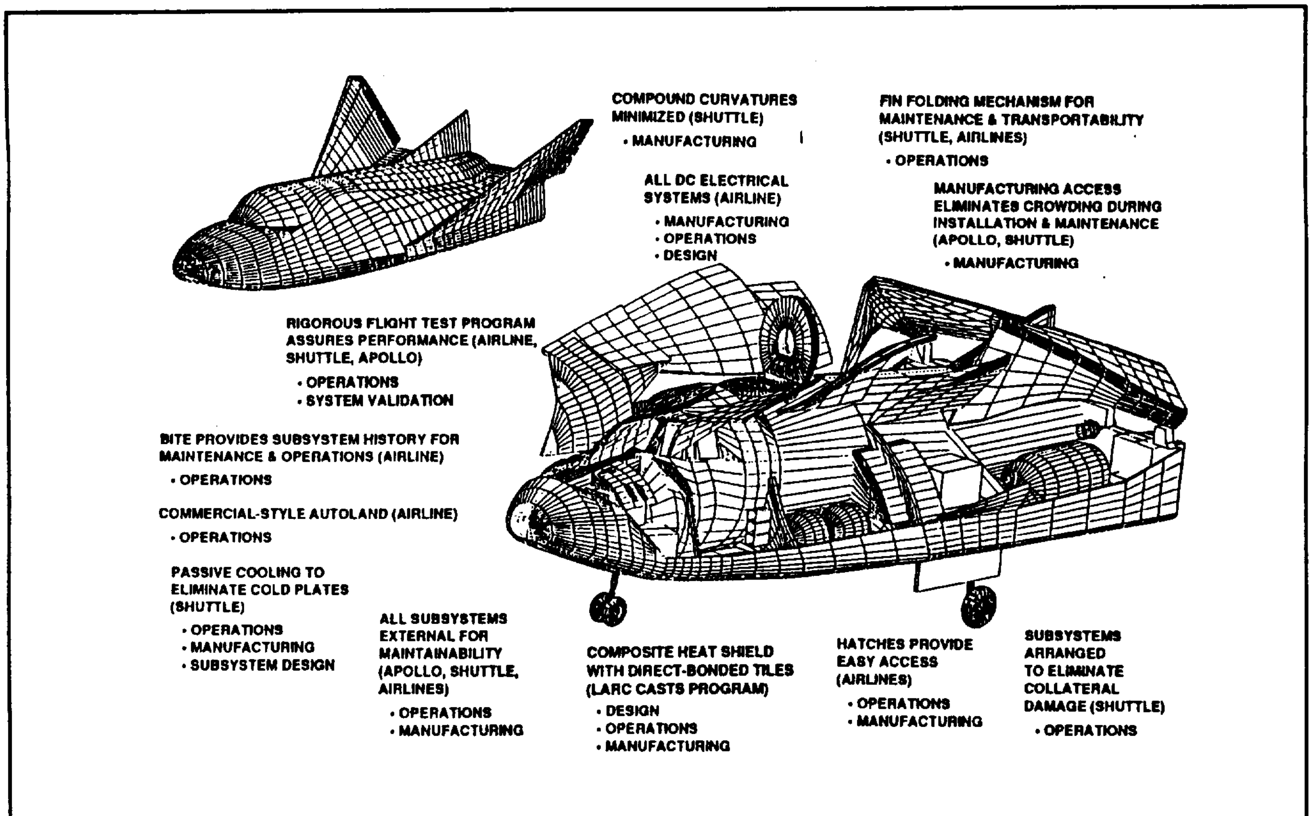


Figure 1-5. The PLS Concept Represents a Composite of Experience.

pad escape capability for the crew. Separation is nominally provided at the normal separation interface, the base of . In a potentially catastrophic launch vehicle event, or during ascent flight, the launch escape system separates at the abort interface and powers to a condition for safe recovery of the crew.

We are currently using an ALS stage-and-one-half concept as the primary launch system. The general arrangement is retained, especially the LES itself. The LES configuration is common to all candidate boosters, including an LRB-derivative concept. Only the LES-to-launch vehicle adaptor design will be changed for each booster.

1.3.3 Surface Transportation

One basic groundrule that drove the vehicle sizing is the requirement that the vehicle fit into the Shuttle orbiter payload bay and carry two pilots and 8 to 10 station crew members. By providing a wing folding mechanism, the PLS easily fits into the payload bay while providing sufficient internal volume for 8 passengers in addition to the flight crew (Figure 1-7). The folding fin feature also facilitates transport by C-5A and C-17 aircraft.

The vehicle is supported at the aft end by a bridge fitting mated to the aft access hatch, which in itself is primary structure. The forward end of the PLS is supported under its nose

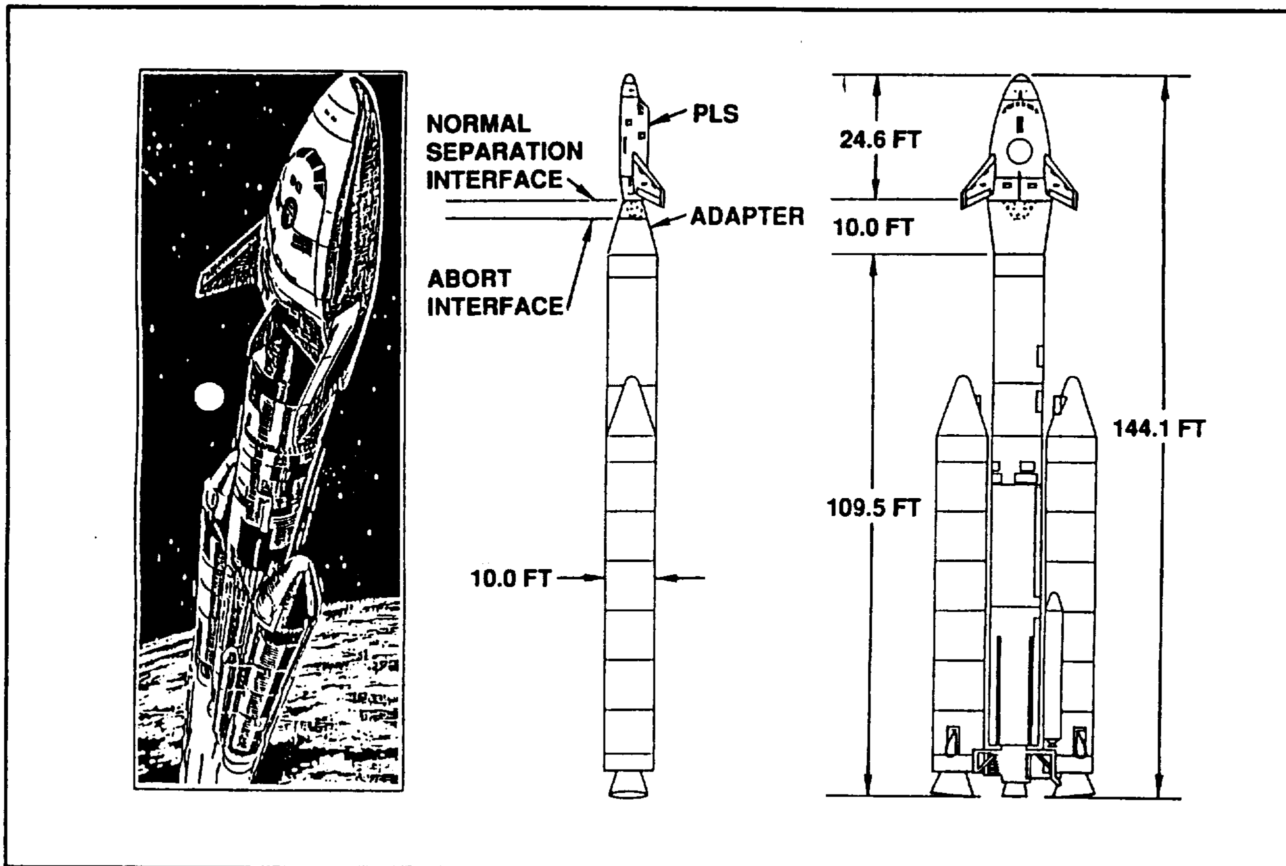


Figure 1-6. Titan\PLS Launch Configuration.

by a second bridge fitting. The folding mechanism is powered on-orbit by integral motors and on the ground by an external drive. The fins are designed to operate in a 1-g environment without additional support.

1.3.4 Crew Safety

A recovery system is provided for aborts that require a water landing. The Launch Escape System (LES) is designed to drive to a condition for safe crew recovery. In the case of a pad abort, the LES fires for about four seconds, providing an 8-g acceleration from the pad (Figure 1-8). The expended LES separates immediately to minimize destabilizing aerodynamic forces. At an appropriate altitude, the parachute system opens and lowers into the offshore waters. Inflatable balloons open to ensure that the aft hatch is maintained above water level in order to provide at least one dry hatch for crew removal in the case of an overturned vehicle.

The principal objective of the system is to ensure safe crew recovery thorough designing for an intact crew module - the primary structure. The crew module is designed to remain intact and water-tight during a water recovery to ensure safe recovery of the crew. Structural damage to the heat shield, wings, secondary structure, and to the subsystems is expected and no consideration is made for reuse of the flight vehicle. In fact, the crumbling

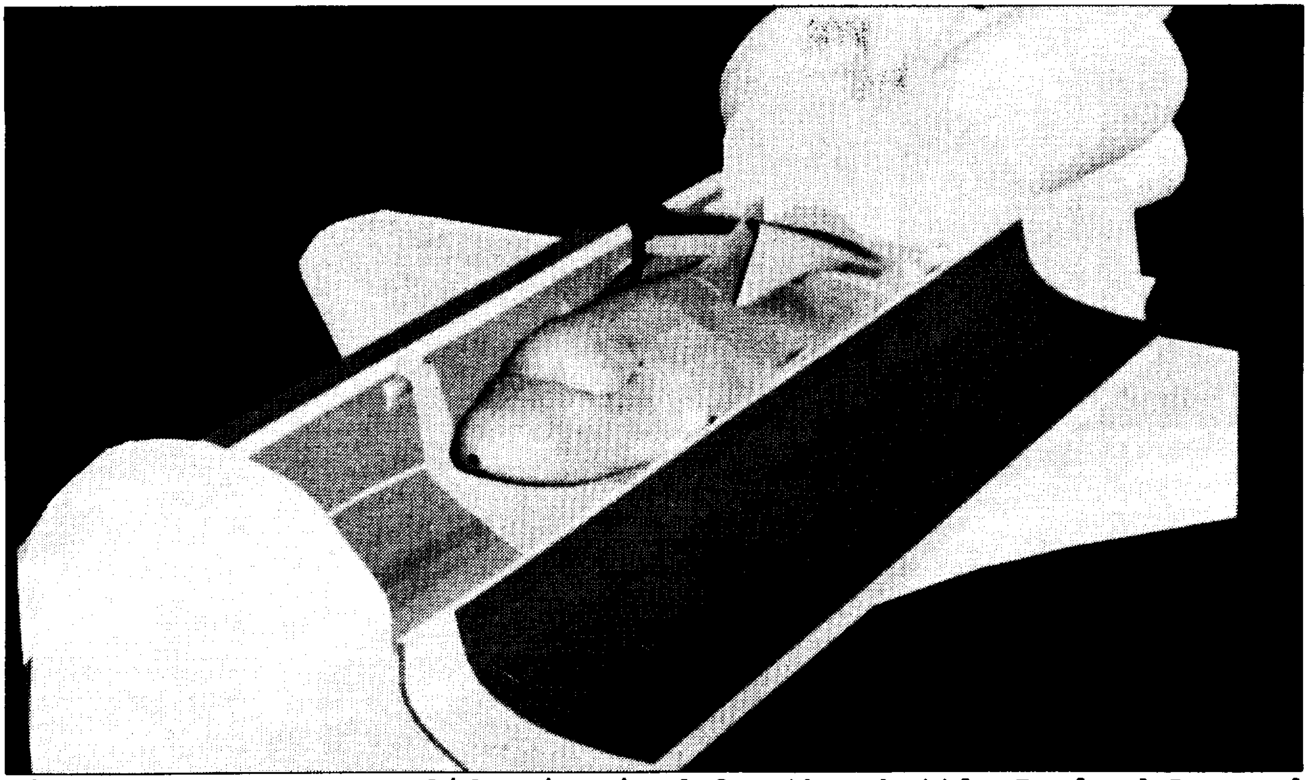


Figure 1-7. The PLS Glider is Sized for the Shuttle Payload Bay and Fits Into the C-5A and C-17 Aircraft.

of the structure will provide some impact load relief to the crew.

1.3.5 Space Station Freedom Operations

Two methods exist for final mating to the space station node: 1) berthing - use of the SSF remote manipulator arm for the final movement to the node while the spacecraft is inactive, and 2) docking - wherein the spacecraft is controlled by the crew and provides all maneuvering functions. The docking concept has been selected for the PLS because finer attitude and movement control using the vernier RCS thrusters is possible and relieves the SSF crew from additional tasks (Figure 1-9).

The vehicle is maneuvered and controlled by a pilot located in the aft tunnel using view-ports in the access hatch for targeting and visual control. The conical interface remains on-orbit with the station. No resources are required from the station but such items as power can be utilized, if available, to recharge or extend the life of the spacecraft batteries. Separation is accomplished in reverse sequence.

1.3.6 Post-Landing Egress

Crew members returning from the Space Station are expected to have been in space at zero-g's for up to six months, some more and some less. In these cases, the returning crew will be deconditioned to the point that they must be removed from in a near horizon-

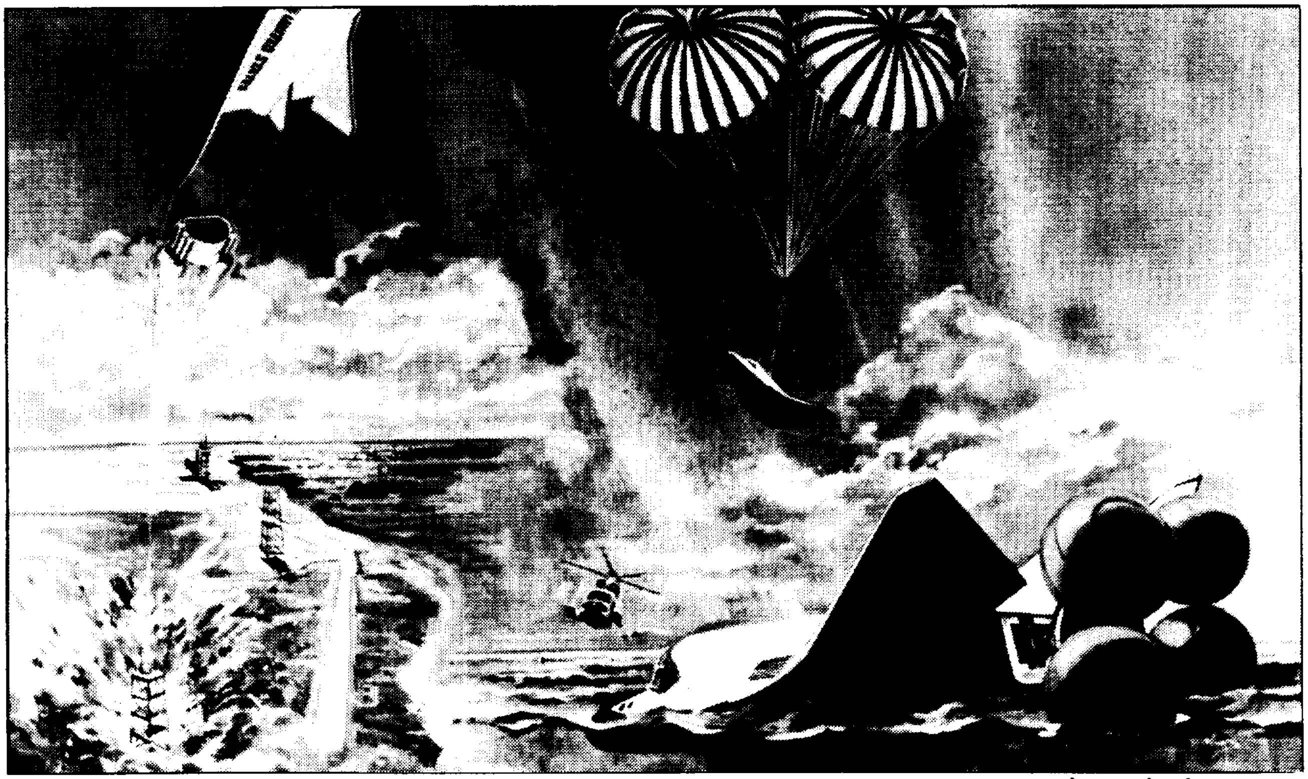


Figure 1-8. The Launch Escape System Ensures Crew Survivability.

tal position, keeping heart and head in a level plane to avoid adverse reaction to the 1-g Earth environment. This precludes use of the top hatch (pad ingress-egress hatch) for crew egress.

The design of accommodates this requirement easily (Figure 1-10). The aft hatch, used for docking with the Station, is the primary egress hatch during ground operations. Returning crew members are removed on a pallet from their seated position (the seats are sharply reclined for entry and landing) and assisted onto gurneys for transport to a crew recovery facility until they regain their ability to stand upright in the Earth's gravity.

1.3.7 Acquisition Plan

The Master Schedule (Table 1-2) outlines the sequence of major events in the development of the PLS concept through to IOC. Major milestones beginning with ATP for Phase A are shown. The manufacturing schedules include lead times for procurement of material and vendor parts as well as inhouse manufacturing and assembly, test, and validation.

Structural and component tests will verify the design of those elements. The need for a long term dynamic test program (test to failure) as used for aircraft is being evaluated; the low flight rates for these vehicles may preclude the need for such tests.

The approach and landing (ALT) flight test program includes full scale low speed tests of launched from a B-52. These tests

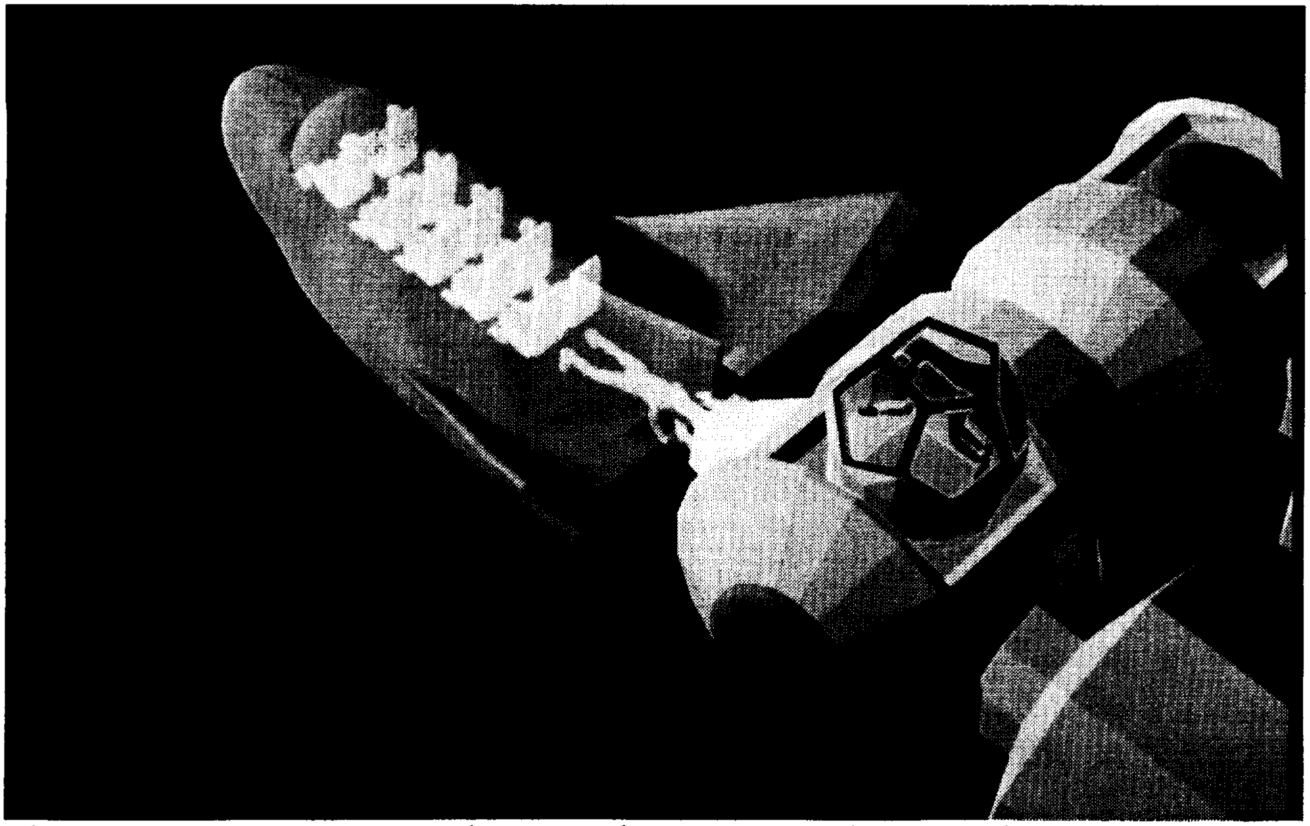


Figure 1-9. The PLS Design Provides for Active Docking at The Space Station.

include unpowered subsonic and landing tests and powered acceleration tests to high supersonic speeds. These tests are designed to verify the post-entry handling qualities, landing performance, and the guidance and control and autoland systems. The orbital flight tests complete the flight test series by verifying the overall operational capability of the full system over a wide range of operating conditions.

1.4 FEATURES ENABLING MINIMUM COST OF OPERATION

We have adopted a wide variety of measures to minimize the cost of operations (Table 1-3). Some of these we adopted from the airline/aircraft operating procedures while many others have been the results of our experiences in designing and developing the Shuttle orbiter and operating the Shuttle system. Our Apollo experience has also been a significant benefit.

While the HL-20 is inherently smaller than the orbiter, we have also minimized the parts count and simplified those for easier maintenance. We have eliminated the hydraulic system and provided and all-electric system to take advantage of the accelerating technology of electromechanical actuators for the aerodynamic surfaces; maintenance requirements are significantly reduced in this manner. Serial launch operations have been eliminated by eliminating toxics propellants: NTO and MMH. We employ hydrogen

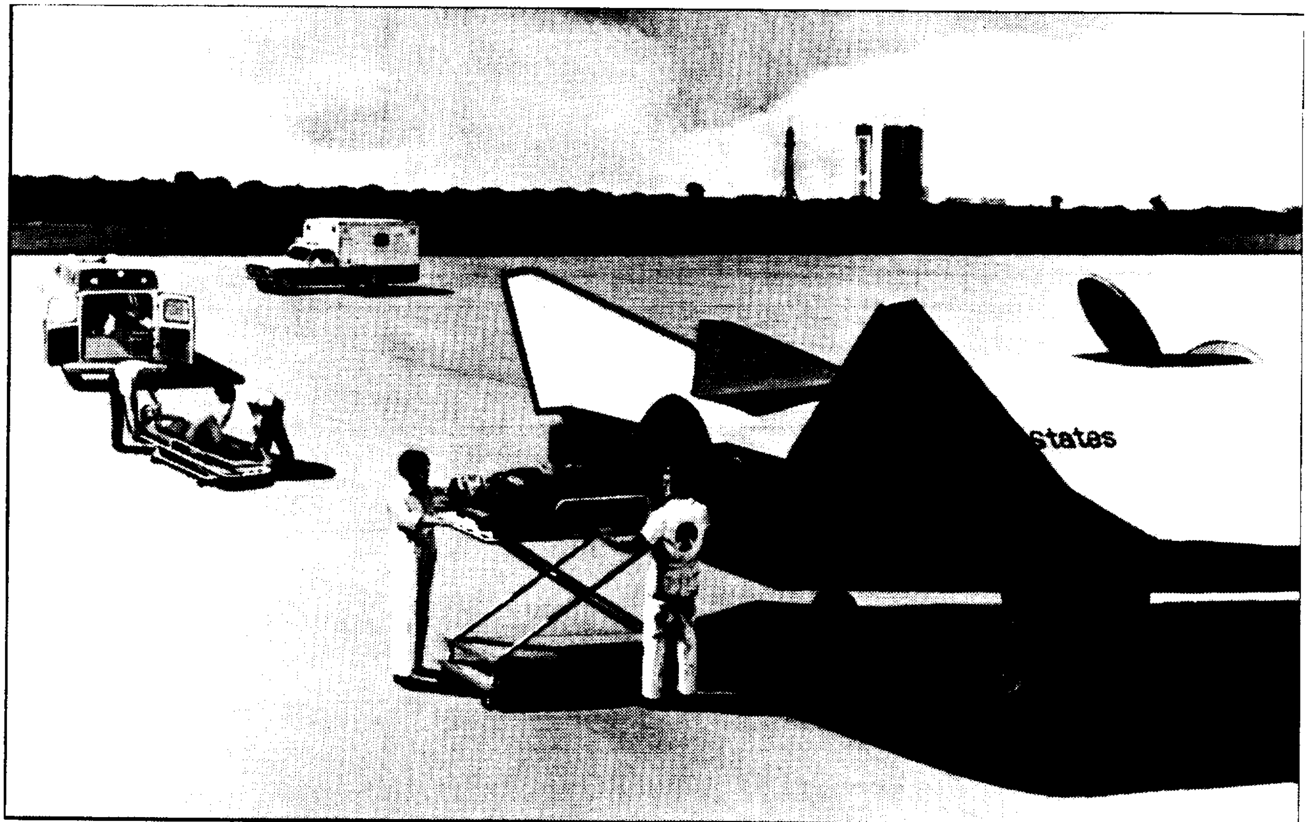


Figure 1-10. Inherent Features of the Lifting Body Permit Extraction of a Deconditioned SSF Crew.

peroxide and JP4 as propellants for the RCS and OMS systems.

Built-in test systems provide a continuous history of system health and status. This information will greatly enhance maintenance operations by enabling the scheduling of maintenance actions and eliminating unscheduled maintenance actions - always expensive. The standardization of missions into a reduced number of preplanned missions combined with a robust launch system design will reduce the requirements for extensive flight planning.

We have found that the PLS system concept outlined here will be more operationally efficient than current systems because we have incorporated the experiences gained from years of developing and operating manned space systems and the experiences of the airline/aircraft large fleet design operations. The latter experiences have been particularly beneficial because they have been the results of millions of flight hours of operation - all being incentivized by a profit making goal. Efficiency in all activities and design approaches to achieve those efficiencies must be major design goals for all new systems in the future.

A major issue outstanding at this time is the lack of a man-rated launch system that is equally cost-effective to operate. The booster system must be cost-effective for not only the PLS concept but also for all applications.

Table 1-2. The Master Program Schedule Highlights the Major Events During System Development.

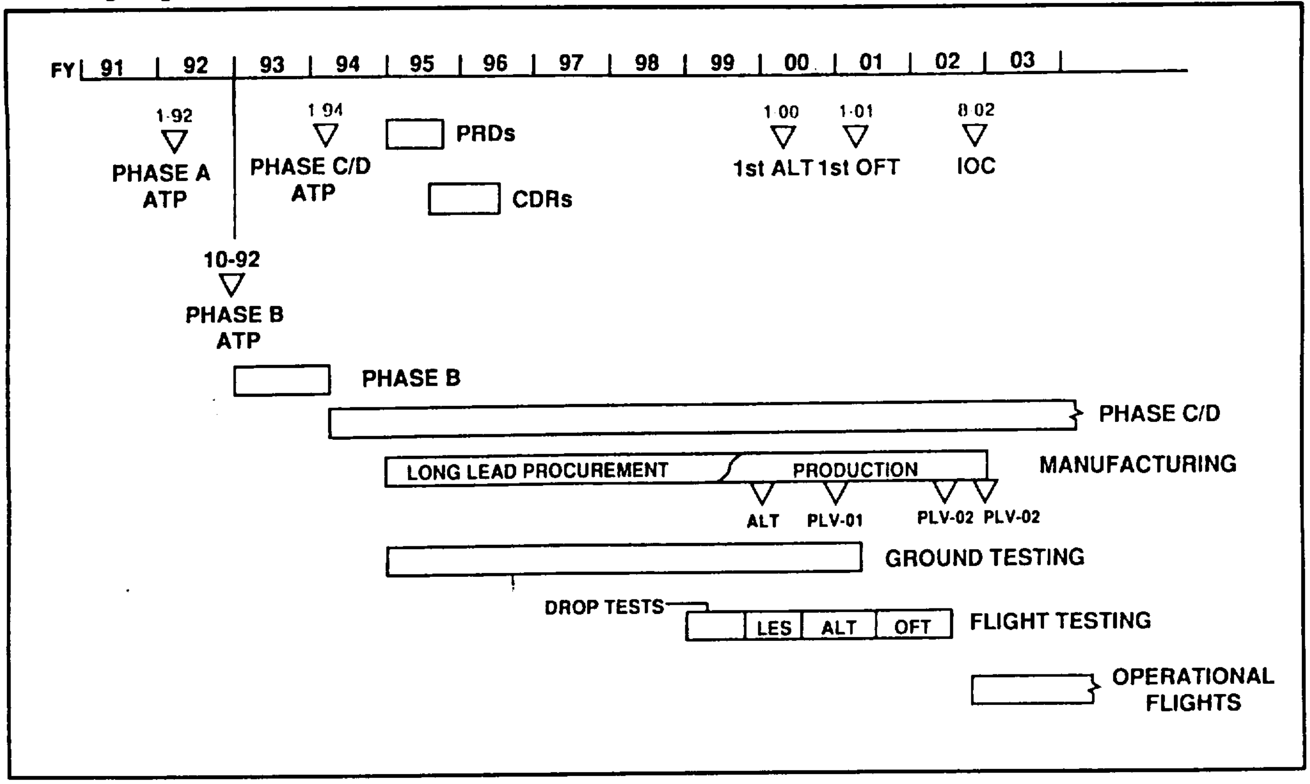
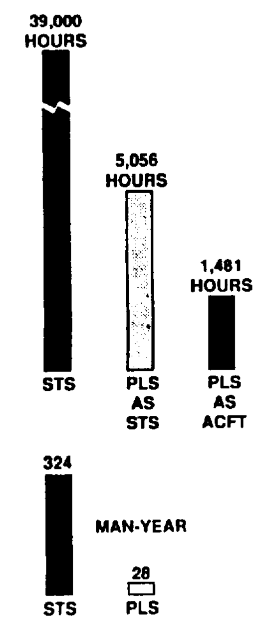
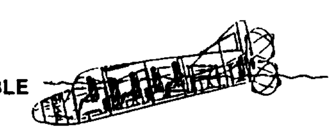


Table 1-3. Why PLS Operations Are More Efficient.

DESIGN FEATURES	OPERATIONS BENEFIT
• SMALL/SIMPLE	• FEWER PARTS COUNT • DESIGN & MAINTENANCE
• ELECTROMECHANICAL ACTUATORS	• NO HYDRAULICS • REDUCED MAINTENANCE
• NON-TOXIC PROPELLANTS	• PARALLEL LAUNCH OPERATIONS
• BUILT-IN TEST & HEALTH MONITORING	• SCHEDULED MAINTENANCE
• ACCESSIBILITY	• PARALLEL MAINTENANCE OPERATIONS • REDUCED MAN-HOURS FOR TURNAROUND
• SMALL, CERTIFIED TURNAROUND CREW	• ECONOMIC UTILIZATION OF SKILLS
• DESIGN FOR OPERATIONS	• REDUCED LOGISTICS SUPPORT
• ROBUST SYSTEM DESIGN	• FLEXIBLE OPERATIONS AT MINIMUM RISK
• STANDARDIZED MISSIONS & PROCEDURES	• REDUCED FLIGHT PLANNING • SIMPLE FLIGHT OPERATIONS
• SAFE ABORT MODES	• CREW SURVIVABILITY FOR ALL CREDIBLE SITUATIONS





2.0 SYSTEM DESIGN PHILOSOPHY AND STUDY GROUNDRULES

This section documents the overall system design philosophy that we have adopted and the groundrules used to conduct the PLS Study portion of the Advanced Manned Launch Systems (AMLS) contract.

2.1 SYSTEM DESIGN PHILOSOPHY

The primary role of the Personnel Launch System is to provide assured manned access to space. In meeting this objective, we have incorporated features into the design which provide for crew safety, cost-effective operations, and high operational utilization. The key to the overall system design is our adoption of airline/aircraft approaches to certification and flight-worthiness: we do not decertify after each flight as is effectively done in the Shuttle program, but rather maintain the system in a flightworthy status. The entire system, glider and support systems, is designed for maintainability and producibility.

In this study, the aircraft/airline approach to aircraft certification and flightworthiness was used as a reference. In this approach, the vehicle and vehicle subsystems are certified one time and regular maintenance is scheduled to maintain that airworthiness. The subsystems, missions, and operations approaches are designed to reflect this basic philosophy - recognizing that this is a major cultural change from the way NASA does business today.

A key feature of the design which results from this philosophy is the provision for easy accessibility for performing turnaround maintenance. Wherever possible, the subsystems are installed outside of a pressurized crew module for direct accessibility during ground support operations. Leeward surface access panels are designed to be removed during the maintenance process. Several features of this basic philosophy as derived from airline/aircraft fleet operations are discussed in the following paragraphs.

2.1.1 Performance Trending

Operationally, the primary concern is to reduce manpower requirements, costs, spares, down-town, and turn-around time thus improving safety, mission reliability, and vehicle availability. This is accomplished mainly by reducing the in-service or unscheduled failures and increasing the out of service or scheduled maintenance actions associated with PLS operations. The most significant improvements to operations can be accomplished by converting the existing unscheduled maintenance actions to scheduled maintenance actions and others as presented in Table 2-1.

The application of data analysis to an operational system must concern itself with anticipation of probable failures and development of methods to determine degradation indicative of these failures (thru data analysis) to allow their prediction and resulting scheduling of maintenance to prevent the in-service

Table 2-1. Performance Trending

WHAT IS PERFORMANCE TRENDING

- OPERATIONAL SUPPORT FUNCTION
  - DETERMINE SPECIFIC DEGRADATION
  - PREDICT SERVICE OR REMOVAL REQUIREMENTS

APPLICATION TO PLS

- CONVERT UNSCHEDULED MAINTENANCE INTO SCHEDULED MAINTENANCE
  - IMPROVED TURN-AROUND AND LOGISTICS
- CONTROL OF INFLIGHT FAILURES
  - IMPROVED MISSION/SAFETY OF VEHICLE
- REFLIGHT VERIFICATION OF VEHICLE
  - REDUCTION IN GROUND PROCESSING
- MANAGEMENT SUPPORT
  - IMPROVED RISK ASSESSMENT

failure of the subject item. This is the primary objective of performance trending.

Use of data analysis in support of performance trending results in determination of specific degradation of individual systems, subsystems and components. This process develops an extensive data base providing insight into the overall performance level of the PLS. This data base provides the necessary qualitative and quantitative health of the total vehicle capabilities necessary to determine reflight verification of the vehicle with a minimum of ground processing while providing management support for improved risk assessment.

2.1.2 Certified Airframe and Powerplant (A&P) Personnel

The present maintenance philosophy used for the Space Shuttle orbiter requires a large pool of personnel highly specialized in very narrow fields of technical capability. This concept dictates a system requiring a significant cadre of multi-discipline support personnel, i.e.: management, engineering, quality, safety, etc. While this maintenance concept requires significant manpower, it typically results in inefficient maintenance operations (requiring extensive downtime) due to limited availability of manpower in specific areas of specialization.

The maintenance philosophy to be applied to PLS is that of using certified A&P personnel (Table 2-2). The typical A&P certification is obtained by attendance in an aviation university requiring a minimum of three years of formal education. A high percentage of those individuals obtaining A&P certification typically complete a full four or more years of education thus obtaining their BS in engineering (maintenance, aeronautical, electrical, etc.) along with pilot qualification and associated management degrees. Utilization of this highly qualified, broadly educated individual allows a maintenance concept which utilizes small numbers of support personnel. Additionally it provides a work force requiring limited specialization resulting in a very efficient maintenance operation (minimum down-time) thru utilization of a broad application of existing personnel.

Table 2-2. Certified Airframe and Power Plant (A&P) Personnel

ORBITER

- UTILIZES NON-A&P PERSONNEL
  - HIGH LEVEL OF SPECIALIZATION
  - REQUIRES LARGE POOL OF MANPOWER
  - REQUIRES EXCESSIVE SUPPORT PERSONNEL
  - RESULTS IN INEFFICIENT MAINTENANCE OPERATIONS
    - LIMITED BY MANPOWER NOT TIME

APPLICATION TO PLS

- UTILIZE CERTIFIED (A&P) PERSONNEL
  - LIMITED SPECIALIZATION
    - AVIONICS
    - ELECTRICAL
    - THERMAL PROTECTION
    - MECHANICAL/SYSTEMS
  - RESULTS IN EFFICIENT MAINTENANCE OPERATIONS

2.1.3 Certification

STS orbiter certification was derived by a design, that in most cases, required the development of all-new components. This requirement to design and certify totally new components even where the items were an outgrowth of existing components utilized in other vehicles (albeit not manned space vehicles) drove orbiter development and operations costs. Here the major shortcoming is the failure to utilize the extensive operational experience data base available for existing components which could be modified for spacecraft application. The use of such components would require only a delta of design and certification therefore reducing the cost in both program time and money involved in the design and testing process (Table 2-3).

Additionally the orbiter certification process did not test a vehicle to destruction. This resulted in an extended flight test applied over a broad conservative operation due to the actual limits being unknown.

Table 2-3. Certification

ORBITER

- LARGE SCALE COMPONENT AND PIECE PART TESTING
  - EXPENSIVE IN BOTH PROGRAM TIME AND MONEY
  - INCONCLUSIVE RESULTS
- VEHICLE NOT TESTED TO DESTRUCTION
  - RESULTS IN CONSERVATIVE OPERATION
- DID NOT TAKE ADVANTAGE OF OPERATIONAL EXPERIENCE
  - CONSTRAINTS ON MAINTENANCE AND OPERATIONS
- REQUIRES EXTENDED FLIGHT TEST

AIRCRAFT

- SMALL SCALE COMPONENT AND PIECE PART TESTING
  - REDUCTION IN BOTH PROGRAM TIME AND MONEY
- AIRFRAME TESTED TO DESTRUCTION
  - INDEPTH KNOWLEDGE OF VEHICLE LIMITATIONS
  - PROVIDES FOR FULL OPERATIONAL CAPABILITIES
- TAKES ADVANTAGE OF OPERATIONAL EXPERIENCE BASE
  - MAXIMIZES MAINTENANCE AND OPERATIONS
- LIMITED FLIGHT TEST
  - SPECIFIC REQUIREMENTS

In the case of aircraft design, most components are developed from existing components which have extensive operational experience data bases. Utilization of this experience base allows reduction of both design and testing efforts and the associated cost in both program time and money. Also, a vehicle is tested to destruction thus providing hard data as to its ultimate capabilities. In this case, a flight test program can be developed which will demonstrate the vehicle to limits less than its ultimate capabilities but in excess of the maximum operational envelope. This results in a flight test program of minimum duration which provides full confidence in the ability to operate the vehicle to the full limits of its operational envelope.

Application of an aircraft philosophy to the PLS certification program also requires the development of man-rated specifications utilizing existing commercial/military components (Table 2-4). This concept will take full advantage of the existing experience data base of selected components resulting in a design/-

Table 2-4. PLS Certification

REQUIRES LOW LEVEL PIECE PART TESTING BY:
- UTILIZING MAN-RATED SPECIFICATIONS
- USING COMMERCIAL/MILITARY COMPONENTS
- TAKING ADVANTAGE OF EXISTING EXPERIENCE BASE

TEST VEHICLE TO DESTRUCTION
- INDEPTH KNOWLEDGE OF VEHICLE LIMITATIONS
- PROVIDES FOR FULL OPERATIONAL CAPABILITIES

REQUIRES LIMITED FLIGHT TEST
- SPECIFIC REQUIREMENTS

certification program which deals mainly with the delta required in meeting the design and testing goals of the PLS.

Additionally, the PLS program will test a vehicle to destruction, therefore, as in aircraft certification, providing its ultimate capabilities. This will allow the establishment of a flight test program which will test the vehicle to limits less than its ultimate, but in excess of its maximum operational envelope, resulting in a vehicle which can be utilized to its full operational capability.

2.1.4 Operational Environment

The reasoning behind the application of aircraft specifications and components, commercial/military, to PLS is based upon the relative similarity of their operating environments (Table 2-5). In many areas, the average aircraft environment is equal to or sometimes more severe than that of a spacecraft. For instance, the thermal cycle of an aircraft in adverse summer operation will quite often extend from as high as 200°F to a low of -65°F, sometimes on a cycle occurring every hour. Conversely most spacecraft internal cavity areas will tend to stabilize at temperatures close to 50°F, with peaks to 150°F during entry. While space craft tend to see large vibration inputs during ascent, their on-orbit vibration is rather benign, whereas an aircraft is constantly exposed to high vibration sometimes exceeding 10G's in

Table 2-5. Operational Environment

AIRCRAFT ENVIROMENTS

- MANY AVERAGES EQUAL OR MORE SEVERE THAN SPACECRAFT
  - THERMAL CYCLE
  - VIBRATION
  - MOISTURE
  - G-LOADS
  - ALTITUDES

PLS ENVIRONMENTS

- MIL SPEC CONVERSION OF AIRCRAFT TO SPACE ENVIRONMENT
  - ELECTRO-MECHANICAL AVERAGES
    - LAUNCH
    - ON-ORBIT
  - SPACECRAFT ENVIRONMENT
    - THERMAL CYCLE

both low and high frequency ranges. While the spacecraft may be called on to operate in a virtual vacuum, the aircraft is not far behind, routinely operating to 51,000 feet for commercial operations and in excess of 90,000 feet for military.

The relative merits of commercial equipment may best be appreciated by looking at the typical certification requirements. Here, most are certified to both DO-160B and MIL-E-5400T.

Temperature:	-65° to +160°F, 30 minutes at 203°F
Vibration:	0.10 DA at low freq (5G), high freq 6G
Shock:	15G 11Ms duration
Altitude:	70,000 feet (1.311" Hg or approximately 4.3% of Sea Level)

2.2 STUDY GROUNDRULES

This section consists of a description of the <u>study groundrules</u> used to perform both the overall study tasks and documentation requirements specified by NASA, and <u>design groundrules</u> for the PLS system. These design groundrules are program-level and project-level requirements, from which lower level requirements are specified in a subsequent requirements allocation process.

2.2.1 Study Groundrules Definition

The following groundrules form the basis for performing the analytical and documentation activities for this study.

General Study Groundrules. The general study groundrules presented in Table 2-6 were derived from NASA PLS documentation (such as the RFP) and groundrules presented at the time of the contract kick-off meeting. They are distinguished from the design groundrules in that study groundrules are derived from the overall study objectives and government direction on assumptions or methods used to conduct the study. They establish the framework from which the NASA task assignments are performed. Design groundrules presented in Section 2.2.2 establish the top-level requirements used in defining the PLS operational system.

Table 2-6. General Study Groundrules

GROUNDRULES	SOURCE
1. THE PLS WILL OPERATE AS A MANNED COMPLEMENT TO THE SPACE SHUTTLE AND OTHER SPACE TRANSPORTATION ELEMENTS. IT SHALL : - BE A LOW DDT&E SYSTEM WITH A LOW COST PER FLIGHT - CAPABLE OF SAFE AND RELIABLE VEHICLE OPERATIONS - INCORPORATE THE USE OF OPERATIONALLY EFFICIENT SYSTEMS	NASA GROUNDRULES DOCUMENT *
2. CHANGES TO THE PLS DESIGN TO ACCOMODATE DRM'S OTHER THAN THAT DEFINED AS "DRM-1" WILL NOT BE CONSIDERED AT THIS TIME.	NASA GROUNDRULES DOCUMENT *
3. THE PLS WILL BE LAUNCHED BY THE TITAN IV	NASA/LaRC DISCUSSIONS (9-26-89)
4. COSTS AND DESIGN MODIFICATIONS REQUIRED TO MAN-RATE THE TITAN IV WILL NOT BE CONSIDERED AT THIS TIME	NASA/LaRC DISCUSSIONS (9-26-89)
5. THE PLS SHALL SUPPORT SPACE STATION FREEDOM CREW TRANSFER	NASA GROUNDRULES DOCUMENT *
6. NO DEVIATIONS FROM THE NASA DEFINED OUTER MOLDLINE WHICH COULD AFFECT AERODYNAMIC PERFORMANCE IS PERMITTED EXCEPT THAT REQUIRED FOR APPLICATION OF TPS (e.g. INCREASE IN FIN THICKNESS)	STUDY KICK-OFF MEETING (9-12-89)
7. ALL PLS DATA SHALL BE PRESENTED IN STANDARD ENGLISH UNITS.	NASA GROUNDRULES DOCUMENT *

* LaRC LIFTING-BODY PERSONNEL LAUNCH SYSTEM (PLS) STUDY GENERAL GROUNDRULES (SEPTEMBER,1989)

Four of the seven study groundrules presented in Table 2-6 were extracted from the lifting-body PLS study groundrules provided by NASA LaRC at the beginning of the study. Of these, the first groundrule in Table 2-6 (specifying that the PLS should be a low-cost, operationally efficient system) is derived from the overall objective of the study.

Two study groundrules address the application of the Titan IV as the launcher for the PLS glider vehicle. These groundrules which evolved from follow-up discussions with NASA, specify that Titan IV and ALS booster characteristics will be used as a throughput into the costing analysis. No attempt will be made in this study to assess the design and subsequent cost impacts to man-rate the Titan IV and the ALS booster.

The overall PLS vehicle outer moldline defined by NASA will be retained throughout this design study. Exceptions to this groundrule are the ability to alter the fin thickness to permit application of TPS or the geometric scaling-up of the vehicle to increase the internal volume capability. Geometry modifications, however, shall not alter the existing wind tunnel aerodynamic/-aerothermodynamic characteristics for this concept, or negate the ability to install it within the geometry constraints of the STS orbiter. This constraint directly influences lower level vehicle design features; however, it has been specified as a study groundrule at this time in order to retain the validity of the existing aerodynamic characteristics.

Figures of Merit. The figures of merit (FOM) to be used to evaluate the relative value of improvement options were selected in direct response to study objectives. These objectives were to conceptually design a PLS that provides a high level of crew safety, is relatively simple to operate, has a high utilization rate, operates efficiently (low cost-per-flight) and is affordable. Measures, or FOMs, were then chosen for each objective. These FOMs were further clarified as noneconomic or economic and were then broken down to provide more specific measures, as shown in Figure 2-1.

Each improvement option will be evaluated in terms of the selected FOMs. The improvements will then be ranked according to each FOM and lower ranked (lower payoff) improvements will be dropped. The higher payoff improvements will then be examined carefully for credibility and additional tradeoffs will be made to clarify evaluation behavior. When complete, Rockwell, using the totality of the evaluation, will recommend which improvements should be integrated into the preferred concept, and will provide supporting rationale. When NASA acceptance is given, the preferred concept's definition will be finalized and its life cycle cost estimated.

MIL-STD Tailoring. The PLS Task Statements referenced specifications to be applied when responding to the data request. A tailoring exercise was performed for each referenced military specification. As a means to initiate the tailoring activity, DOD-HDBK-248A ("Guide for Application and Tailoring of Requirements for Defense Material Acquisitions") was reviewed. This document provided general guidelines and a suggested format to perform the tailoring exercise. In addition, Appendix B of Rockwell's ALS

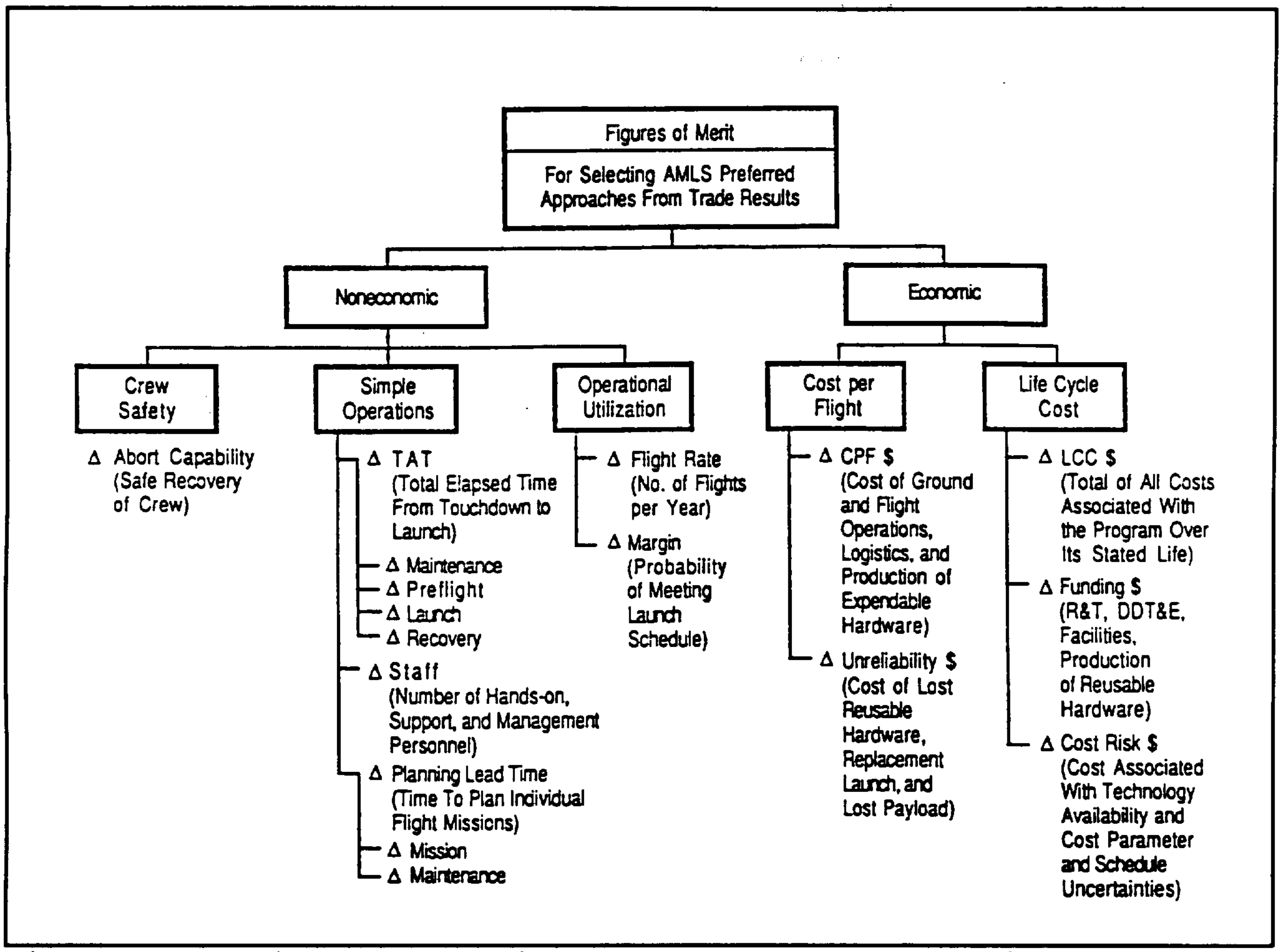


Figure 2-1. Figures of Merit For Trade Study Evaluation

Phase 1 System Design Data Package, which provided specification review sheets from the ALS specification tailoring exercise, was reviewed. Using these documents as guides, a specification tailoring template was developed. Each task leader reviewed the appropriate specifications and recommended modifications or deletions to specification paragraphs with an appropriate rationale. Appendices within several specifications provided a guide to assist in the tailoring by identifying the paragraphs that are appropriate for various program phases. Where these inputs were available, they were used to justify appropriate paragraph deletions.

In compliance with the Subsystems task requirements, specifications DoD-STD-100C and MIL-STD-490A were reviewed and tailored. The drawing practices defined in DoD-STD-100C are adopted as specified, with the exception of the use of English units for the new PLS design. The CAD-produced drawings will use the Rockwell version of the various ANSI Y14 drawing format conventions since it is part of the installed software package. These variations are few and minor. The PLS vehicle/subsystem description will follow MIL-STD-490A, Type A (System/Segment Specification) except for sections pertaining to the requirements of production and delivery of hardware. The paragraphs of the specification MIL-STD-490A other than those relating to the characterization of the PLS vehicle and system will be top level in nature.

The acquisition task requires the development of an Acquisition Plan which extends from concept development through the operational phase of the PLS program. Tailoring of the referenced specifications will be an on-going process as part of refining the acquisition plan prior to the initiation of the next PLS program phase. Several of the referenced specifications such as MIL-STD's-1547A, 1546A, 1540B, DoD-STD-167A, and MIL-Q-9858A are applicable to hardware, and such, will be deferred until hardware procurement in program phases B, and/or C/D. Specifications which are applicable to a pre-phase A type study with the appropriate tailoring are those pertaining to program management, or system engineering, such as MIL-STD-483A.

The recommended tailoring of MIL-STD-1388-1A supports the logistics analysis) for the operations task. The major change in the documentation level reporting is Task 401. This task is normally not required during this phase of a program; however, in-house military aircraft data and STS orbiter data will be used as a point-of-departure to determine task requirements for the cost and operations estimating activities. The remainder of the task definitions were chosen at the level that would normally support a conceptual-development type of analysis. The MIL-HBK-266(AS) requirements are addressed by the Reliability Centered Maintenance (RCM) activity per MIL-STD-2173(AS). This activity is to be closely coordinated between the logistics and reliability/maintainability organizations during the Logistics Support Analysis (LSA) process. RCM factors that drive operations support (MTBF, MTTR, and Availability) are to be evaluated for each subsystem to identify their impact on support and logistics and trades that are required for determination of optimum repair levels/procedures.

The reliability and maintainability standards were reviewed for this effort. MIL-STD-1629A will be tailored to include RCM as adopted by the airlines' Maintenance Steering Group (MSG). All references to weapon system applications are not to be reflected in this analysis. Failure Modes and Effects analysis (FMEA) will be performed on selected high maintenance system/subsystems scheduled for investigation to a phase B level. These specific systems-/subsystem studies will provide additional insight into the reliability, maintainability, and maintenance philosophies being applied to the PLS design process. Abbreviated Reliability Centered Maintenance (RCM) shall also be the integration of design and maintainability engineering and the development of design driven maintenance programs.

- MIL-STD-470A data will cover only those task numbers that apply to this preconcept phase.

- MIL-STD-1543A will include most of the tasks that are evaluated at a Phase B level of documentation.

- MIL-STD-785B will be selectively applied in most of the tasks (except where detailed design data are needed).

- MIL-STD-2173 (AS) will support the RCM activity in Task 1.5 and also supports MIL-STD-1629A in this task. Only minor modification is required except where detailed hardware and operations data would be needed. This standard will support MIL-HBK-266(AS)-type activities.

Adoption of Airline Specifications. During the study, the Airline Transport Association (ATA) coding system, ATA-100, will be studied for possible tailoring to the PLS systems and operations in support of system design breakdown and WBS. This system would provide tracking capability for schematics, maintenance manuals, maintenance specifications, part number system, FMEA, MSG, procurement specifications, design specifications and technical correspondence.

Additional areas of review not applicable to pre-phase A which would be included in future program phases include Aeronautical Radio, Inc. (ARINC) specifications. ARINC provides standards for all major aircraft vendors line replaceable unit (LRU) venders and airline engineering departments. These include design specifications for various types of connectors, interface configurations, environmental requirements, and racking configurations. Other areas of future study shall include application of ATA 300 to the PLS. This specification provides standards for the shipping, handling and storage of flight and GSE hardware including standardization of containers.

2.2.2 Design Groundrules Definition

The design groundrules shone in Table 2-7 were identified as a means to establish the important program- and project-level set of requirements for the requirement allocation process. Since the PLS study groundrules provided by NASA established a thorough listing of groundrules applicable to PLS, the majority of recommended groundrules were extracted from this document. Vol. X, Flight and Ground Systems Specification (Reference 2-1), and the Shuttle-C Requirements Document, were also reviewed as a means to ensure that a comprehensive set of groundrules is established.

Conclusions from prior applicable studies listed in Table 2-8 were also used, either to recommend a new groundrule or to justify an existing one. Many of the reviewers selected to perform this review task were either program managers or heavily involved with these studies. Others are aware of study results as a means to stay abreast of developments in their areas of expertise. The results of this study review activity indicate that the majority of findings from prior studies support the PLS groundrules presented in Table 2-8. New groundrules in Table 2-8 that are traceable solely to results of prior studies are those pertaining to Space Station docking, cleanliness levels, and airline-type operations.

Table 2-7. PLS Design Groundrules

REQUIREMENT NUMBER	DESIGN GROUNDRULE	FUNCTION	SOURCE (1)
1	THE PLS SHALL BE DESIGNED TO ACCOMPLISH DRM-1 AS FOLLOWS: • 8-10 PERSONNEL TO AND FROM SPACE STATION AT 220 NMI AND 28.5 INCL • 72 HR MISSION DURATION • ΔV = 1100 FT/SEC FOR ORBITAL MANEUVERS	DSN REQT	A
2	ALL ELEMENTS SHALL BE MAN-RATED PER JSC 23211(2) WITH SPACECRAFT SYSTEMS DESIGNED FOR FAIL OPERATIONAL/FAIL SAFE OPERATION AND OPERATE WITHIN THE SAFETY REQUIREMENTS SPECIFIED IN JSCM 8080 AND OTHER APPLICABLE DOCUMENTS	DSN REQT	A
3	THE PLS SHALL USE PROVEN STATE-OF-ART COST-EFFECTIVE TECHNOLOGIES AT NASA TECHNOLOGY LEVEL 5 OR BETTER IN 1992	DSN REQT	A
4	THE PLS SPACECRAFT SHALL BE ABLE TO FIT WITHIN THE NSTS ORBITER PAYLOAD BAY (WITH POSSIBLE MINOR DISASSEMBLY)	DSN REQT	A
5	THE PLS SPACECRAFT SHALL BE CAPABLE OF DOCKING WITH SPACE STATION USING STANDARD SPACE STATION DOCKING PROCEDURES WITH REMOTE MANNED CONTROL CAPABILITY	DSN REQT	B, NEW
6	THE PLS SPACECRAFT SHALL BE MANNED WITH A 10 TO 15 PSI N_2/O_2 ATMOSPHERE AND CAPABLE OF TWO PURGES AND TWO REPRESSURIZATIONS PER MISSION	DSN REQT	A
7	THE CLEANLINES LEVELS WITHIN THE PLS CREW MODULE SHALL COMPLY WITH SPACE STATION ENVIRONMENTAL REQUIREMENTS	DSN REQT	NEW
8	THE PLS CREW MODULE INTERNAL VOLUME SHALL ACCOMODATE ALL FLIGHT PERSONNEL (5 TO 95 PERCENTILE) WEARING PARTIAL PRESSURE SUITS	DSN REQT	A, MODIFIED
9	THE PLS SPACECRAFT SHALL BE DESIGNED FOR EASE OF MAINTENANCE	DSN REQT	A
10	AFTER BECOMING OPERATIONAL, THE PLS SHALL USE AIRLINE-TYPE OPERATIONS WITH A PROGRESSIVE PROGRAM OF SCHEDULED HARDWARE & SOFTWARE MAINTENANCE ACTIVITIES	2.0	NEW
11	THE PLS SYSTEM SHALL BE CAPABLE OF SUPPORTING UP TO EIGHT FLIGHTS PER YEAR PER THE BASELINE PERSONNEL MISSION MODEL FROM KSC OR CCAFS	3.0	A
12	THE PLS SHALL HAVE AN ENHANCED ANNUAL LAUNCH PROBABILITY COMPARED TO STS DUE TO WEATHER CONSTRAINTS. THE PLS VEHICLE SHALL ALSO HAVE NIGHT LAUNCH CAPABILITY	3.0	A, MODIFIED

(1) A - LaRC LIFTING BODY PLS STUDY GENERAL GROUNDRULES
B - NASA/LaRC DISCUSSIONS (9/26/89)
C - JSC-31017, CERV SPRD

(2) "GUIDLINES FOR MANNED SPACE SYSTEMS"

Table 2-7. PLS Design Groundrules (Continued)

REQUIREMENT NUMBER	DESIGN GROUNDRULE	FUNCTION	SOURCE(1)
13	THE PLS SHALL HAVE CONTINUOUS ABORT CAPABILITY FROM THE TIME OF CREW ACCESS ARM RETRACTION THROUGH ON-ORBIT OPERATIONS	4.0, 6.0	A, MODIFIED
14	THE PLS AND ADAPTER SHALL NOT PRODUCE LONG LIFETIME ORBITAL DEBRIS	5.0	A
15	ACCELERATIONS SHALL NOT EXCEED THE FOLLOWING VALUES: $\ddot{X}$ (G'S) / $\ddot{Y}$ (G'S) / $\ddot{Z}$ (G'S) ASCENT: 4 / 1 / 0.5 ORBIT/ENTRY/LANDING: 3 / 1 / 0.5 LAUNCH ESCAPE: 8 WATER IMPACT (ABORT): 15 / 10 / 5.0 THE PRODUCT OF LOAD FACTOR AND TIME SHALL NOT BE DETRIMENTAL TO DECONDITIONED FLIGHT PERSONNEL	5.0	A
16	THE PLS SPACECRAFT SHALL HAVE A MAXIMUM LANDING CROSSRANGE CAPABILITY OF 1100 NMI AND BE ABLE TO LAND ON AN 11,000 FT LONG RUNWAY IN DAYLIGHT OR AT NIGHT USING AVAILABLE MICROWAVE LANDING SYSTEM OR OTHER EXTERNAL GN&C ASSETS	5.0	A, MODIFIED
17	RECOVERY SYSTEMS ON THE PLS SPACECRAFT SHALL ENGAGE AUTOMATICALLY TO PROVIDE A SAFE ENVIRONMENT FOR THE CREW UNTIL RESCUE, FOLLOWING NOMINAL AND ABORTED MISSIONS. THIS CAPABILITY SHALL APPLY FOR LAND AND WATER, NIGHT AND DAY, AND ALL WEATHER CONDITIONS	6.0, 7.0	A
18	THE PLS SPACECRAFT DESIGN SHALL PROVIDE FOR QUICK CREW EGRESS AUTONOMOUS OF GROUND CREW SUPPORT ON THE LAUNCH PAD AND FOLLOWING LANDING FOR NOMINAL AND ABORT MISSIONS	7.0	A
19	THE PLS SPACECRAFT SHALL BE CAPABLE OF BEING FERRIED BY LAND, SEA, AND AIR USING EXISTING COMMERCIAL OR MILITARY TRANSPORT SYSTEMS WITH MINIMUM SPECIALIZED GSE	8.0	A
20	THE INITIAL OPERATIONAL CAPABILITY SHALL BE BY 2000 WITH A 20 YEAR OPERATIONAL LIFE AND REDUCED LIFE CYCLE COSTS	11.0	A
21	THERE SHALL BE ADEQUATE SPARES TO AVIOD CANNIBALIZING AND PROVISIONS SHALL BE MADE FOR ATTRITION	12.0	
22	THE PLS VEHICLE SHALL HAVE THE CAPABILITY OF AUTONOMOUS OPERATIONS PRE-LAUNCH THROUGH LANDING		A

(1) A - LaRC LIFTING BODY PLS STUDY GENERAL GROUNDRULES
B - NASA/LaRC DISCUSSIONS (9/26/89)
C - JSC-31017, CERV SPRD

Table 2-8. Prior and On-going Studies Reviewed

Studies for Required Review
NASA Shuttle II Study
Shuttle Ground Operations Efficiencies/Technologies Study, March 21, 1989, Boeing, Contract NAS10-11344, NASA/KSC
Operationally-Efficient Launch Site (OELS) Study Final Report (05-88-KSC-016), October 1988, Vitro Corporation, Contract NAS10-11436, NASA/KSC
Advanced Launch Systems (ALS) Design Study, Phase 1 System Design Review (STS 88-0686), June 1988, Rockwell International, Contract F04701-87-C-0139, AF/SD
Space Transportation Architecture Study (STAS) (STS 87-0532), November 16, 1987, Rockwell International, Contract F04701-85-C-0158
NASA/JSC Design Goals and Technology Requirements for Future Launch Systems Final Report (88-187), April 19,1988, Eagle Engineering/Lemso, Contract NAS2-17900, NASA/JSC
Air Force Structural Definition Study, Contract F33615-87-C-3243, Rockwell International, 1987
National Aerospace Plane (NASP), Contract F33657-86-C-2127, AF/NASA Joint Project Office
Space Transportation Main and Booster Configuration Studies, Phase A, NASA/MSFC
Reducing Launch Operations Costs (New Technologies and Practices) (OTA-TM-ISC-28), September 1988, Office of Technology Assessment
Crew Emergency Return Vehicle (CERV) System Performance Requirements Document (SPRD), JSC-31017, November 9, 1988

The recommended design groundrules presented below are ordered by the function to which they apply per the preliminary top-level functional flow diagram shown in Figure 2-2. When defining groundrules, a typical issue that must be addressed is whether a groundrule is actually a lower level requirement. Since this is usually subjective, rationale should be provided to justify each proposed groundrule. The following paragraphs provide this rationale for each proposed groundrule.

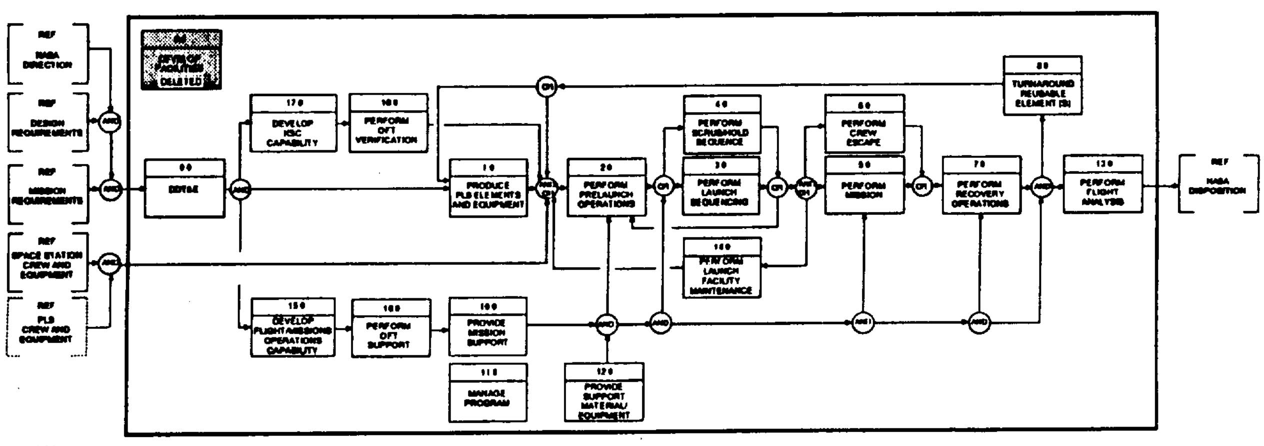


Figure 2-2. Functional Flow Block Diagram

<u>PLS Designed to Accomplish DRM-1</u>. Design reference missions are important to the design of the system, especially that of the flight vehicle, since they guide the further definitions of many of the functional requirements. In particular, they guide the sizing

of the spacecraft power, propulsion, and life support systems and provide the basis for launch vehicle performance requirements. For the initial phase of this study, only DRM-1 is to be considered. This DRM requires the PLS to provide for crew rotation at Space Station, which also implies that the PLS spacecraft design must be compatible with Space Station requirements. In addition, reference missions also serve as an operations baseline against which the vehicle design can be measured. Implicit in the 72 hour mission duration of DRM-1 is that 2 crew and 8 passengers will enter the SSF following docking and power down of the PLS. Critical systems functions which will allow the PLS to remain functionally independent of SSF may remain active during this powered down phase.

Man-Rated Elements/Safety. The fact that the PLS vehicle is to be manned requires that all elements be man-rated, which affects the design of all hardware of the system. It requires that the hardware design has appropriate safety factors for adequate design margins, high reliability, and minimal hazardous or highly toxic materials. It also requires quality assurance methods, redundancy in critical systems, and a level of fault tolerance, specified as fail-operational/fail-safe for the PLS. This is required for crew safety, as specified in applicable documents such as JSCM 8080 (Manned Spacecraft Design Criteria and Standards), and KHB 1700.7A (STS Ground Safety Handbook).

Technology Level. The required technology level helps determine the number of options available to the subsystem designer when attempting to satisfy functional requirements within cost and schedule risks constraints. The requirement for PLS is NASA Technology Level 5 (later raised to Level 6) or better in 1992. This requires that the component or a brass-board model has been tested in the relevant environment. As a point of comparison, the following provides the definition for the various technology levels:

- Level 1 - Basic principles observed and reported
- Level 2 - Conceptual design formulated
- Level 3 - Conceptual design test performed analytically or experimentally
- Level 4 - Critical-function breadboard demonstration
- Level 5 - Component or brass-board model tested in relevant environment
- Level 6 - Prototype or brassboard model tested in relevant environment
- Level 7 - Engineering model tested in space
- Level 8 - Baselined into production design

Typically, the first three technology levels are considered technology development while the fourth to seventh level are advanced development. Technology Level 8 is off-the-shelf technology, which could be modified to satisfy unique design requirements.

Spacecraft to Fit Within the Shuttle Payload Bay. The requirement that the PLS spacecraft shall fit within the Shuttle orbiter payload bay places strict limits on the dimensions of its outer mold-line. This, in turn, puts limitations on the available volume and allowable size of spacecraft subsystems.

Spacecraft Docking to Space Station. In order to achieve Space Station crew rotation as specified in DRM-1, the PLS spacecraft will require the capability of docking to Space Station. This requires rendezvous maneuvering, a docking mechanism compatible with Space Station, and cold gas RCS thrusters for proximity operations. An alternative to a hard-docking system is using a Berthing technique with manipulator arms providing the final closing maneuver. Since the PLS will be manned as specified in DRM-1 while it is approaching the Space Station, it is anticipated that a similar docking procedure that the STS orbiter employs will be retained. However, due to dimensional differences between the PLS and STS orbiter, it is assumed that the PLS will use a different docking port and docking interface hardware than the orbiter.

Manned/Cabin Atmosphere. Requirement 2 requires a man-rated design, which typically affects mechanical, avionic, propulsion, and structural design characteristics. This requirement addresses the need for the life support of the flight personnel. The capability to purge and repressurize will allow multiple on-orbit EVA's, if they should become necessary.

Crew Module Cleanliness. As a consequence of the PLS docking with the Space Station as defined in DRM-1, it is important that the PLS spacecraft comply with Space Station environmental requirements. This requirement affects prelaunch operations in terms of accessibility to the crew module and design of the crew access arm.

STS payload processing is accomplished in a Class-100,000 clean room. If launched as a STS payload, this clean room requirements will impact the PLS processing cost. This is due to additional cleanliness provision and procedures required in the Operations and Checkout (O&C) building. However, Since the PLS is not a STS payload in DRM-1, these additional costs will not be addressed in this study.

Crew Module Internal Volume. Just as Requirement 4 sizes the external dimensions of the PLS spacecraft, this requirement sizes the interior of the crew module and places further restrictions on the allowable sizes of spacecraft subsystems. Since all flight personnel will wear partial pressure suits during all critical flight phases in case of cabin depressurization, it is necessary to allow sufficient space to accommodate them. The specification for 5- to 95- percentile personnel sizes has been derived from Space Station requirements and has been used previously in other manned vehicle design studies.

Spacecraft Designed for Ease of Maintenance. Designing for ease of maintenance reduces turnaround time and launch delays due to equipment failures. This requirement affects the design of all spacecraft subsystems by ensuring that they are easily accessible and repairable. This will result in built-in test equipment, modular subsystem components, low-maintenance TPS, and possible elimination of hydraulics and APU's. This requirement also has implications for the allocation of mean time before failure and mean time to repair.

Efficient Operations. Advanced launch system operational approaches are required to ensure efficient and thus low cost PLS operation. One such operational approach which offers promise is applying, where appropriate, methodologies and techniques from the airline industry to PLS ground processing. With this approach, routine verification will be replaced by hardware and software performance trend analysis and monitoring.

Flight Rate and Launch Sites. The specification of flight rate is important in determining booster production rate, number of spacecraft in the fleet, number of launch pads, and scheduling for crew training and ground support. The identification of launch sites determines the orbital inclination range, which directly affects launch vehicle performance requirements/PLS spacecraft injected weight, and is a significant factor in specifying logistic requirements.

High-Launch Probability. Launch probability is dependent on launch vehicle design margins and launch site weather statistics. By designing the flight vehicle to be able to launch under adverse weather conditions (temperature, wind, and rain), its ability to meet target launch dates and launch windows is significantly increased. An advantage of having the PLS launch site at KSC and Cape Canaveral Air Force Station (CCAFS) is that extensive weather statistics are available to establish accurate weather requirements. This requirement will be an important design consideration for the design of the first-stage flight control and guidance hardware and software. The ability to launch (and to recover, following an abort) at night will also increase launch probability and will potentially reduce the length of launch delays.

Continuous Abort Capability. A significant flight crew safety issue is the ability of a manned flight vehicle to safely perform aborts. The requirement for continuous abort capability maximizes the probability of safe crew return. This will necessitate the capability of on-the-pad aborts. This groundrule will have a direct impact on the design of the abort and recovery subsystems hardware and software and the ground recovery operations.

No Long Lifetime Orbital Debris. A growing concern for designers of spacecraft in low earth orbit is the increasing population of man-made debris in earth orbit. This debris ranges in size from small particles to large upper stages and nonfunctioning satellites and is distributed nearly uniformly in orbital inclination in low orbits (less than 250 NMI). Concern stems from the fact that even small particles can do significant damage to spacecraft, due to hypervelocity impact. The probability of such impact increases with the size and duration of a satellite in low earth orbit -- characteristics that apply to Space Station. An effective way to prevent the increase of this population is to design upper stages, spacecraft, and separation devices to preclude the generation of debris. This is a policy that NASA has agreed to pursue.

Acceleration. The maximum accelerations and exposure times to accelerations are important considerations for crew safety and subsystem design. The maximum values for ascent are consistent with Titan IV and ALS capabilities. Those for other nominal mission phases were chosen to be acceptable for deconditioned crew personnel. The peak values for launch escape and water impacts, both of which would be encountered only during aborts, are of very short duration.

Landing Capability. The requirements for landing capability are important to both flight vehicle design and landing site characteristics. Specification of cross-range capability provides the lift-to-drag ratio required by the spacecraft and figures in the availability of a particular landing site during any given period. This availability is an important factor when satisfying an abort-from-orbit requirement.

The runway length required for safe landing at alternate landing sites is an important consideration for emergency conditions. The use of existing landing aids, which should reduce DDT&E and operational costs, is important to the design of the on-board landing system and should increase the number of acceptable landing sites.

Recovery Systems. The safe recovery of the spacecraft and flight personnel following nominal and aborted missions is an important consideration for the design of the PLS spacecraft and recovery operations. This capability must be provided for land and water, day and night, and a wide range of weather conditions. The ability to automatically engage appropriate recovery systems on the PLS such as location beacons and stabilization devices enhances crew safety, especially following an abort or landing at a remote site. The stabilization floatation devices must be sufficient to expose an access hatch following water impact.

Quick Crew Egress. A lesson learned from the Apollo program is the importance of a manned spacecraft to provide quick egress of flight personnel on the launch pad and following landing. This is especially true for aborts in which conditions may exist where egress is critical for flight personnel survival. This is an important consideration in the spacecraft design (specifically hatch size and location) and has obvious safety implications.

Ferry Capability. Since the PLS spacecraft is the only reusable element of the PLS flight vehicle, its ability to be ferried easily is an important factor to providing rapid turnaround capability. By specifying that existing transports are used with a minimum of specialized GSE, the PLS vehicle turnaround time should be minimized. It also reduces the amount of PLS-unique ground support required for turnaround operations, which decreases operating costs.

Schedule and Cost. The development of credible cost and schedule estimates are necessary to provide design and program decision information. In addition to nominal life-cycle-cost estimates over an agreed upon milestone and operational schedule, a cost risk estimate is required along with cost/benefit analyses for each major technological and for process innovation incorporated into the final, preferred design. The latter data provide the cost arguments for the innovations by providing the negative cost impacts that could be felt by a nominal PLS program if a particular innovation was not incorporated.

The specification of the IOC enables the determination of the schedule of important program milestones. It also has an indirect influence on the test program type and duration which must be performed to demonstrate, at a high confidence level, the maintainability and reliability goals. The ATP for phase C/D which corresponds to this 2000 IOC will be derived from the study based on a development schedule which provides a low risk, low cost development program.

Adequate Spares. An important lesson learned from the Space Shuttle program is the importance of logistics, especially with regard to adequacy of spares for all vehicle systems. By having adequate spares to avoid the necessity of cannibalizing another flight vehicle for replacement parts, the PLS program can significantly reduce the chances of having "hanger queens" and can increase the chances of retaining a full fleet of operational and flight-ready vehicles.

Autonomous Vehicle Operations. The ability of the PLS vehicle to perform autonomously (i.e., independently) from ground mission control has significant implications for the design and operations of several PLS functional areas and vehicle subsystems such as GN&C, data processing, and health monitoring. Synergy exists with the capability and reduced ground check-out during vehicle processing due to having on-board fault detection and isolation at the component level.

Although the vehicle is capable of automatic operation throughout all mission flight phases, the ability of the flight crew to command and monitor automatic mission sequences and to take over active control will be provided. Automated mission sequences which will have crew monitoring and take-over capability are pre-launch, launch to the desired orbit, abort, performing necessary on-orbit maneuvers including docking with the SSF, executing the de-orbit burn, entry control, and approach and landing at the selected landing site.

3.0 RELIABILITY/MAINTAINABILITY ANALYSIS

The Reliability/Maintainability (R/M) Analysis Methodology applied to the Personnel Launch System (PLS) represents the culmination of a multi-year Rockwell SSD IR&D activity directed at the problem of how to credibly perform R/M analyses for conceptual and preliminary designs, i.e., those designs for which detailed design specifics are minimum or absent and which, further, are subject to substantial configuration change over a short time period. Credible R/M analyses, in the context used herein, imply that significant estimating errors and/or oversights do not exist, and that if the design under study was subsequently to be built and flown its operationally observed R/M attributes would closely approximate the conceptual phase estimates.

Further, to be useful the R/M analysis tool used during conceptual phases must be capable of rapid update as the configuration evolves and, further, it must be capable of responsively providing R/M-influenced data required by analysts from other disciplines (Logistics, Life-Cycle Costing, etc.)

Matrix is the R/M analysis methodology developed for such applications. First funded in FY 1989 as IR&D Project #89140, continued work was authorized as FY 1990 IR&D Project #90140 entitled "Advanced Spacecraft Reliability and Maintainability Trade Methodologies."

We began our assessment of the PLS to determine its support requirements during turnaround operations at KSC. We also evaluated the vehicle from a maintainability and reliability standpoint to assure that we could maintain and operate the PLS in a cost effective manner. One of the key items required by maintainability was the requirement for performing test and vehicle on-board checkout without requiring an elaborate ground test system environment (similar to Launch Processing System (LPS) at KSC). The primary requirement provided in the PLS Guidelines was to have an autonomous vehicle. This requires lower level requirements that would drive testability, and diagnostic subsystem requirements. Our goal for reducing costs and minimizing manpower and resources to support the PLS led us to use an airline type operation for performing turnarounds. We have developed an extensive data base that indicates the feasibility of this concept in the ground operations section of the report.

The R/M analyses performed during this phase of the study were used as a tool to influence the PLS design with respect to its maintenance and operational impacts based on subsystem and system configurations. Quantitative and qualitative R/M analyses provided the "yardstick" to measure the degree of R/M and Supportability inherent in a given design. Additionally, this analysis yielded data essential to the LCC estimation process in the area of spares, repair, and manpower requirements.

3.1.1 Top-down R/M analysis

The top-down parametric analysis technique illustrated in Figure 3-1 is a direct result of Rockwell IR&D conducted in FY 1989. The technique was designed to yield credible and defensible R/M numerical estimates for conceptually-defined spacecraft at both the vehicle and system levels, and to require as inputs only the gross design characteristics available at the time the analysis is conducted.

Top-down derived numerical R/M values did not represent a final result. They "bracketed" the range of achievable R/M values, and thus "scope" the magnitude of potential R/M attributes by direct comparison with known vehicles and systems for which well-documented R/M histories were available. The top-down technique's strength and utility stem from the fact that a definite numerical relationship exists between an aerospace vehicle's design parameters (weight, function, mission duration, etc.) and its resultant reliability achievements and maintenance expenditures. Table 3-1 identifies the Air Force reported man-hours per maintenance action (MH/MA) - the number of manhours the average repair requires) for a range of USAF aircraft. Note that none are lower than the C-5's value of 4.69 MH/MA, and that the very sophisticated F-15C is the highest at 7.34 MH/MA. It would seem prudent to be within these two extremes when estimating MH/MA for a new design unless, for example, the design differed markedly from the design practices employed for new aircraft.

Table 3-1. Aircraft Maintenance Man-Hour Predictions Provide Operations Insight for Future Aircraft

Aircraft	MH/MA
F-15C	7.34
B-1B	7.06
T-39	6.44
F-16A	6.37
E-4B	5.53
C-141	5.39
C-5A	4.69

What seems to be evident is that densely-packaged aircraft (such as fighters) offer little in the way of accessibility for maintenance and, accordingly, require an appreciable number of manhours per repair. On the other hand, large aircraft are not as densely packaged and require fewer manhours per repair.

The PLS preferred design concept has a low density and the MH/MA values are warranted and justifiable. This is mainly due to the elimination of hydraulic systems and turbo-mechanisms for

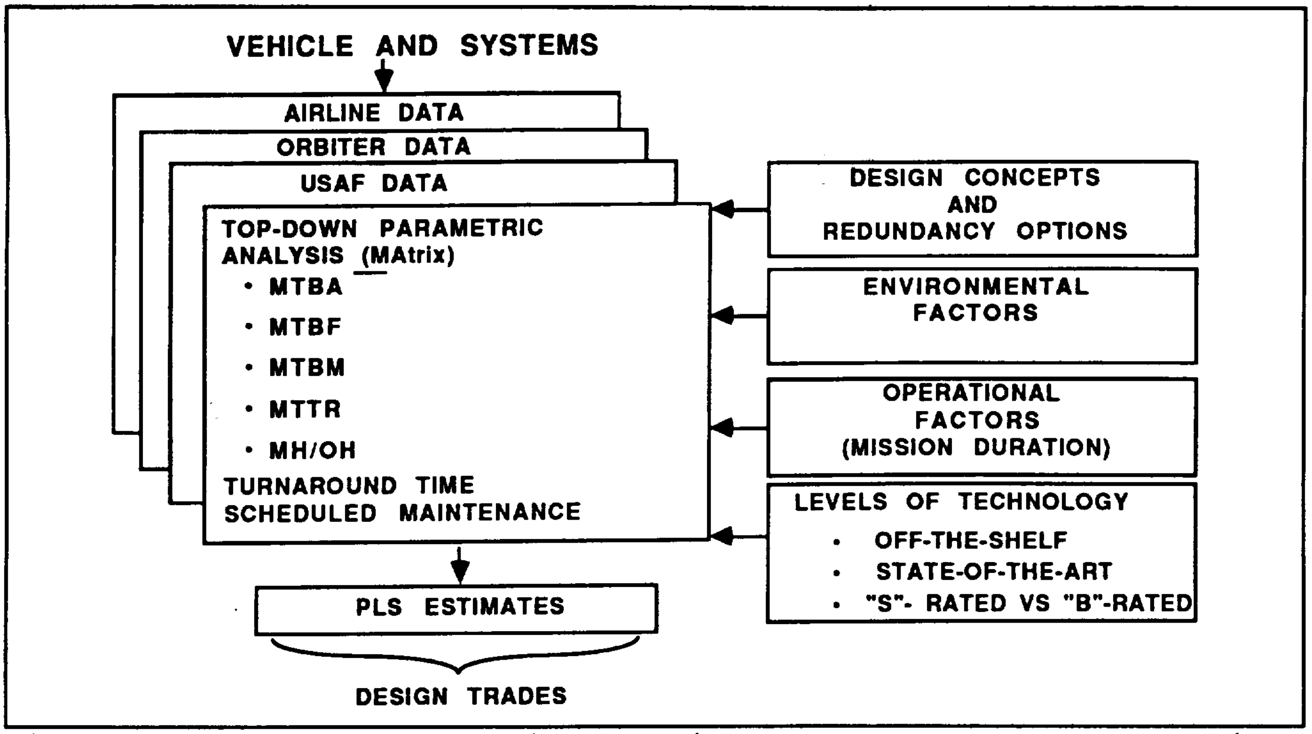


Figure 3-1. Top-Down Analysis at Vehicle and Subsystem Level Provides Direct Link to R/M History

propulsion. Our estimates have been set at 4.5 MH/MA based on these design attributes.

The data base for PLS has been developed using extensive reliability and maintainability data from contemporary aircraft, subsystems and components (see Table 3-2). The following list of aircraft and the number of flights that are available from our data set at Rockwell was used in evaluating the PLS reliability and maintainability characteristics. In contrast the Shuttle orbiter has less than 40 flights for estimating mature system R/M values. Therefore, we have tailored our values from this data base and applied space system R/M factors.

3.1.2 Bottom-up R/M analysis

The bottom-up analysis, depicted in Figure 3-2, provides the rationale to confirm or refute the top-down analysis result. In the bottom-up analysis LRU and component-level estimates are summed "upwards" to subsystem, system and vehicle-level values. It is not expected that the upwards summation will precisely coincide with the top-down result since:

1. The full complement of LRU's and assemblies will not have been identified during the study. This tends to give optimistic "bottom up" R/M results.

2. Frequently, items such as connectors, cables, etc. are omitted from the bottom-up analysis. This, too, leads to a optimistic "bottom up" result.

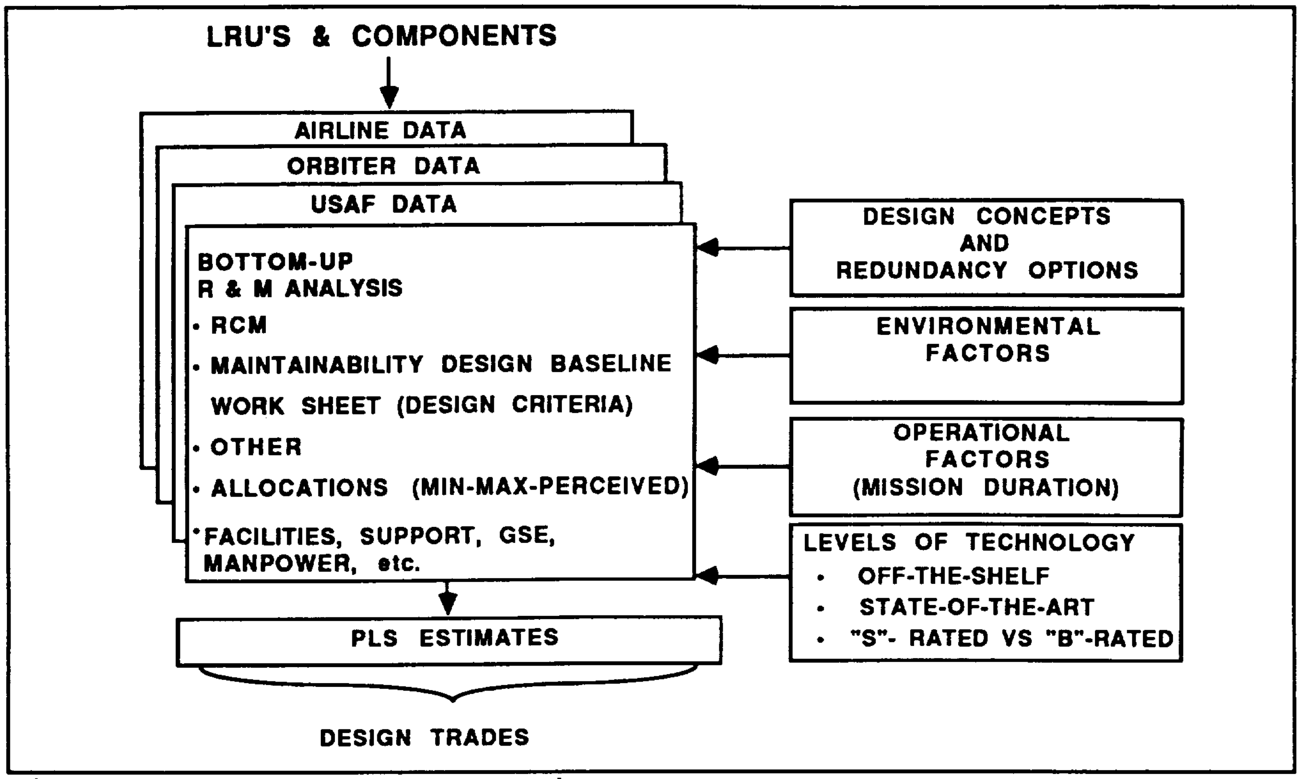


Figure 3-2. Bottom-up Analysis at LRU/Component Level Adds Depth and Credibility to the Parametric Results

Table 3-2. Aircraft Data Base Used for PLS R/M Assessment

Aircraft	Flights
C-5A	32143
C-141A	181114
F-16C	109125
F-15C	78372
FB-111A	8833
OV-10	66298
AC-130H	2565
B-52G	41372
B-1B	72746
A-10A	22182

The Reconciliation Phase of the R/M analysis (Figure 3-3) seeks to rationally account for the differences between the top-down and the bottom-up results, and may result in modifications being made to either, or both, results. Outputs from the Reconciliation Phase "drive" Repair Level Analyses, Spares, LCC estimates, etc.

The R/M Analysis process yields the steady-state R/M values that can confidently be expected when the PLS is put into operations. Our analysis also includes an estimate of reliability growth, which is based on orbiter experience.

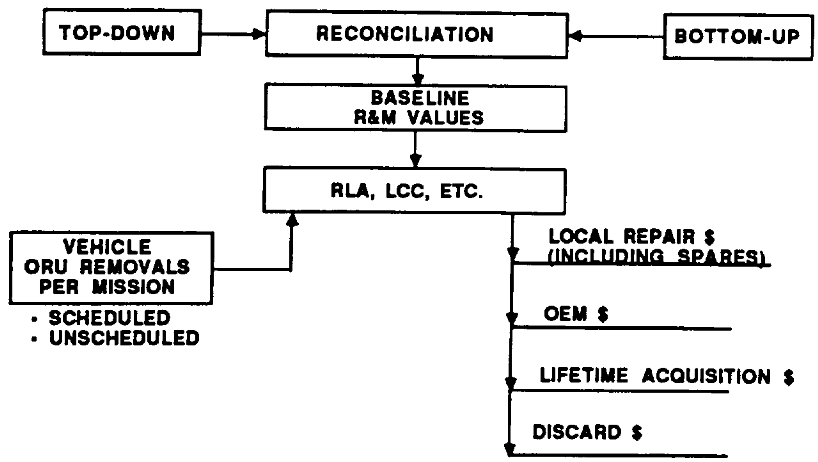


Figure 3-3. R/M Data Base Reconciled Through Iterative Process

SYSTEM (LEVEL 1) / SUBSYSTEM (LEVEL 2) / COMPONENT (LEVEL 3)	UNIT WT (lbs)	MTBM (LVL1)	MTBM (LVL 2)	MTBF (LVL1)	MTBF (LVL 2)	MTBA (LVL1)	MTBA (LVL 2)
STRUCTURE	4136	139		348		109496	
WING GROUP	1059		544		1360		428531
TAIL GROUP	62		9259		23148		7293685
BODY GROUP	3015		191		478		150458
FUSELAGE							
COCKPIT							
THRUST STRUCTURE							
TANKAGE							
BODY FLAP							
PARCHUTE SYSTEM	1145	503		1258			
THERMAL PROTECTION	3423	168		420		132340	
EXTERNAL TANK INSULATION							
PURGE AND VENT							
PROPULSION	979	168		420		95194	
RCS	235		700		1750		396641
OMS	744		221		552		125225
POWER	1425	55		138		18121	
SUPPLY	400		195		448		64247
DISTRIBUTION & CONTROL	1025		76		190		25040
MECHANICAL	964	56		140		32158	
SURFACE CONTROLS	327		191		478		32158
LANDING GEAR	637		79		198		INFINITE
AVIONICS	815	17		42		207317	
ENVIRONMENTAL CONTROL	833	10		25		5430	
PERSONNEL PROVISIONS	2085	79		198		174140	
	16769	4.447		11.117		3245	
					R=	0.978	

Figure 3-4. We See a Highly Reliable and Maintainable PLS Design

3.1.3 PLS Top-down R/M assessment

Figure 3-4 presents the results of our preliminary top-down R/M assessment of the current PLS configuration. It shows that the PLS has the potential to realize an mean-time-between-maintenance (MTBM) of approximately 4.4 hours for a 72 hour mission. (The MTBM value is not a constant: it varies with flight duration). The preliminary MTBM value of 4.4 hours is subject to considerable variation as various design options are examined in our Trade Study activity. For example, additional design redundancy will improve mean-time-before-abort (MTBA), but will decrease MTBM (more maintenance required). Technology issues may also drive MTBM one way or the other.

The average system-level mean-time-to-repair (MTTR) is estimated to be 2.5 hours, and each unscheduled maintenance task is estimated to require an average of 1.8 men. Therefore our projected MH/MA equals 4.5 (i.e., 2.5 x 1.8). The predicted unscheduled maintenance actions/flying hour (UMA/FH) equals 0.2273 (i.e., 1 divided by the MTBM of 4.4 hours) times 4.5 MH/MA = 1.02. For a 72 hour mission a total of 73.44 (i.e., 1.02 x 72) unscheduled maintenance man-hours is estimated.

Historically, for aerospace vehicles in general, scheduled maintenance is approximately 55% of unscheduled maintenance. This equates to 40.39 scheduled man-hours for a 72 hour mission. Total man-hours for a 72 hour mission (scheduled and unscheduled) therefore is estimated to be 113.83.

3.1.4 PLS Bottom-up R/M assessment

Figure 3-5 presents the results of our preliminary bottom-up analysis. This information is based on airline and military aircraft R/M data. These data favorably compare with our original top-down analysis (see Figure 3.1.3-1). The bottom-up assessment includes the parameter mean-time-between-removal (MTBR), used for the determination of the range and depth of spares required to support the PLS at specified flight rates.

The PLS configuration used for the bottom-up assessment was estimated to weigh 15,891 lbs (dry), whereas its weight during the much earlier top-down assessment period was 16,769 lbs (dry). The bottom-up estimated MTBM as 4.97 hours (top-down it was 4.447), and MTBF is now 10.52 hours versus the earlier 11.117 hours. No significant or unexplained differences exist at the subsystem level.

Unscheduled man-hours/flying hour (UMA/FH) therefore equals 0.2012 (i.e., 1 divided by the MTBM of 4.97 hours) times 4.5 MH/MA = 0.905. For a 72 hour mission a total of 65.16 (i.e., 0.905 x 72) unscheduled maintenance man-hours will be required. Scheduled maintenance = 55% of 65.16, or 35.84 man-hours for a 72 hour mission. Total man-hours for the 72 hour mission is 65.16 + 35.84 = 101.0.

WBS	NOMENCLATURE	QTY	UNIT WT (lbs)	TOTAL WT (lbs)	ACFT MA'S/HR	SPACE MA'S/HR	MTBM (Flt Hrs)	MTBR (Flt Hrs)	MTBF (Flt Hrs)
1.7.12	ACTUATORS SYSTEM		-	173	0.005761	0.002769	361	1,445	687
	ACTUATORS								
	WING ELEVON	2	12	24	0.000799	0.000384	2,604	10,416	4,948
	UPPER ELEVON ACTUATOR	2	6.5	13	0.000433	0.000208	4,808	19,232	9,135
	LOWER BODY FLAP ACTUATOR	2	6.5	13	0.000433	0.000208	4,808	19,232	9,135
	WING FOLD ACTUATOR	2	6.5	13	0.000433	0.000208	4,808	19,232	9,135
	NOSE LANDING GEAR ACTUATOR	1	1.5	1.5	0.000050	0.000024	41,667	166,668	79,167
	MAIN LANDING GEAR ACTUATOR	2	2	4	0.000133	0.000064	15,625	62,500	29,687
	TOP HATCH COVER ACTUATOR	1	1.5	1.5	0.000050	0.000024	41,667	166,668	79,167
	RUDDER	1	3	3	0.000100	0.000048	20,833	83,332	39,583
	CONTROLLERS								
	WING ELEVON CONTROLLER	2	19.5	39	0.001299	0.000625	1,600	6,400	3,040
	LOWER BODY ELEVON CONTROLLER	2	10.5	21	0.000699	0.000336	2,976	11,904	5,654
	UPPER BODY ELEVON CONTROLLER	2	10	20	0.000666	0.000320	3,125	12,500	5,937
	WING FOLD CONTROLLER	2	10	20	0.000666	0.000320	3,125	12,500	5,937
1.7.13	AVIONICS	-	-	1337	0.085343	0.041032	24	52	49
1.7.13.1	AVIONICS (GN&C)	-	-	316	0.025529	0.012273	81	171	163
	STAR TRACKER	1	42	42	0.003340	0.001606	623	1,308	1,246
	IMU	2	15.5	31	0.002465	0.001185	844	1,772	1,688
	MICROWAVE LANDING SYSTEM	1	70	70	0.005566	0.002676	374	785	748
	GPS RECEIVER	2	8.5	17	0.001754	0.000843	1,186	2,491	2,372
	ALTIMETER	2	7	14	0.001113	0.000535	1,869	3,925	3,738
	AIR DATA PROBE	1	14	14	0.001113	0.000535	1,869	3,925	3,738
	AIR DATA SYSTEM ASSEMBLY	1	14	14	0.001113	0.000535	1,869	3,925	3,738
	DOCKING SYSTEM	1	56	56	0.004453	0.002141	467	981	934
	EARTH SENSORS	2	12.5	25	0.001988	0.000956	1,046	2,197	2,092
	HORIZON SENSOR ELECTRONICS	2	9.5	19	0.001511	0.000726	1,377	2,892	2,754
	HORIZON SENSOR HEAD	2	7	14	0.001113	0.000535	1,869	3,925	3,738
1.7.13.2	AVIONICS (COMMUNICATION/TRKING)	-	-	208	0.016541	0.007953	126	264	252
	AUDIO TERMINAL	2	7	14	0.001113	0.000535	1,869	3,925	3,738
	HEADSET/MIKE	2	3	6	0.000477	0.000229	4,367	9,171	8,734
	S-BAND SYSTEMS								
	S-BAND ANTENNA SWITCH	2	1.5	3	0.000239	0.000115	8,696	18,262	17,392
	S-BAND ANTENNA	2	6	12	0.000954	0.000459	2,179	4,576	4,358
	S-BAND PREAMPLIFIER	2	14	28	0.002227	0.001071	934	1,961	1,868
	S-BAND POWER AMPLIFIER	2	21	42	0.003340	0.001606	623	1,308	1,246
	S-BAND TRANSPONDER	2	28	56	0.004453	0.002141	467	981	934

Figure 3-5. MAtrix Projection Demonstrates PLS Maintainability

Based on these data it is evident that the PLS as configured can be considered a low maintenance, highly reliable, and affordable system. This is mainly due to its mission character. It operates more like an aircraft than the Shuttle and it is one-tenth the orbiters size. It also is planned to have state-of-the-art systems integrated into a autonomous network that reduces manpower intensive maintenance and diagnostics.

3.1.5 Design Evolution

As the PLS design has matured, so have the estimates for maintenance. Our model is used for estimating design impacts on a system where most models are use to reallocate the R/M values based on a stated reliability number. Table 5.1.5-1 illustrates this based on the three design cycles that we went through during the study.

Table 3-3. Dynamic R/M Process Tracks Evolving Design

Baseline	MTBM	MTBF	MTBR
1/25/90	5.03	20.30	10.74
4/ 1/90	5.88	24.05	12.49
6/27/90	4.92	16.66	10.49

As illustrated in the table, we have increased the maintenance requirements for PLS, however, the values have not grown radically but have increased due to the added weight and new system additions (avionics, air re-vitalization, dual propulsion, batteries, electrical distribution, etc.).

4.0 OPERATIONS AND SUPPORT

Our study of the PLS vehicle has centered on the philosophy of "Design for Operations". During this effort we have been actively involved in developing a support package that covers the disciplines of; Reliability/Maintainability, Ground Operations, Flight Operations, and Supportability (logistics). Our primary goal was to develop an operations concept that was affordable and that minimized safety and costs risks.

The Operations Analysis Task products have been used in the development of PLS design guidelines and operations allocations/-requirements that reduce overall life cycle costs. To achieve the PLS operations goals of low recurring cost and rapid turnaround, application of airline-based operations philosophies was instilled wherever applicable in development of the following PLS preferred operations concept. Task 1.5 efforts have also provided early definition of ground and mission/flight operations and support requirements for each PLS concept reflecting the Government furnished vehicle configuration. We have developed a data base that substantiates each of the support estimates for PLS. This data base provides us with the means to show traceability to each defined value used in our estimates for support, failure predictions, manpower, task times, spares, and repair.

The overall scope of this task has been to drive the design from an operations standpoint. This requires the operations team to develop functional flows to identify requirements and resources (manpower and equipment) for each activity associated with ground operations, mission planning, flight operations, recovery operations, and supportability. Based on the flows, a series of trades and operations scenarios is developed to determine optimum operational characteristics. The design team approach to meeting the task objectives are illustrated in Figure 4-1. Our recommendations in developing subsystem layouts, selections, and preferred operational characteristics have been coordinated with the PLS design team. The design team has been addressing operations driven requirements in defining an optimal design approach.

Detailed manpower estimates were developed for the ground, flight and mission support tasks. These data indicate that the PLS can be supported economically with a minimum amount of support personnel. This is mainly due to the airline approach to operations and the simplicity of design associated with the PLS. The use of a standard flight to Space Station Freedom also reduces the flight/mission planning complexity associated with an orbiter operation.

In evaluating the PLS, our goal has been to design a safe, durable, low life cycle cost system. To achieve this, many new features that have been learned from past manned spacecraft and aircraft experiences and "lessons learned" from Shuttle are incorporated in the PLS design. Fast turnaround requires

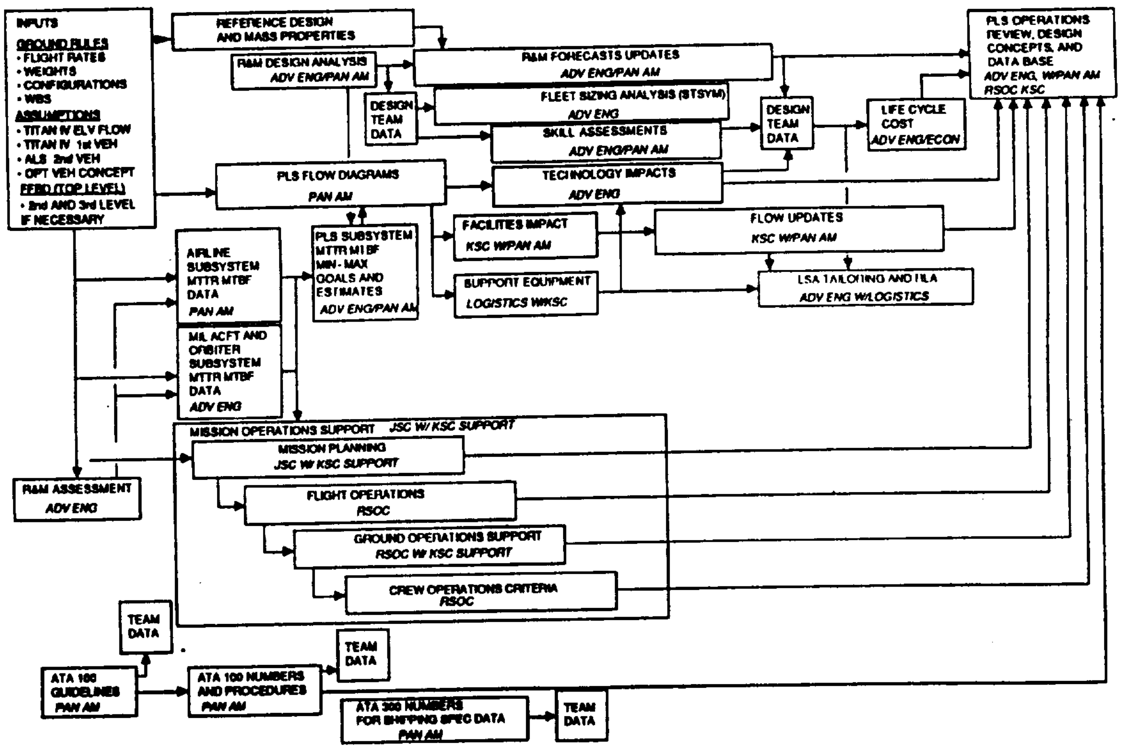


Figure 4-1. Team Approach to Design for Operations

accessibility and compliance with strong maintainability features. Our layout of subsystems has benefitted from the overall maintainability philosophy and the design architecture of the PLS.

The development of detailed staffing levels was accomplished by our on-site support teams at KSC and JSC. These data include: 1) hands on technician levels, 2) support personnel requirements, and 3) management staffing levels. We used the appropriate STS and airline estimating factors for ground operations. Since the flight and mission operations task is basically STS unique, we applied our knowledge of this factor and reduced the support requirements according to complexity of vehicle, and simplicity of mission.

The following sections provide further insight into our efforts to determine operations requirements for PLS. They provide an overview of the operating scenarios, task analyses, procedures definitions, support estimates, implementation philosophy with assumptions, staffing requirements, trade studies, and system analyses dealing with fleet sizes and attrition associated with ground operations, mission/flight operations, and supportability (logistics).

4.1 OBJECTIVES

The objective of this task has been to identify operations scenarios and support systems that enable the development of an operationally efficient Personnel Launch System (PLS). Design requirements and innovative low-cost operations and support concepts have been defined and used by the engineering design team in their selection of an optimal subsystem complement for the PLS. The principal objectives of this task were to: 1) define traceable, realistic, complete, simple, and low cost operations scenarios, 2) establish procedures and support requirements for each operational phase selected, and 3) define operational flows, logistical requirements, and support methodologies and techniques both by procedure and by subsystem. The study has provided credible data that come from existing manned-space experience and technical resources. Within the report structure, verifiable data and rationale in the selection of the operations processes are identified.

As illustrated in Figure 4-2, we have identified four specific areas that reduce support requirements for any vehicle. The selection of airframe and power plant (A&P) type personnel to perform turnaround operations at KSC has shown that we can reduce the head count by not having a large mixture of certified personnel. The A&P technician is a highly qualified airline/-aircraft mechanic that is certified to perform any maintenance that may be required on a vehicle. They are also responsible for the safety and operation of the vehicle. Our design for operations requirements for PLS have established certain constraints on the design. These are: easy access to the subsystems and equipment within each bay of the vehicle, embedded fault isolation and health monitoring, and a simplified structural inspection method.

Our analyses provided operations support data in the area of man-hour estimates from STS flows, staffing levels, flight/mission operations and scenarios, logistics spares and repair estimates (including GFE system requirements) and the facilities needed to support the PLS. These data supported the development of specific requirements for mission operations, ground refurbishment/-turnaround, preflight, launch, flight, and PLS/crew recovery. These data have also been input into a data base for future retrieval. Most of the format is either in Excel (tables and spread sheets), Draw (Briefing Charts and facility layouts), Project (timelines and schedules), and Microsoft Word (Text).

4.1.1 Functional Analyses

Functional allocations were developed to specify each operational function. This was accomplished using functional flow block diagrams (FFBD) that led to the development and definition of requirements for ground processing, launch, mission, recovery and

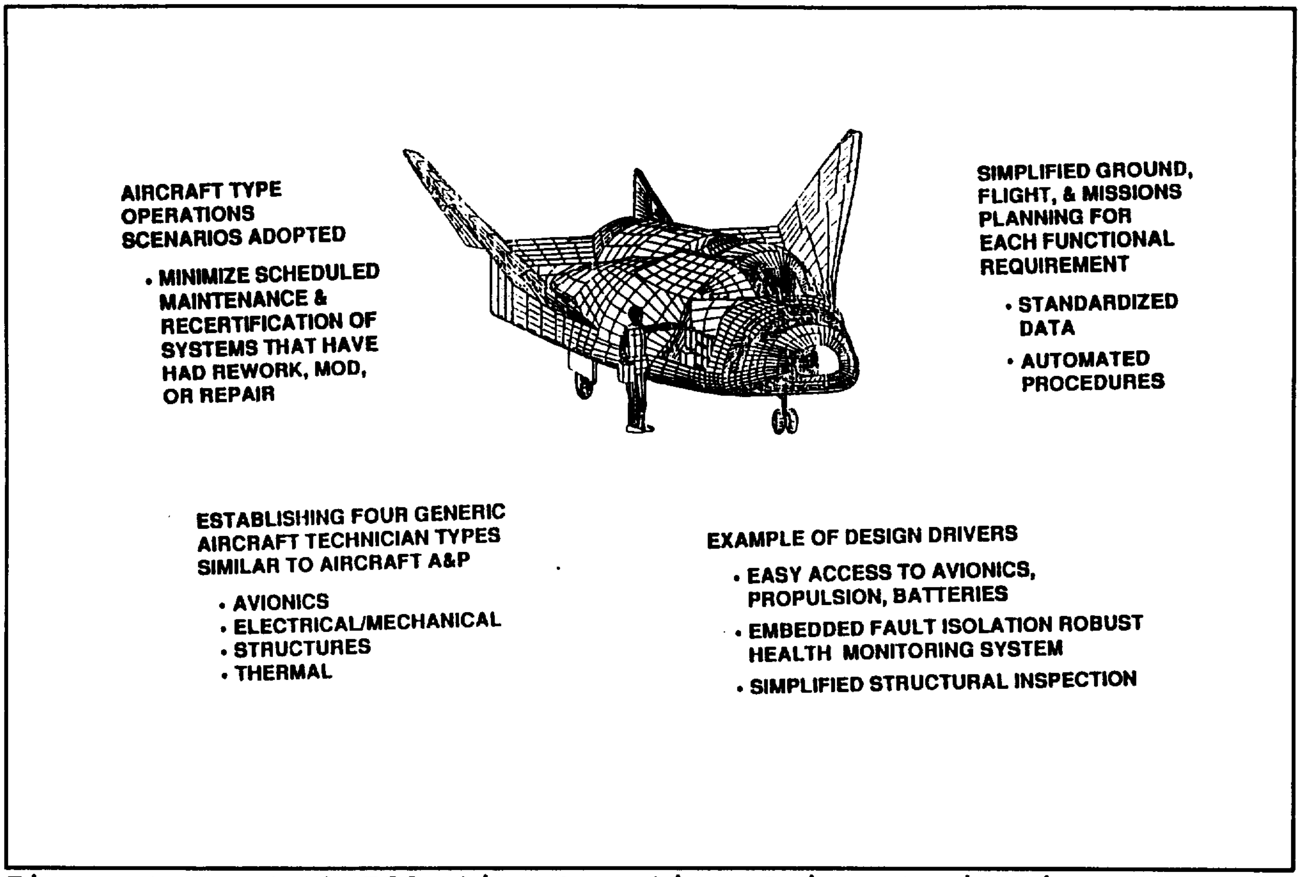


Figure 4-2. Cost Effective Operations-Driven Design is Key to PLS Program Objectives.

attrition. The FFBD's (See Section 3.0) define the steps in the operations and logistics support scenarios to a sufficient level to identify the requirements for each reference concept during each operations phase. In support of this activity each discipline listed below participated in the development of the operations data base.

Rockwell's Logistics function provided support data (spares, repair, GSE, training, publications, and support requirements) to lower functional flows and in the development of operational goals and requirements.

Rockwell's Manufacturing/Facilities function provided an assessment of facilities to support launch, recovery, servicing, and testing operations (including operational flight tests at KSC).

Pan Am World Services provided airline maintainability quantitative goals/requirements, maintenance concepts and related design ground processing guidelines. They also developed the performance trending application for PLS.

Rockwell's Space Operations Company (RSOC), in Houston Texas, provided mission and flight engineering support data, personnel levels, training simulators, and facilities data that were needed for development of the flight data files and crew activity plans. We also developed timelines using STS GFE system capabilities (modified from the existing data bases on mission and flight operations data). The data supported the development of flight and crew timeline allocations for mission operations, ground preflight and launch, flight, and recovery.

Rockwell/KSC supported the development of booster ground processing flows (using the Titan IV as a candidate), facilities impact requirements, and support system requirements (such as, launch control complex, ground support operations for fueling, etc.). They also reviewed the conceptual designs to determine ground impacts.

Specific inputs from the reference vehicle concepts, study ground rules, and the requirements established by NASA Langley Research Center (LaRC) were used during the study process. Data from Task 1.3 (Subsystems), and 1.7 (Technology Assessment) provided information of the maintenance requirements, reference subsystem, and technology assessments when determining the operations and support functions and requirements. Manufacturing requirements, and evolving technologies (i.e., items that require new/modified support concepts) were also used in development of the functional flows.

The returns on investment and a reduction in operational risk for each functional area can include reductions in costs to develop: 1) monitoring and support equipment, 2) personnel training on complex systems, 3) quantity of monitoring personnel, 4) the use of commercial off-the-shelf hardware (with off-the-shelf buffers that can be programmed), and 5) use of commercial on-board aircraft system architectures (with delta requirements for space application). This method of analysis leads to the development of standardized ground support systems, ease of fault isolation, reduction of procedures, and reduction in training)

4.1.2 Traceability of Requirements

Our current approach for developing and tracing requirements are outlined in the following example. This example defines how requirements can be traced back to the original guidelines that drive the PLS design to meet the specific mission requirements established for a selected scenario. The key factor in controlling costs, reducing support personnel and resources, and eliminating excessive design activities can be determined through a series of steps that drive lower level functions and subsequent requirements.

An example of this approach would be the PLS Guideline for Autonomous Operations. In evaluating this guideline, the first functional requirement on vehicle operations is to require that the vehicle operate without the control of ground/mission support personnel during ascent, orbital operations, descent, and landing. This means that the large cadre of personnel used on the Shuttle orbiter during a mission could be eliminated or drastically reduced (the exception would be for Range safety, flight design, crew activity planning/training, ELV monitoring, proximity operations at Space Station Freedom, ground operations for landing and turnaround. supportability to handle spares and repair inventories, training, and certification of personnel to support PLS).

Within the requirement, the ground function for processing the vehicle should be considered less costly since on-board equipment must be available to "status" the vehicle. The use on health-monitoring and a process for annunciating the status will identify more functions and requirements to be developed or levied against the guideline.

Impact on Ground Processing: From the ground processing standpoint the method to monitor and verify repairs plus determining system status or "flight readiness" can be accomplished through an automated system (Sun Work Station with appropriate buffers). If the vehicle design has a signal port (hard wire or comm) that can interrogate the PLS computer and health-monitoring networks we could eliminate the need for multiple types of equipment. This levies a requirement for providing the port as part of the PLS vehicle design:

> Requirement: The vehicle shall provide a signal port that provides data on system status and health-monitoring of all designated subsystem. (designated subsystems could be those that are considered active - electrical, pressure, temperature, electronic, mechanical, latching, actuating, quantity, leak, etc.). This assists us in the development of ground operations costs associated with an automated network to handle the requirement.

Impact on Mission operations: From the Mission/Flight support standpoint, the method to monitor the flight could use the same ports (if they were communications type). This eliminates the multiple paths that are used to down-link data and it also allows use of the same automated station to perform monitoring of the vehicle status (if required, or if an anomaly is annunciated and we need to track it.). This levies an additional requirement for the system design:

> Requirement: The vehicle shall be capable of down-linking flight data on vehicle health during ascent, orbit, descent, and landing.) This assists us in the development of mission/-flight operations costs associated with a simplified monitoring network to handle flight anomalies/ and operations (if required).

Impact on Logistics: From the logistics support standpoint, the method to perform fault isolation could use the same ports. This eliminates the multiple diagnostic paths that are used to perform test and checkout similar to the Shuttle orbiter capability. This levies an additional requirement for the systems design:

Requirement: The vehicle shall be capable of fault isolating failures using embedded diagnostic circuits that monitor input and output ports. This assists in reducing the amount of diagnostics equipment when monitoring anomalies.

4.1.3 PLS Operational Features

In brief, our design concept provides operational benefits from its relative simplicity. Table 4-1 illustrates the operational benefits of our preferred design concept. The use of electrical-mechanical actuators, and non-toxic propellants reduces the incidence of maintenance risk for complex systems similar to orbiter. We have also developed requirements that meet the autonomy guidelines set forth in the PLS contract. Our "design for operations" philosophy leads us to an operations policy that reduces manpower requirements for ground, flight, mission, and logistics support. We have also been active in supporting the safety requirements to assure safe operations and robust flight capabilities that reduce the complexities associated with a manned space vehicle.

Table 4-1. Features and Benefits of PLS Operations Concept.

Design Features	Operations Benefit
Small/Simple	Fewer parts count Design & Maintenance
Electrical-mechanical Actuators	No Hydraulics Reduced Maintenance
Non-toxic Propellants	Parallel Launch Operations Reduced Hazards
Built-in-test & Health Monitoring	Schedule Maintenance
Accessibility	Parallel Maintenance Operations Less Time Required to Process
Small, Certified Turnaround Crew	Economic Utilization of Skills
Design for Operations	Reduced Logistics Costs
Robust System Design	Flexible Operations at Minimum Risk
Standardized Missions & Procedures	Reduced Flight Planning Simple Flight Operations
Safe Abort Modes	Crew Survivability for all Credible Situations

4.1.4 Operating Scenarios

Operating scenarios were developed for the basic NASA-defined system concepts and study ground rules. Lower level functional flows were then developed for each operation. The resulting requirements were then identified and allocated to each function, from either top-level system requirements or self-imposed requirements to meet resource/system/personnel needs, consistent with meeting the operational objectives. These data were built on STS, ELV, launch/support/mission operations, logistics, and airline experience. The data were used to develop system/operational requirements, operations scenarios, and documentation associated with operations management, logistics, maintainability, reliability, training, facilities, transportation, and services. The data also support the development of manpower/resource life cycle cost estimates as the costing analysis required.

The first efforts associated with the development of our operations concept was to develop a series of functional flow block diagrams (FFBD's) that would capture the operational functions associated with the PLS. The addition of the DDT&E blocks associated with "capabilities development" and operational flight test (OFT) verification provided the important links to the pre-production and operational periods that are necessary ingredients in our "design for operations" philosophy. In order to simplify the diagram, the management and support blocks do not have input and output lines since they support every facet of the program. As

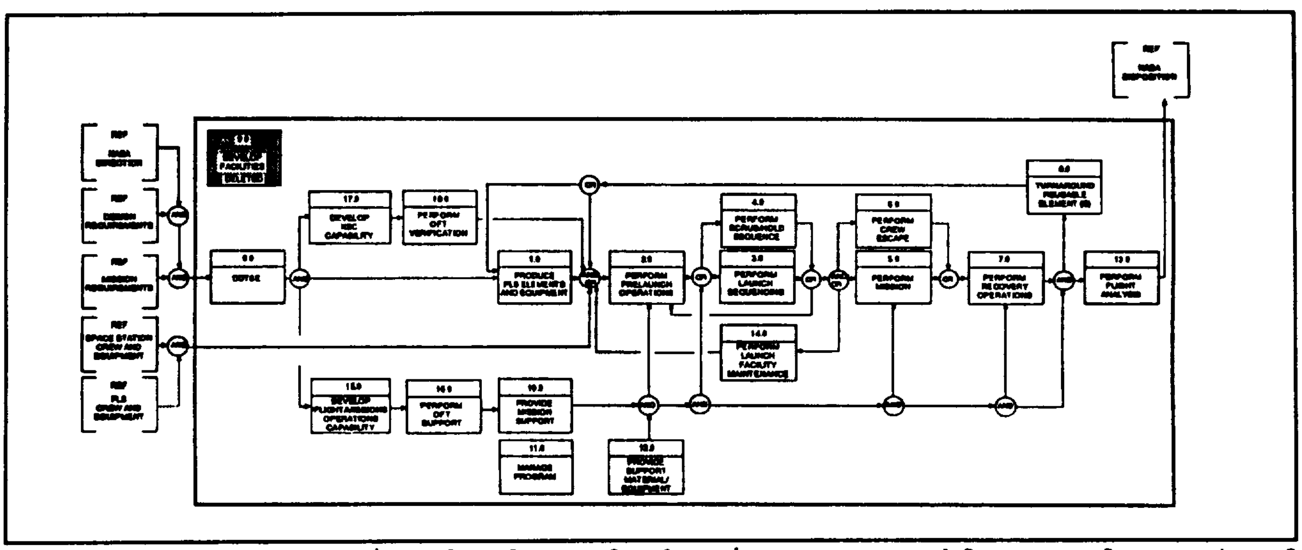


Figure 4-3. Functional Flow Block Diagrams Enable Development of Operations Scenarios.

described below, Figure 4-3 illustrates the final FFBD that has been developed for the study at this time.

The DDT&E block represents all the activities associated with planning a Phase C/D program. It also includes the flow through of a DDT&E flight test article for operational verification and the evaluation of the ground and orbital flight capabilities.

The "operations capabilities development" analyses were very important to the initial study since they drove the production, test, operations, logistics facilities, resource acquisitions, and documentation processes. Early identification of the operations facility requirements assisted us in defining facility cost acquisitions and determining life cycle costs to support each area.

The management function includes the support of all scheduling and program activities that impact each area within the flow. Cost and schedule reporting, action item status, and program direction is contained within the block.

The mission support block includes the flight design, crew planning, and ground mission support (prelaunch and flight) activities during launch. These flows were used to derive detailed requirements for the DDT&E and operational phases.

The support material/equipment block covers all logistics support activities including the logistics support analysis (LSA) tasks that are required to determine the logistical and operations support criteria for PLS and includes spares, repairs, and logistics management. The remainder of the chart is a typical functional flow that follows space system launch processing, flight operations, turnaround, and post flight analyses.

One of the driving factors for operations is the identification of support requirements based on a set of estimates from the Reliability and Maintainability (R/M) study effort. A short description of the analyses is provided in the following paragraphs for continuity.

4.2 GROUND OPERATIONS

This study requires an understanding of the complex relationships that the PLS ground processor has to a number of supporting contractors, and NASA overseeing organizations. Figure 4-4 illustrates this interface and the multiple support paths that are required. An ideal arrangement from the PLS launch site perspective would be a flight ready booster provided on a commercially contracted basis. The base operations services and limited Shuttle support provided to the PLS would both be shared with other launch programs. After the initial OFT period of pre-operational testing and training the PLS manufacturer could interface with the ground processing contract through an on-site launch site services office. The relative size of the NASA program office and the KSC and JSC staffs should be much reduced from what is required for Shuttle.

The ground operations activity was conducted by two teams:

1) Pan Am/KSC doing the airline type flow processing analyses, estimates of manpower (technicians, staff support, management), timelines, support equipment analysis, facilities analysis and trades, and

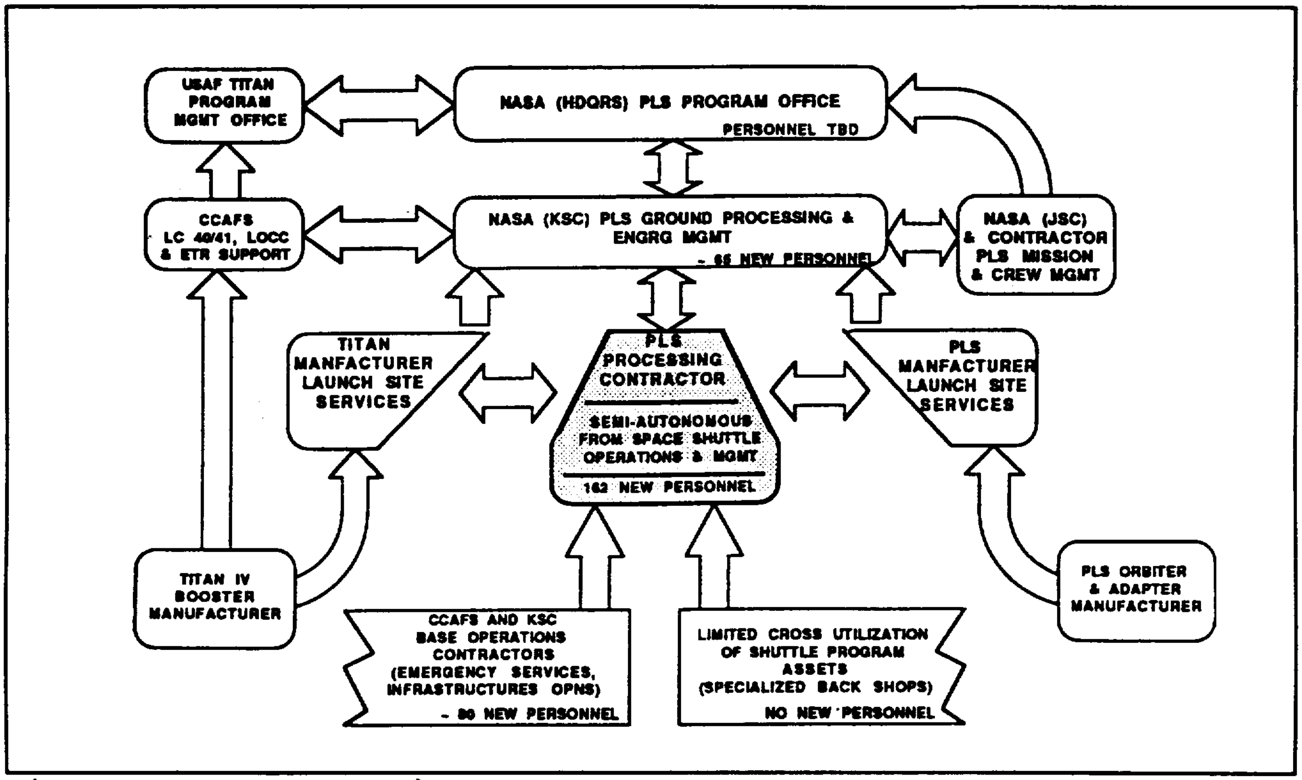


Figure 4-4. Launch Site Management Concept.

2) Rockwell Space Systems Division, Cocoa Beach Florida, doing Titan IV analyses, determining search and rescue (SAR) requirements, and determining Launch Control Center impacts in the area of facilities and software impacts. Detailed ground operations data that define each of the activities conducted by the ground operations teams are presented in Reference 4-1.

One of the key factors in our processing approach is to utilize the airline approach to ground processing. The certification concept of the Shuttle is to re-certify all systems prior to re-flight. Our concept is to certify the PLS system during Operational Flight Testing (OFT) and to then use the airline approach to maintain the vehicle in a flight-worthy status. This process verifies only those systems that have had routine and non-routine maintenance performed on them. Basically this means that the PLS is "released for service" after maintenance and not subjected to a full certification process again.

Air Transport Association (ATA) specifications should be adopted and used for technical documentation. This allows a common standard system identification process for drawings, manuals, work documents and other technical data. The PLS manufacturer provides maintenance, overhaul, repair, parts, and nondestructive inspection (NDI) manuals in accordance with ATA Specification 100 (Reference 4-2).

Maintenance and inspection requirements should be in accordance with a maintenance specification controlled at the launch site with NASA concurrence. This document will constantly change based on actual operating experience. Trend analysis would play a major role in these adjustments. Computer based work instructions consisting of work cards support by detailed, accurate, complete manuals. Emphasis is placed on using highly trained and experienced A&P technicians. This improves quality and allows more responsibility and accountability at the source of the work.

The operations scenario is illustrated in Figure 4-5. This scenario shows one of several options we have developed to enhance ground support at the launch site.

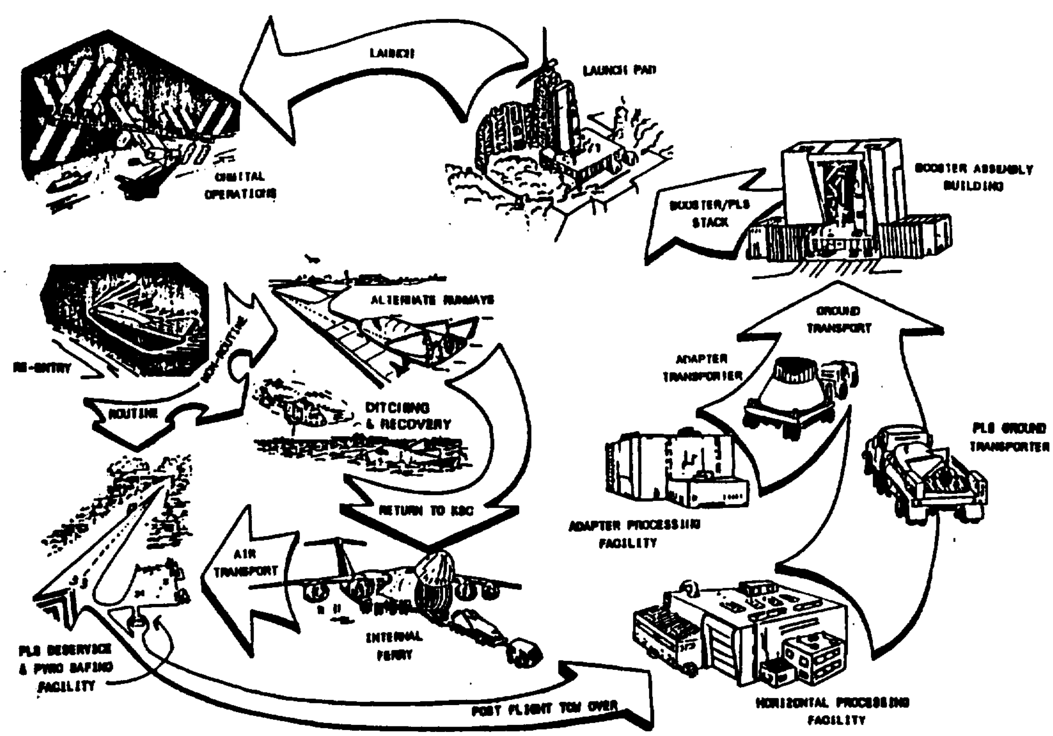


Figure 4-5. PLS Ground Operations Scenarios.

A PLS Deservice and Pyro Safing Facility (DP&SF) has been identified at the landing area to avoid introducing unnecessary hazards into the Horizontal Processing Facility (HPF). The hazards include pyrotechnic devices on the landing gear doors and a parachute system that must be safed. The HPF, including support shops and office areas are the focal point for all scheduled and unscheduled maintenance. The PLS is moved on a transporter to the launch area where it is integrated with the PLS adapter and booster (Titan IV used for this analysis). The Adapter Processing Facility (APF) is located in a "safe" area to preclude the shutdown of Shuttle operations and other projects during hazardous processing activity.

Abort and contingency landing variables include the options for the PLS to land at any airport (after reaching appropriate velocities and altitudes) or aborting at sea (during early flight period). Ferrying PLS between contingency landing sites and the KSC launch site can be accomplished by a C-5 or C-17 transport aircraft (using current NASA guidelines for mold-line). No landing site convoy is required by PLS due to its simplicity and the use of flight crew personnel to safe and monitor the PLS after landing, whereas the Shuttle orbiter requires an extensive support team and support equipment.

A thirty-one day processing flow (forty-three calendar days) has been defined for PLS. This minimizes the level of technical personnel to handle rapid turnarounds. Eight flights per year with three vehicles would not require special teams to support refurbishment of PLS. These data were developed using PLS design characteristics, R/M maintenance estimates, and Shuttle experience in determining manpower and resource requirements. Figure 4-6 illustrates the flow used in developing the manpower and resource requirements.

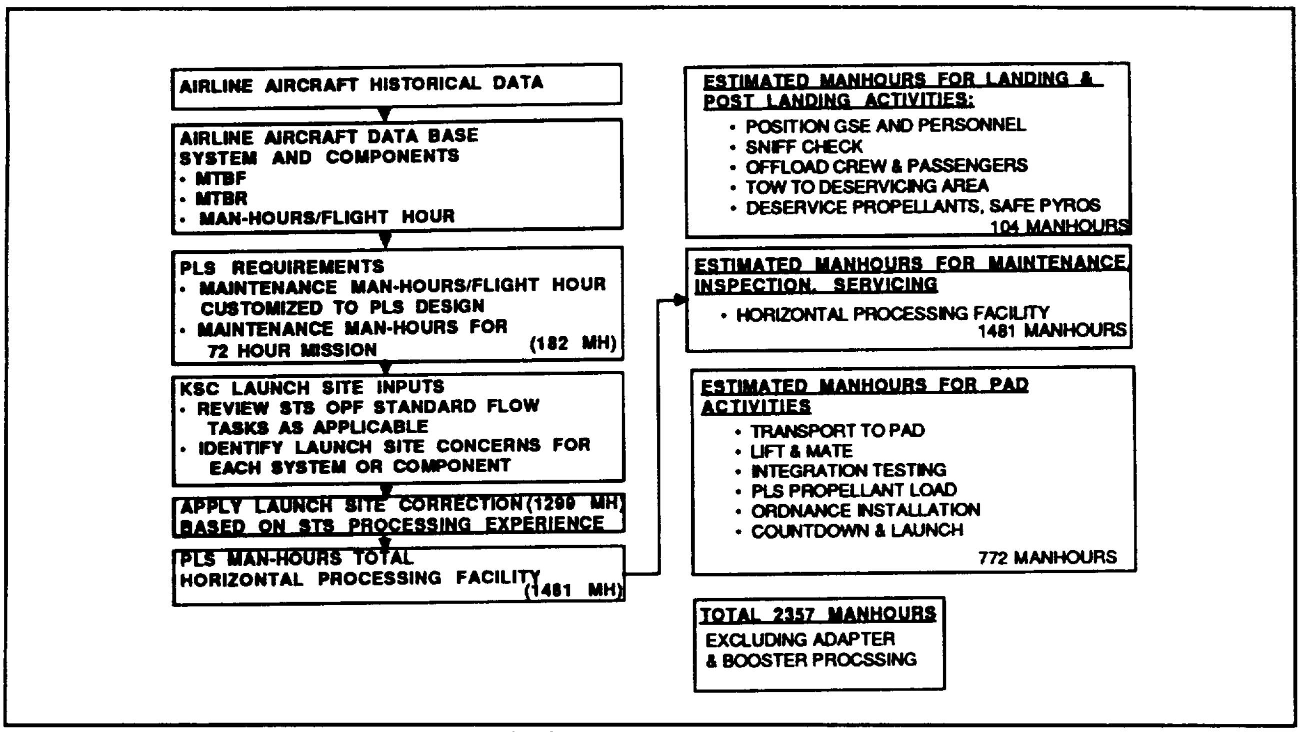


Figure 4-6. PLS A&P Technician Man-Hour Development.

Technician man-hour estimates were developed using a series of analyses that identified the specific tasks associated with PLS and relating them generically to Shuttle type processing. The estimated maintenance, inspection, and servicing man-hours for the HPF were developed using an actual historical airline/aircraft data base (747 systems, subsystems, and related hardware maintenance estimates with a factor for space system reliability added). The airline data base was used to develop requirements for each component or line-replaceable-unit (LRU) within the PLS systems based on history of similar components. Other requirements (space application only) were identified by considering STS orbiter standard flow tasks similar to PLS. Figure 4-7 is a summary compilation of the man-hours identified for hands-on processing of the PLS. Total hands-on personnel, staff support, management, and office personnel requirements have been estimated at 162 for PLS. Further detail of how these quantities were identified are described below.

ATA NO.	ATA SYSTEM TITLE	AIRLINE DATA BASE M/H PER FLT HR	M/H PER 72 HRS 1	LAUNCH SITE ADJUSTMENT 2	ASSUMED FOR PLS 1 + 2	PLS USING SHUTTLE METHODS
21	AIR CONDITIONING	0.4126	29.71	193	223	546
23	COMMUNICATIONS	0.2508	18.06	24	42	56
24	ELECTRICAL POWER	0.0806	5.80	101	107	159
25	EQUIPMENT AND FURN.	0.1326	9.55	56	66	270
26	FIRE PROTECTION	0.0476	3.43	0	3	41
27	FLIGHT CONTROLS	0.1752	12.61	22	35	257
28	FUEL SYSTEMS	0.0882	6.35	104	110	129
31	INSTRUMENTS	0.0034	0.24	20	20	50
32	LANDING GEAR	0.2024	14.57	178	193	230
33	LIGHTS	0.0305	2.20	5	7	17
34	FLIGHT DATA/MGT	0.2147	15.46	0	15	162
35	OXYGEN	0.0328	2.36	16	18	54
38	WATER/WASTE	0.0601	4.33	8	12	54
52	DOORS	0.0432	3.11	32	35	299
53	FUSELAGE	0.0278	2.00	64	66	409
55	STABLIZER	0.0050	0.36	16	16	84
56	WINDOWS	0.0045	0.32	64	64	168
57	WINGS	0.0290	2.09	32	34	226
58	THERMAL CONTROL	0.4668	33.61	232	266	1650
72	ENGINES	0.2216	15.96	132	148	195
	TOTAL		182.12	1299	1481	5056.00

NOTES:
1 REFLECTS PROCESSING IN HORIZONTAL PROCESSING FACILITY ONLY
2 STS ORBITER OPF PROCESSING REQUIRED APPROXIMATELY 39,000 MANHOURS FOR STS-31
3 PLS REQUIREMENTS ARE FOR A TYPICAL 72 HOUR MISSION

Figure 4-7. PLS Maintenance Requirements Development.

Facilities identification, sizing and usage information has been developed and is included in Section 4.2.4. These facilities represent a resilient capability at KSC. More than four vehicles can be processed at the Horizontal Processing Facility if the need should arise. The De-servicing and Pyro Safing Facility is in use less than two days per flight and no major issue has been identified to preclude meeting launch and landing requirements. The launch pad becomes the major driver and programmatic issue for meeting launch rates that have been identified in the PLS guidelines. It is understood that the Titan IV launch complex can not presently handle the PLS flights as identified in the manifest. However, our analyses have identified launch pad requirements for any ELV. Our main concern was to identify resources needed to process and provide safe operation for the PLS vehicle and crew.

4.2.1 Task Analysis

The following paragraphs further define the processing methods and discusses the development of flow timelines, resource identification, facilities utilization, and personnel assessments.

A set of timelines and operations flows were developed for PLS. Some of the key results were: 1) ground processing flows indicate the ground processing team can meet launch demands with a small cadre of support personnel, and 2) estimates indicate PLS can expect a tenth less maintenance than the orbiter. The R/M estimate was developed from the current Shuttle orbiter in-flight anomaly data base and our PLS estimates from the MAtrix model using modified aircraft and space related data). The low amount of projected maintenance is mainly due to the PLS being smaller, lighter, and less complex.

The PLS maintenance requirements were developed using representative Boeing 747 maintenance and resource data. We also developed PLS system and line-replaceable-unit (LRU) maintenance requirements. These requirements formed a basis of both PLS design specifications and ground processing maintenance requirements. These requirements, developed by Pan AM from the 747 data base in terms of man-hours per flight hour, were converted to man-hours for a typical 72-hour PLS mission.

The launch site team reviewed the requirements from the airline data base and added man-hour adjustments based on Shuttle processing experience and the unique requirements of space flight systems. The sum of the requirements from the airline data base and the launch site adjustments from the expected PLS maintenance requirements for a typical 72-hour mission. This represents maintenance, inspection and servicing done in the horizontal processing facility.

The PLS uses an entirely different and simplified approach to ground operations. Experience gained from manned space operations are combined with airline concept operations to provide an optimal approach. By using on-board and ground automated systems, plus the use of state-of-the-art software and hardware designs, the PLS can be processed with a minimal amount of personnel and resources. Our approach is not to decertify the PLS after each flight but to perform processing similar to the airlines where they require re-certification only of repaired systems. The use of airframe and power (A&P) mechanics for performing PLS processing can reduce overall costs. This is accomplished by maintaining a small work force of highly trained personnel to perform all tasks on the PLS (maximizing the use of cross trained personnel to perform multiple tasks). On the other hand, large quantity of specialist personnel are required to qualify for Shuttle maintenance and testing. Table 4-2 illustrates the simplified ground operations process that reduce overall operations costs associated with this effort.

Table 4-2. Ground Operations Approach.

PLS PROCESS	TYPES OF FUNCTIONS
Airline Maintenance Procedures	Maintenance Manuals ATA Specifications Work Cards AIRINC Specifications Zone Control NDI Manuals Repair Manuals Overhaul Manuals
Airline Maintenance Program	In-service Limits & Tolerances Full Use of Maintainability Departure from OMRS Concept
Automated On-Board Checkout	On-board Health Monitors Automated Test Equipment Local Testing (not Firing Room)
Highly Trained A&P Technicians (Reduced Staffing)	Cross Trained Four Basic Skills Cross Utilization Use of Automation
Local Controlled Maintenance Program	Full Utilization of Trend Analysis Full Utilization of Operating Histories and Anomalies

Based on the these data, we have developed the processing times by A&P technician, with an expansion to cover landing and launch activities. Table 4-3 identifies these processing times required for a PLS turnaround. The 31-day period represents the active time for accomplishment of work. The actual (elapsed) time is 43 calendar days. We have identified special areas that the A&P personnel will support beyond the "normal A&P activities" that they can perform. These are avionics, electrical, thermal protection, and mechanical system. Each person could handle the special areas; however, in order to maintain higher proficiency, we have designated these areas of expertise. Detailed data used in the development of these data can be found in Reference 4-1.

Figure 4-8 identifies the typical booster interfaces that must be understood for any vehicle that supports PLS operations. We have chosen the Titan IV as the candidate because of the known interfaces provided for the Centaur upper stage. The illustration identifies the expected or estimated reaction times for failures which have occurred over the past 20 years or which can be anticipated as a failure mode. The PLS goal is to have a reaction time of at least 2 seconds. This requires that an on-board

Table 4-3. Thirty-One Day Turnaround Estimates of Manpower.

Tech Manhours	Day	-1	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
Avionics		0	0	6	15	15	15	15	9	0	0	0	0	0	0	0	0	0	0
Electrical		0	24	16	24	24	24	24	24	0	0	0	0	16	16	16	0	12	12
Thermal Protection		0	10	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16
Mechanical System		16	64	59	54	44	64	64	64	64	64	64	64	64	64	64	56	32	52
		16	98	97	109	99	119	119	113	80	80	80	80	96	96	96	72	60	80

QTY TECHS	Day	-1	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
Avionics		0.0	0.0	1.0	2.0	2.0	2.0	2.0	2.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Electrical		0.0	3.0	2.0	3.0	3.0	3.0	3.0	3.0	0.0	0.0	0.0	0.0	2.0	2.0	2.0	0.0	2.0	2.0
Thermal Protection		0.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0
Mechanical System		2.0	8.0	8.0	7.0	6.0	8.0	8.0	8.0	8.0	8.0	8.0	8.0	8.0	8.0	8.0	7.0	4.0	7.0
		2	13	13	14	13	15	15	15	10	10	10	10	12	12	12	9	8	11

Tech Manhours	Day	18	19	20	21	22	23	24	25	26	27	28	29	30	31	Totals
Avionics		0	12	12	16	16	16	12	0	0	0	0	8	8	16	191
Electrical		0	24	24	16	16	16	16	0	24	24	0	8	8	16	424
Thermal Protection		0	0	0	0	0	0	0	0	0	0	0	0	0	0	266
Mechanical System		40	48	48	30	30	30	30	48	0	0	0	48	48	64	1481
		40	84	84	62	62	62	58	48	24	24	0	64	64	96	2362

QTY TECHS	Day	18	19	20	21	22	23	24	25	26	27	28	29	30	31	Max =
Avionics		0.0	2.0	2.0	2.0	2.0	2.0	2.0	0.0	0.0	0.0	0.0	1.0	1.0	2.0	2
Electrical		0.0	3.0	3.0	2.0	2.0	2.0	2.0	0.0	3.0	3.0	0.0	1.0	1.0	2.0	3
Thermal Protection		0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	2
Mechanical System		5.0	6.0	6.0	4.0	4.0	4.0	4.0	6.0	0.0	0.0	0.0	6.0	6.0	8.0	8
		5	11	11	8	8	8	8	6	3	3	0	8	8	12	15

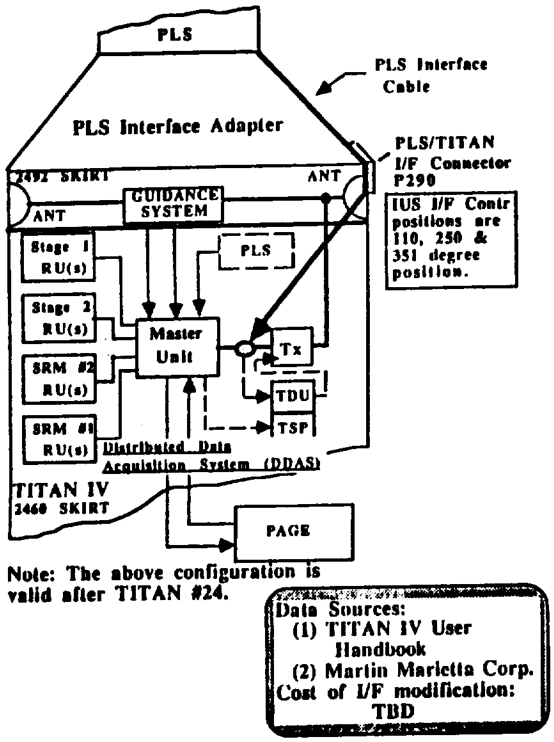
automatic system make the abort decision. For those failures which are shown to be less than 2 seconds, it may be possible to modify the booster system to detect some of the short duration anomalies.

In the event of an abort condition the crew has two methods of abort based on the type of failure. If a Titan IV type abort exists the PLS can be separated using the Launch Escape System (LES). A PLS failure could cause the crew to escape by the slide wire system or down the umbilical tower.

Simple PLS launch preparation is desired to provide the crew and passengers with the least amount of stress. The Mobile Service Tower (MST) is withdrawn from the launch pad at about T-3 hours. A minimum sized PLS ground crew can make their final configuration and operational checks after the MST is cleared. The crew and passengers are loaded and the hatch is secured at T-1 hour. The Umbilical Tower is configured for launch in the remaining time. The Solid Rocket Motor (SRM) plug is not inserted until just prior to launch time. A relatively safe environment exists until SRM ignition.

4.2.2 Procedures Definition

The procedures required to support PLS scheduled operations and maintenance (O&M) tasks are identified in Figure 4-9.



GEMINI Program: Manned capsule monitored the following booster measurements:

1. S1 & 2 Tank Pressures
2. S1 & 2 Thrust Chamber Pressures
3. Vehicle Turning Rate
4. Command Shutdown & Destruct System

TITAN MEAS. PLS ABORT ASSESSMENT

Failure Modes	Flight Phase	Meas Type	React Time
SRM Case	T+1s-2m	None	< 1 Sec
SRM INJ SRMU ACT	T-1s-58s T-58s-2m	SRM 1 & 2	2 Secs+ < 1 Sec
SRM PWR	T+1s-2m	SRM 1 & 2	2 Secs+
SRM SEP	T+1s-2m	SRM 1 & 2	2 Secs+
CORE STRUC	T+1s-8m	None(ISDS)	20-30 msec
STAGE 1 & 2 PROPELLN'T	T+2m-5m T+5m-8m	Propulsion	2 Secs+ 2 Secs+
STG 1 & 2 HYD/ACT	T+2m-5m T+5m-8m	Guidance Flt. Control	2 Secs+ 2 Secs+
CORE PWR	T+2m-8m	Electrical	2 Secs+
CORE STG 1/2 SEP FAIL	T+2m-8m	Flt. Mech. & Sequencing	2 Secs+
GN&C FAIL	T+1s-58s T+58s-5m T+5m-8m	Guidance MGC/IMU Flt. Control	2 Secs+ < 1 Sec 2 Secs+
Distrubuted Data Acquisit'n System	T+1s-8m	Instrument-ation	2 Secs+

Figure 4-8. PLS/Booster Interfaces.

Reference 4-1 provides complete definition of O&M task requirements in the context of total system requirements including contingencies. As shown in the figure, ATA system numbers and titles are used to identify inspection and maintenance procedures required to be performed in the PLS facilities. Other required procedures are to be developed in accordance with ATA 100 so that the complete set will be in conformance. We consider ATA 100 to be an existing national asset and a fundamental element in the PLS program. This concept is explained in the following.

ATA 100 was developed, and has been continuously improved, by airline operators over the past several decades. It is imposed on every new airliner and provides commonality that permits an A&P technician to navigate efficiently through operation and maintenance documentation for any vehicle. This commonality of documentation has permitted airlines to pool their resources through the ATA to focus efforts on continuing development of a single technical data specification. An even more important result of commonality, combined with universal acceptance, is that it permits contractors, subcontractors, vendors, suppliers, and operators to efficiently communicate technical data to each other. This is significant since operations cannot be efficient unless communication of technical data is also efficient. Since efficient operations are a prerequisite to airline profitability, airline operators not only voluntarily comply with the specification, they

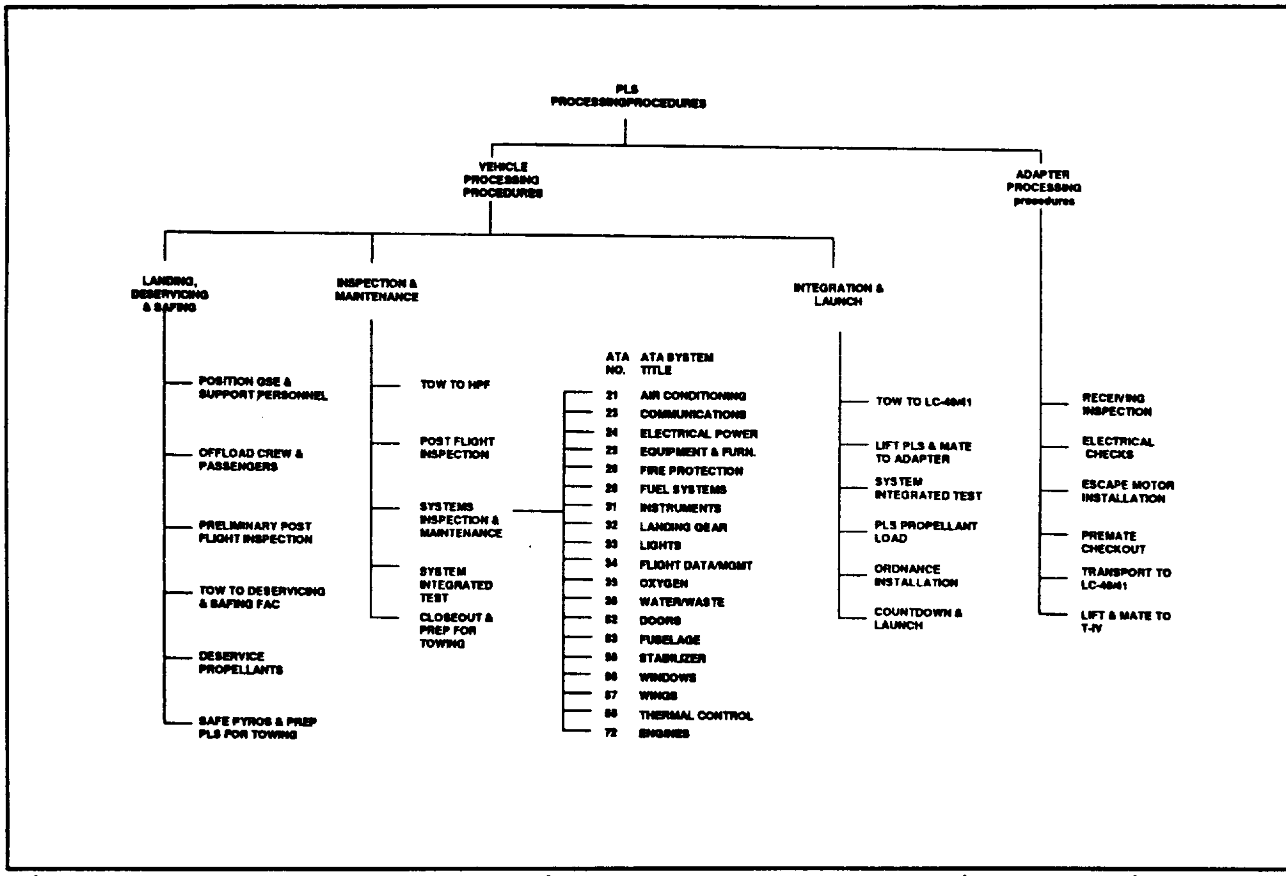


Figure 4-9. Procedures Required to Support Operations & Maintenance Tasks.

also contribute the efforts of their top maintenance professionals to support continued improvement of ATA 100. As a result, it now embodies the highest level of operations expertise in the World.

In addition to airline operators, airline accessory and component suppliers are accustomed to working with ATA 100 and know how to satisfy its requirements. As stated in the introduction to ATA 100:

> "Material supplied by accessory or component manufacturers shall cover all their equipment in accordance with pertinent sections of this specification. In addition to component maintenance manuals required by this specification, manufacturers of customer furnished equipment shall prepare documentation following the standards of this specification as applicable which will describe the equipment, tell how to install it, show how to check it for proper operation after installation, and provide procedures and troubleshooting guides for maintaining it in service."

One of the more painful lessons learned from many previous space projects is that, lacking timely imposition of understandable requirements, subcontractor and supplier documentation will be provided in a variety of formats and qualities. Some will be almost useless, or even misleading, and some will be individually excellent. However, even the individually excellent documentation will generally be unique and not readily correlatable with other documentation in the same program. This situation results in inefficient and therefore relatively large maintenance staffs because technicians must specialize in a narrow set of equipment or subsystem elements in order to remember data that should be readily available through documentation.

Proper timing of the imposition of requirements is also critically important to ensure that designers of new equipment will be responsible for provision of O & M documentation as well as design documentation. This approach will assure that O & M is appropriately considered during the design process, and that documentation is prepared up front by those most capable of preparing it. Up front preparation is even more important since the advent of CAD/CAM/CAE technology because timely imposition of requirements across the board will assure that O & M documentation for the entire system can be electronically prepared, stored, and intelligently accessed as a subset of the system data base.

<u>4.2.3 Launch abort warning time measurements.</u>

The determination of launch abort warning time raises several questions:

1. What indicators are required to provide at least a 2 second warning of launch escape?

2. How does the crew and vehicle escape system get the warning signal?

3. What is the role of health monitoring system in launch escape?

Responses to these questions are addressed below:

<u>Question 1- Required indicator.</u> The Martin Marietta Corp. Advanced Programs, Denver, Colorado, organized a telephone conversation to discuss the hazardous conditions which could develop on the TITAN IV. The telephone conversation included representatives from the following disciplines: Propellants, Avionics, Electrical, Guidance and Control. The following information was obtained:

TITAN IV Catastrophic Hazards:

Countdown Phase. The conditions at the TITAN pad were considered relatively safe and stable up to T-31.7 seconds when the transfer of power is made from ground to the TITAN on-board systems. The pad work crew actually departs the pad at about T-30 minutes. At T-20 seconds, the on-board destruction system is armed.

The real potential for a PLS abort begins at T-1 second or at SRM/Core Stage 0 ignition. The major possibilities for a catastrophic event lie in the propulsion and control systems. No danger exists until engine ignition followed by subsequent lift-off (150 milliseconds).

Lift-off/Ascent. From ignition through ascent, potential conditions which could drive an abort exists. As mentioned above under the countdown phase, the major potential for catastrophic failure lies in the propellant and control systems.

SRM. This type of failure may present a warning only in milliseconds if it is structural, or, in adequate time if the failure is a slow degradation. Within the last 20 years, TITAN IV 34D experienced one SRM case failure at T+8 seconds. The time to destruction was in milliseconds - insufficient time for a PLS escape.

Liquid. This type of failure should be detectable through instrumentation and allow adequate time for a PLS escape if conditions so warrant. A liquid failure may cause shutdown of an engine(s). This situation may not be catastrophic but may jeopardize the success of the mission. Within the last 20 years, TITAN 34D had a propellant failure characterized by a low thrust indication. There was, in fact, a large leak. The vehicle remained stable but the propellant pump ceased operation resulting in the shutdown of an engine.

Core Break-up. The TITAN IV has an inadvertent separation self-destruct system (ISDS) which senses TITAN structural break-up and starts a sequence which terminates in destruction at 20-30 milliseconds after initiation. The PLS could not escape without more warning time.

Actuators. Thrust actuators control the flight path of the vehicle as well as the loads on the structures. These actuators are commanded by guidance and driven by hydraulic systems. A hydraulic failure or mechanical failure in the actuator could result in a catastrophic event.

A vehicle at high velocity and high load pressure (MAX Q BAR) which experiences a sudden change in trajectory or initiation of a tumble could end in a catastrophic event. The PLS could not stand the stress. The warning could be in milliseconds. Hydraulic systems or mechanical failure could be the cause. If one or two actuators failed, guidance might be able to compensate. Under those conditions, the PLS may be able to escape if conditions dictated that path.

Of the 14 TITAN 34D's launched, two have experienced a catastrophic failure. One of these was a hydraulic failure resulting in a loss of control: it instantaneously diverged from the planned flight path. Failures like this may not be survivable.

Other Failure Modes:

Avionics. Avionics failures are not likely to cause a catastrophic event but should allow time for abort preparation, planning, and execution should conditions warrant - given some flight control.

Guidance, Navigation, and Control. A failure of this software/control system could cause a catastrophic event. Monitoring of the TITAN IV should provide warning but whether this is adequate for sufficient time to escape depends on the individual failure.

SRM Instrumentation Power Failure. This type of failure indicates that SRM instrumentation measurements are unusable. This may not alone indicate a catastrophic event.

Question 2- Escape warning. This action is answered by Figure 4-8, data taken from the Titan IV User Handbook (Reference 4-2), and telephone discussions with the Martin Marrietta Corp., Denver, Colorado. Figure 4-10 represents the derived data. The interface between the PLS and the TITAN is through a connector at the 2492 skirt. This is used by the Inertial Upper Stage (IUS). From this connector, cable would be routed to a hardware connection point where all TITAN measurements are obtainable.

When the crew is on board the PLS and the PLS systems are in operation, the PLS on-board computers could monitor TITAN health through telemetry obtained through the interface described above. Expert Systems can rapidly analyze conditions which are off-nominal, identifying faults and predicting degradation of TITAN systems, mission completion probability, PLS survivability and the need for abort from the period of time beginning at T-40 seconds and extending through the ascent phase. If an immediate abort becomes necessary, without human intervention, the PLS software would sound the crew alarm and execute the abort sequence. If an abort is imminent, the PLS software will sound the alarm and provide time for crew preparation before executing the abort sequence with manual override enabled.

Question 3- Role of health monitoring system. The role of the health monitoring system in the Launch Escape System is to:

- Detect out-of-nominal TITAN IV conditions.

- Analyze True Conditions - Look at all measurements which would support or refute the detected condition.

- Determine the threat of the condition to the crew, PLS, mission and command the appropriate action for the current phase of the mission profile.

- Keep the crew & ground informed via telemetry, crew instrumentation/displays, and warning indications of TITAN health and out-of-nominal conditions.

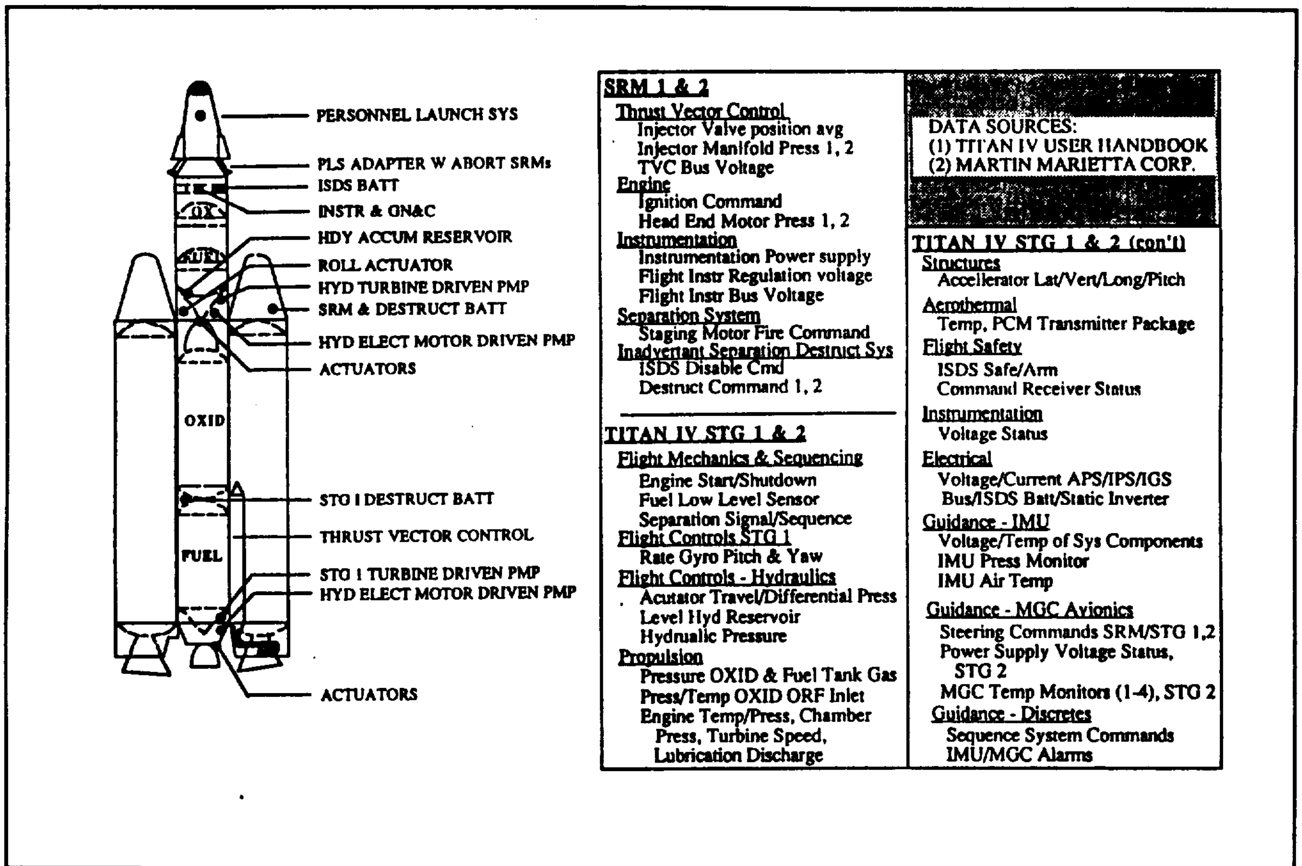


Figure 4-10. PLS Abort Assessment - Titan IV.

- Display expert system information or several levels of data strata down to an individual measurement for individual selection.

- Execute the appropriate abort sequence software if an abort is necessary.

- Track the TITAN ascent GN&C for nominal computation during lift-off to TITAN separation. Deviations would dictate a possible abort condition. A possible solution to a TITAN IV Guidance problem might be for the PLS to become a back-up for TITAN (this was used in Apollo).

The responses to the launch abort question indicate that: 1) the need to execute an abort of the PLS using the abort motors may only be for T-40 seconds through the ascent phase, 2) systems which could cause catastrophic events which might not be survivable are: structural - SRM/Core break-up; control - GN&C, control actuators, hydraulic failures, and 3) warning time varies with the type failure. The most catastrophic occur in milliseconds but these failures are rare. The NSTS lives with the similar possibilities. The PLS can monitor TITAN systems and detect other failures and make an escape, and 4) more instrumentation can be added to the TITAN.

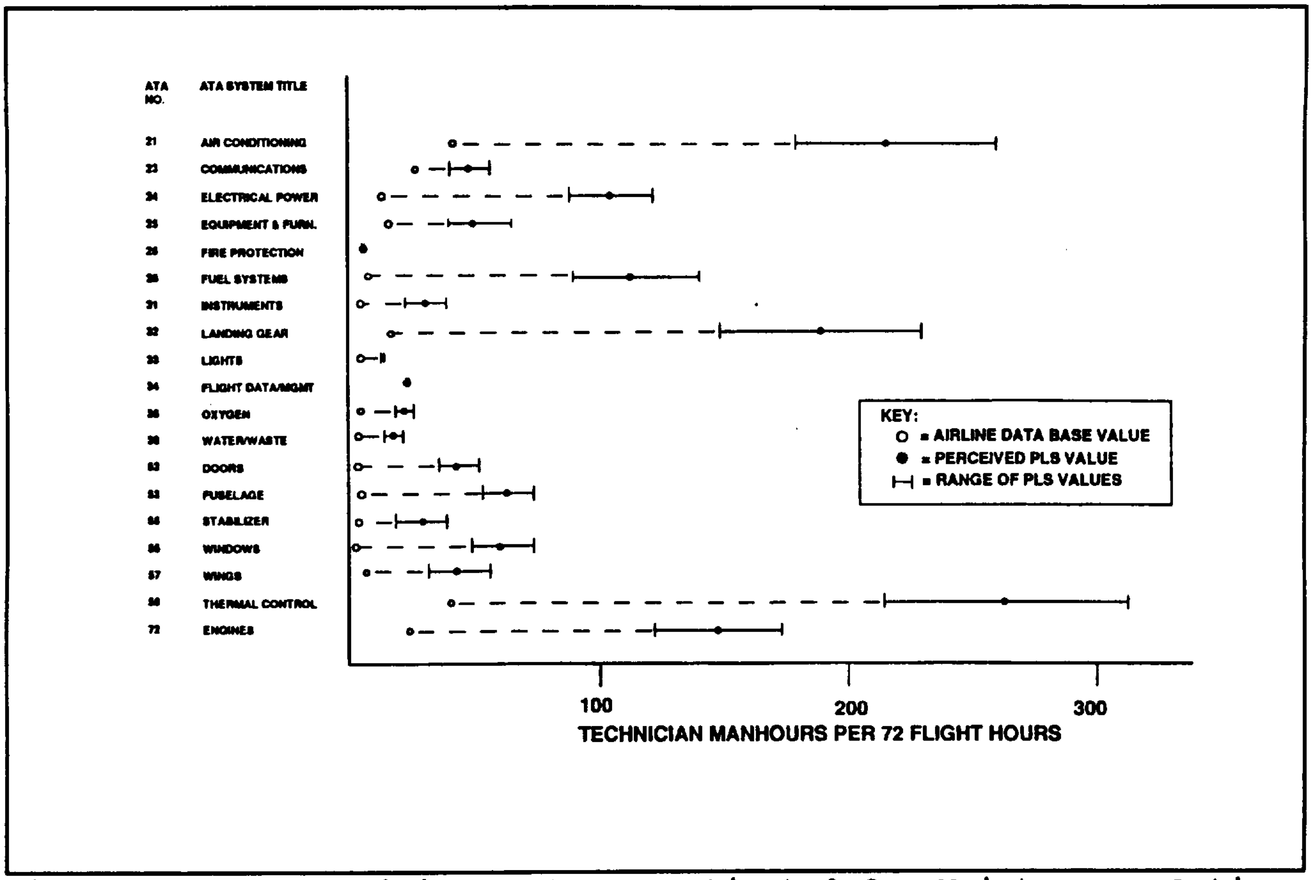


Figure 4-11. Technician Manhours Estimated for Maintenance Actions on Each System.

4.2.4 Detailed Support Estimate

Our estimate of the range of technician hours expected for maintenance actions on each system are summarized in Figure 4-11 for the ground operations and processing analysis. Following is an explanation of the analysis on which Figure 4-11 is based:

- Basic maintenance manpower requirements for each system were estimated using ATA 100 standards. These estimates were then adjusted upward to account for relatively high operational complexity of PLS due to the launch site operational environment and the uniqueness of manned spacecraft. The values derived in the estimation process are summarized in Figure 4-7.

- The range of technician-hour requirements was then estimated by applying an uncertainty factor of ± 20% to these launch site adjustments. The rationale for this method is that since the amount of the adjustment is an indication of PLS relative operational complexity, the range of processing man-hours required should be proportionately greater for those subsystems with large adjustments. This step of the process is summarized in Figure 4-12.

ATA SYSTEM NUMBER	TITLE	ATA MAN-HOURS PER 72 FLT HRS	LAUNCH SITE ADJUSTMENT MANHOURS	*+ 20 % UNCERTAINTY	*- 20 % UNCERTAINTY	PERCEIVED VALUE	MAX VALUE	MIN VALUE
21	AIR CONDITIONING	29.71	193	231.6	154.4	222.71	261.31	184.11
23	COMMUNICATIONS	18.06	24	28.8	19.2	42.06	46.86	37.26
24	ELECTRICAL POWER	5.81	101	121.2	80.8	106.81	127.01	86.61
25	EQUIPMENT & FURN.	9.55	56	67.2	44.8	65.55	76.75	54.35
26	FIRE PROTECTION	3.43	0	0	0	3.43	3.43	3.43
27	FLIGHT CONTROLS	12.61	22	26.4	17.6	34.61	39.01	30.21
28	FUEL SYSTEMS	6.35	104	124.8	83.2	110.35	131.15	89.55
31	INSTRUMENTS	0.24	20	24	16	20.24	24.24	16.24
32	LANDING GEAR	14.57	178	213.6	142.4	192.57	228.17	156.97
33	LIGHTS	2.21	5	6	4	7.21	8.21	6.21
34	FLIGHT DATA/MGMT	15.46	0	0	0	15.46	15.46	15.46
35	OXYGEN	2.36	16	19.2	12.8	18.36	21.56	15.16
38	WATER/WASTE	4.33	8	9.6	6.4	12.33	13.93	10.73
52	DOORS	3.11	32	38.4	25.6	35.11	41.51	28.71
53	FUSELAGE	2.11	64	76.8	51.2	66.11	78.91	53.31
55	STABILIZER	0.36	16	19.2	12.8	16.36	19.56	13.16
56	WINDOWS	0.32	64	76.8	51.2	64.32	77.12	51.52
57	WINGS	2.09	32	38.4	25.6	34.09	40.49	27.69
58	THERMAL CONTROL	33.61	232	278.4	185.6	265.61	312.01	219.21
72	ENGINES	15.96	132	158.4	105.6	147.96	174.36	121.56

Figure 4-12. Range of Manpower Values Expected for Maintenance Actions on Each System.

The skill level of all PLS technicians is required to be extremely high. As a minimum, each must be a licensed A & P Technician, must have completed certificated formalized A&P training at a nationally recognized institution, and must have a minimum of 5 years experience directly related to maintenance of subsystems used in the PLS. This skill level is necessary to enable efficient operations through self-certification of work and cross utilization of technicians. Thus, individual technicians will certify much of their own work across functional areas (e.g. de-servicing and safing, inspection and maintenance, and integration and launch). This approach, combined with built-in-test (BIT) capability, permits a small staff of highly qualified A&P Technicians to support a range of launch rates up to 10 or 12 flights per year using a one-in-flow, five day week, single shift approach. Figure 4-13 demonstrates this capability, and shows that above 10 or 12 flights per year manpower requirements would start to increase rapidly, and a two-in-flow or multi-shift operation would be considered for implementation.

A 31-day (43 calendar days) processing flow was developed. This flow accommodates 8 launches per year using a single eight hour shift, five days per week. The flow was developed from the maintenance manhours developed for the airline data base supplemented by Shuttle experience. Figure 4-14 identifies the overall flow concept and identifies requirements for major

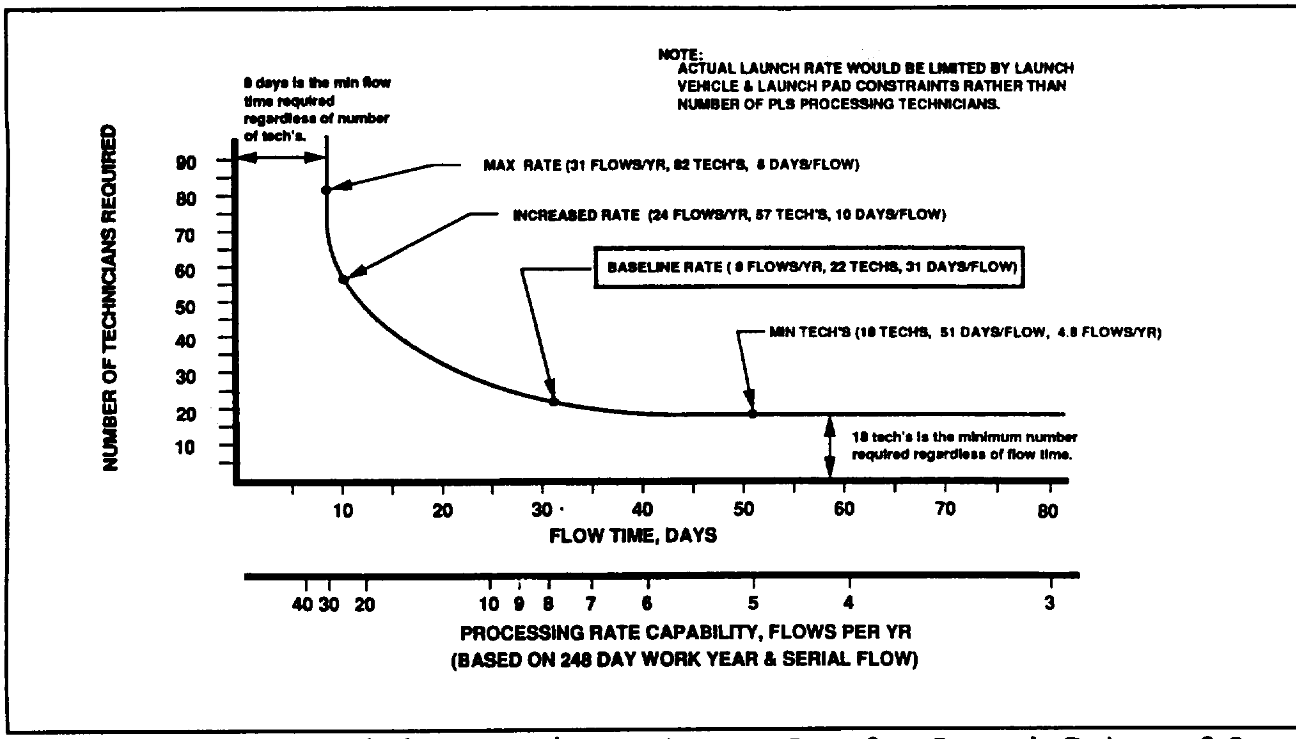


Figure 4-13. Technician Requirements are Low for Launch Rates of Less Than 10 per Year.

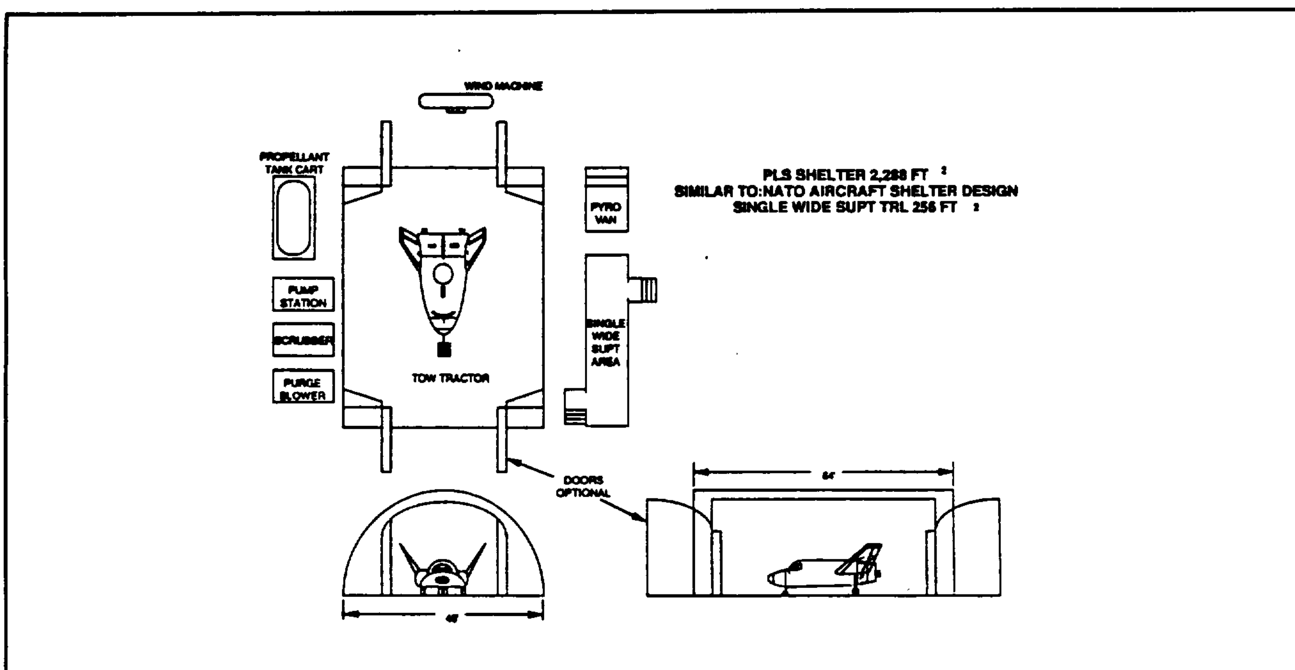


Figure 4-15. Deservice and Pyro Safing Facility.

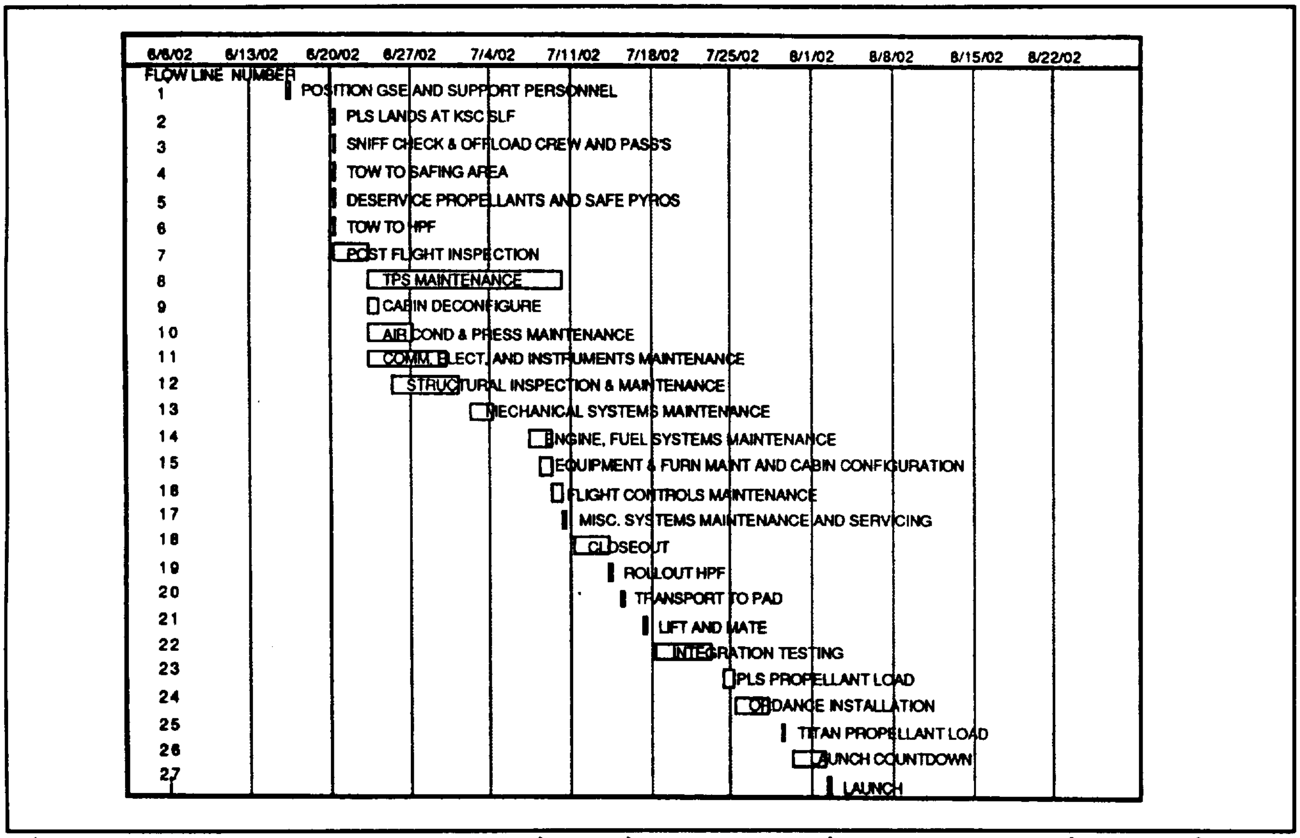


Figure 4-14. PLS Processing is Accomplished Effectively in 43 Calendar Days (One Shift Operation).

facilities and support systems. Based on this flow, three new facilities will be required:

1. PLS Deservice & Pyro Safe Facility (D&PSF) (See Figure 4-15),

2. Horizontal Processing Facility (HPF) (See Figure 4-16),

3. Adapter processing Facility (APF) (See Figure 4-17).

The facilities are further defined in Reference 4-1. In addition, significant Titan Launch Complex (LC-40/41) modifications will be required to accommodate PLS as a Titan IV payload, as indicated in Figure 4-18. Recognizing, of course, that another launch system such as ALS may be used. Reference 4-1 also provides pertinent information on modification and use of existing Titan IV facilities and operations. An additional major support system which will be required is the Emergency Egress, and Search and Rescue system.

NOTE: Although the ELV analysis was accomplished on a Titan IV vehicle, we believe that the functional interfaces to any ELV would be basically the same. The primary function within this task was to derive requirements against the ELV to assure safe efficient operations.

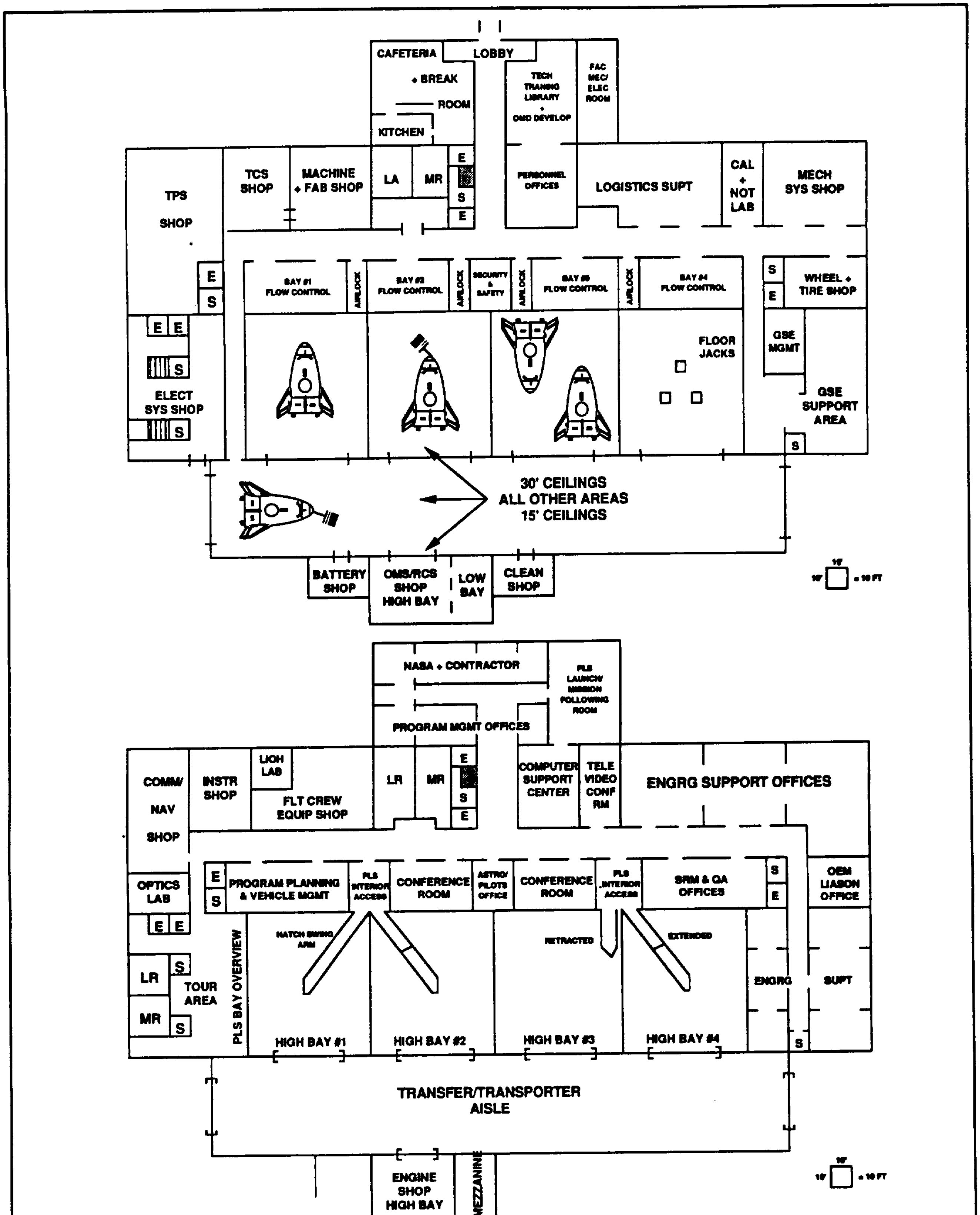


Figure 4-16. PLS Horizontal Processing Facility (Ground Floor - Upper; Second Floor - Lower)

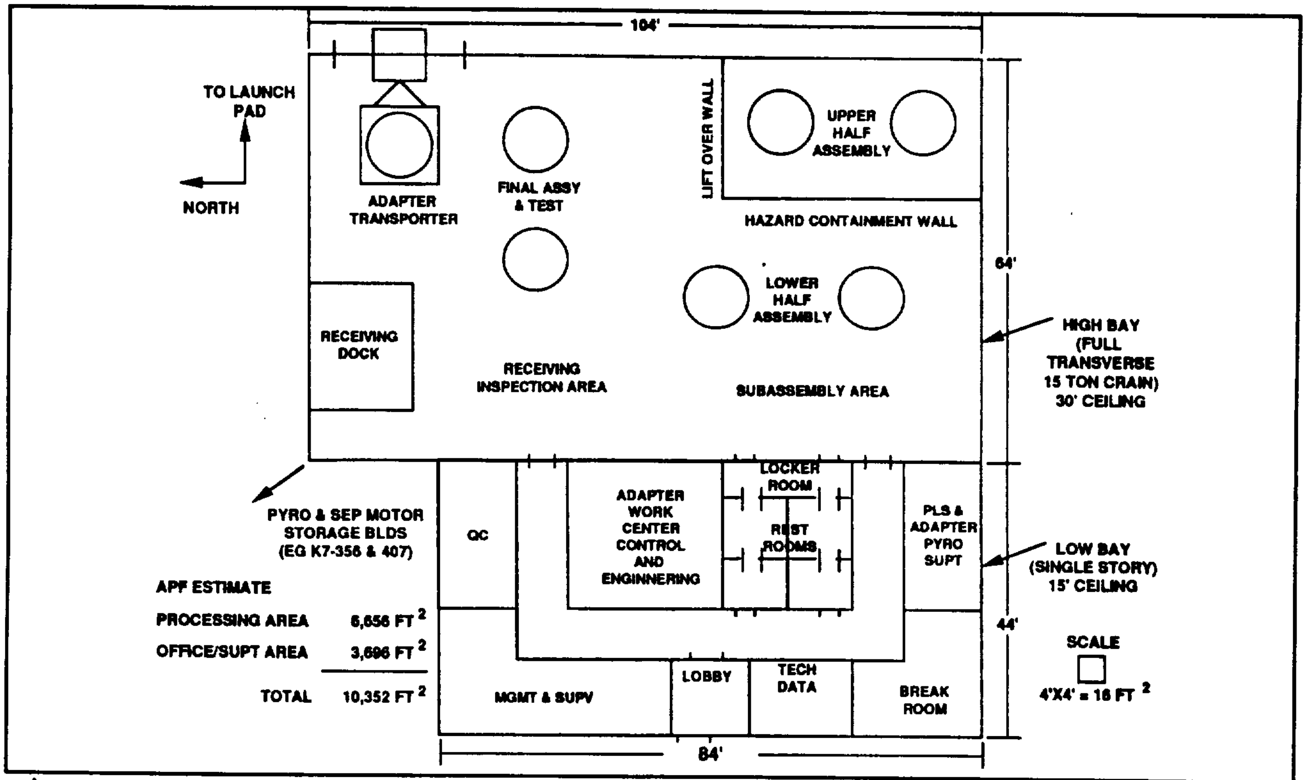


Figure 4-17. PLS Adaptor Processing Facility.

The Horizontal Processing Facility (HPF) contains four bays consistent with the baseline fleet size of four vehicles. This facility design provides flexibility to start with one bay fully outfitted for processing one-in-flow, and expanding to two bays for two-in-flow if launch rates start to exceed 10 or 12 per year. In addition, adequate capability is provided for a variety of configurations to accommodate such potential requirements as surge storage of ready vehicles, or long term contingency major maintenance, without choking ongoing critical path operational flow. It would be possible to defer some front end costs by constructing the facility in phases, starting with 2 bays and adding additional bays as needed.

The De-servicing and Safing Facility (D&PSF) provides a remote PLS de-fueling and pyrotechnic safing location. It is also a readily available shelter for SSF crew and equipment down load during inclement weather. Due to its relatively low utilization, a NATO aircraft type shelter with support space provided in a 256 sq. ft. trailer is seen as adequate for this requirement.

The Adapter Processing Facility (APF) is remotely located to provide for "clear area" processing and checkout of the adapter. PLS/booster separation motors are processed in this facility.

4.2.5 Implementation

PLS will be operated within the KSC organizational framework and will be supported by the base support infrastructure, however,

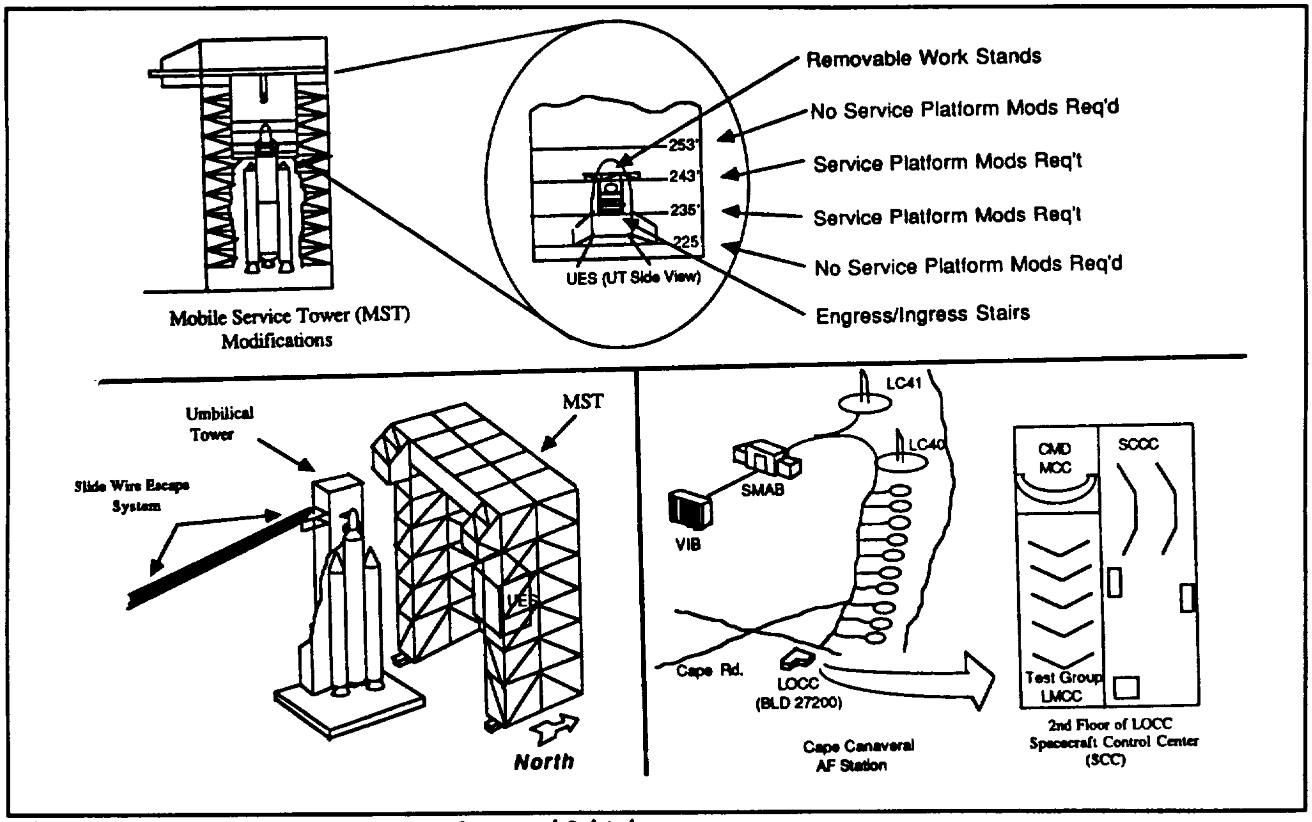


Figure 4-18. Launch Pad Facilities.

operations in the D&PSF, HPF, and APF will be completely off-line, separate from and independent of other flight programs at KSC. Figure 4-19 illustrates the launch site organizational team for PLS. It does not include the base support organization. This independence is critical to implementation of the A & P Technician concept because the concept is not fully compatible with any other space flight program at KSC. Operations in the above three dedicated facilities offers the most opportunity for dramatic improvement in efficiency through use of the A & P Technician concept. The ratios of management to technician levels are further defined below.

The PLS functional area which will not be separate and independent from other programs and which may interfere with efficient operations is the Titan IV integration and launch area at Launch Complex (LC 40/41). Figure 4-20 shows the baseline Titan IV processing flow which requires 27 weeks. PLS uses Titan IV as a reference baseline because Titan is a relatively mature system, its interfaces are well understood, and processing data are available. However, because Titan has several users competing for limited launch opportunities, scheduling problems are likely. The PLS approach to minimizing these problems is to simplify interfaces thereby minimizing Pad operations and providing flexibility to fly on alternate vehicles or from alternate launch pads. While Titan IV uses LC-40/41 as the baseline, the analyses indicates there is a variety of launch vehicle options which could

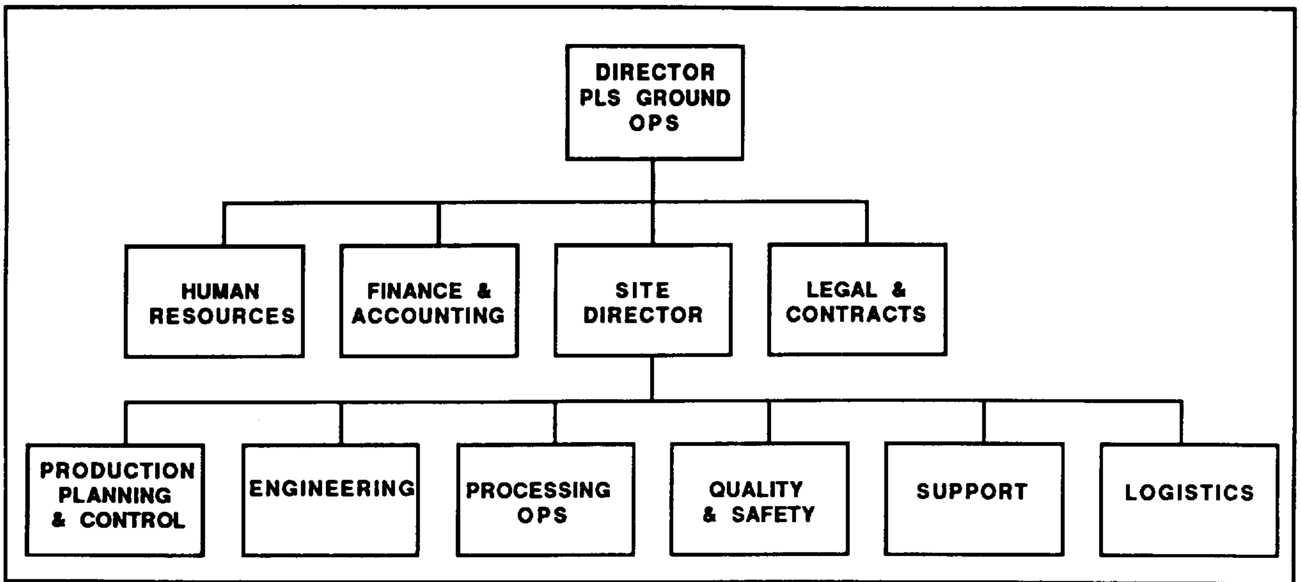


Figure 4-19. PLS Ground Operations Staffing.

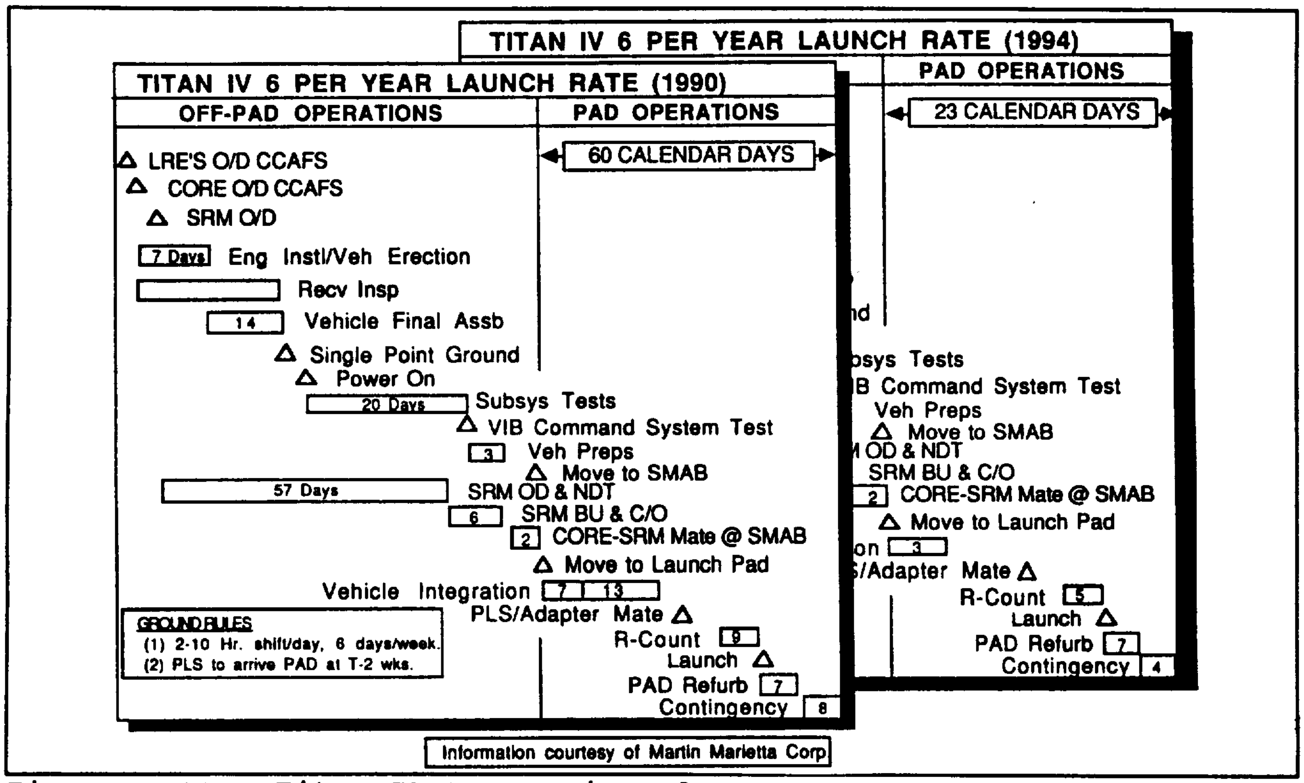


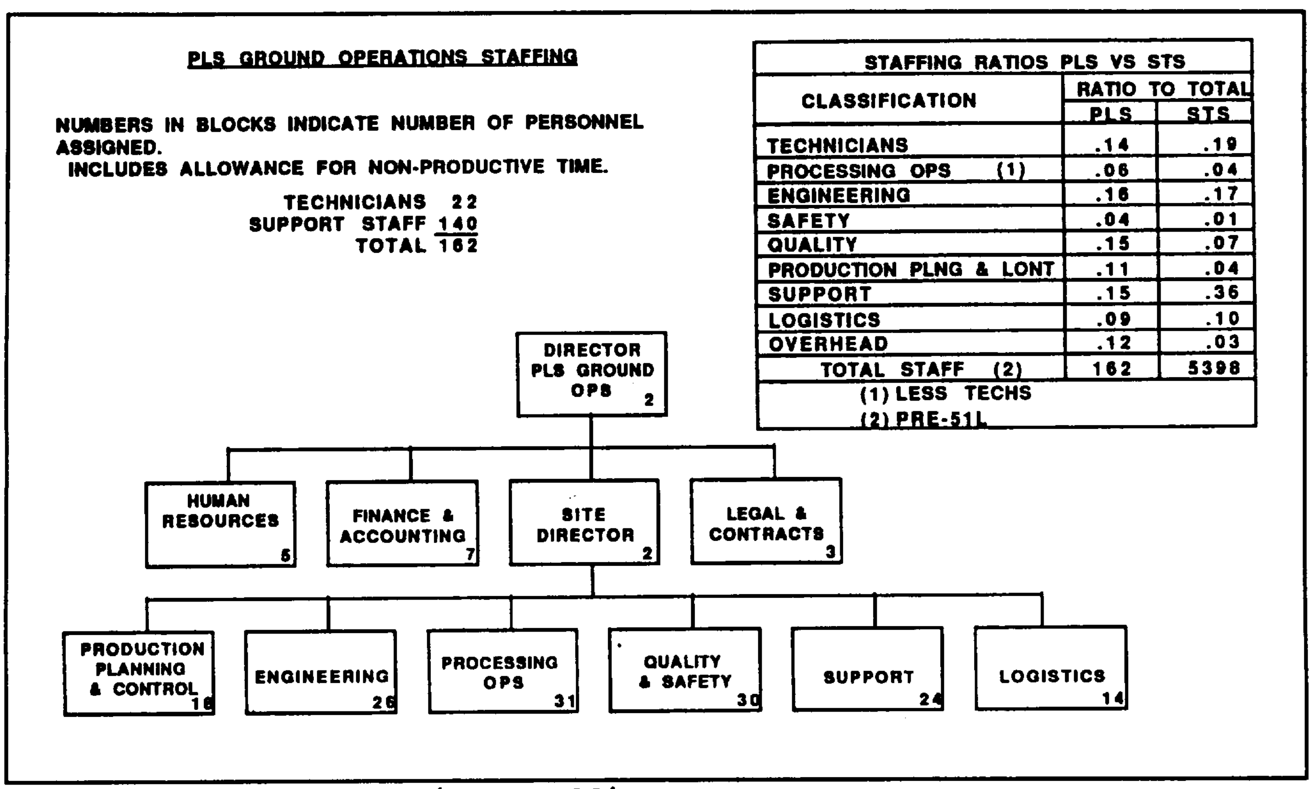
Figure 4-20. Titan IV Processing Flow.

accommodate PLS in the future. These range from Titan IV launched from LC-39, to future transportation system elements (e.g. ALS or Shuttle-derived Liquid Rocket Boosters) launched from LC-39 or a new site. The simple-interface approach projected here will enhance system flexibility to take advantage of the full range of launch vehicle options.

4.2.6 Staffing Levels

During the study, various levels of manpower were evaluated that would be required if we varied the turnaround times. Using a flight rate of 8 flights per year and varying turnaround time, we found that 31 days (direct-hands on time) was an optimum point to meet the launch schedule and minimize the manpower loading. This came from Figure 4-13 on the "knee" of the curve.

Technician staffing was developed from the definition of processing flow. The flow considered elapsed time for maintenance tasks and skills required. Matrices were developed showing manhours per skill required by processing line items and processing days. Summing the manhours per skill per day provided the number of technicians required per day for each of the four skill levels identified previously. The concepts of using four basic skills, highly trained A&P technicians and some degree of cross training provides an efficient approach to maintenance and inspection tasks. An allowance of 12 percent has been included to account for sick leave, vacation, and holidays. It is also expected that when not actually needed for processing, technicians would either be enhancing their skills though training, or would be performing minor repairs in the shops.



STAFFING RATIOS PLS VS STS		
CLASSIFICATION	RATIO TO TOTAL	
	PLS	STS
TECHNICIANS	.14	.19
PROCESSING OPS (1)	.06	.04
ENGINEERING	.16	.17
SAFETY	.04	.01
QUALITY	.15	.07
PRODUCTION PLNG & LONT	.11	.04
SUPPORT	.15	.36
LOGISTICS	.09	.10
OVERHEAD	.12	.03
TOTAL STAFF (2)	162	5398

(1) LESS TECHS
(2) PRE-51L

Figure 4-21. Launch Site Staffing.

Figure 4-21 shows the overall PLS Launch Site organization and staffing level for flight rates up to 10 or 12 per year. The core of the staff is the 22 A&P technicians required for inspection and maintenance. Other staff positions are derived from the nature of the project (e.g. Director, Ground Operations). Staff size was estimated by a bottoms up approach, and was cross checked with support staff to technician ratios of other programs such as Shuttle. The small total staff required for PLS relative to other programs such as Shuttle is a direct result of the small number (22) of technicians required. For comparison purposes, the ratio for each staff classification compared to the Shuttle is shown. The STS figures shown are pre STS-51L. Many of the ratios are quite similar. Where major differences occur, they are because of differences in function or because certain minimum personnel are needed regardless of total staffing.

4.3 FLIGHT OPERATIONS

Our operations analysis approach for mission and flight operations for PLS was to develop a standard system for preparing documentation and processes that can be used on all flights for PLS Design Reference Mission (DRM-1, SSF crew rotation; KSC to SSF and return). We have baselined our data to reflect the STS mission elements that drive costs and manpower. From this data base, we have extracted flight and mission specific tasks that relate to the PLS and developed a distinct flight and mission planning scenario. Our approach was to: 1) maximize standardization of flight systems, missions, groundrules and constraints, mission rules, procedures, and processes and products, and 2) build and utilize a knowledge base for flight crews, ground support personnel, and automated systems. The knowledge base buildup requires the following criteria to be in place for cost effective operations.

1) Flight Crew
   - Re-fly dedicated crews for "tour of duty"
   - Minimize simulator time to certification
   - Maximize auto-land capability
   - Provide refresher training for "off-tour" veterans

2) Ground Support Personnel
   - Subset of shuttle ascent/entry teams
   - Core set of mission managers (rotate flight crews)
   - On-call support during docked operations (after arrival and before departure)

3) Artificial Intelligence
   - Use vehicle information
   - Use ground system data bases
   - Stream line support requirements

We assessed current NSTS Mission Operations to determine which processes, methods, and products used by the NSTS would be necessary and applicable to the PLS. PLS flight operations program control documentation has been outlined, along with flight and mission documentation. Flight Design analysis requirements have been outlined, along with post-flight evaluation needs. This work has led to an understanding of how to develop the PLS mission operations capability. We developed preliminary staffing levels and cost estimates also as a result of this work.

4.3.1 Task Analyses

In our study of flight and mission operations planning activity, we followed the STS documentation process, as a point of departure. From this analysis process we then tailored the PLS requirements based on its mission uniqueness. The STS requires that the flight documentation process be repeated every flight due to the unique mission requirements associated with each flight. The re-configuration requirements for each payload impacts the flight software requirements and also drives the amount of flight planning documentation. Figure 4-22 illustrates the approach we used in developing our data base. As illustrated in Figure 4-23, the amount of savings that can be seen for the PLS is quite significant; this is based on the DRM-1 mission to Space Station Freedom and return. Any changes to the mission requirements would impact operational costs due to consequent changes to the re-configuration and flight requirements.

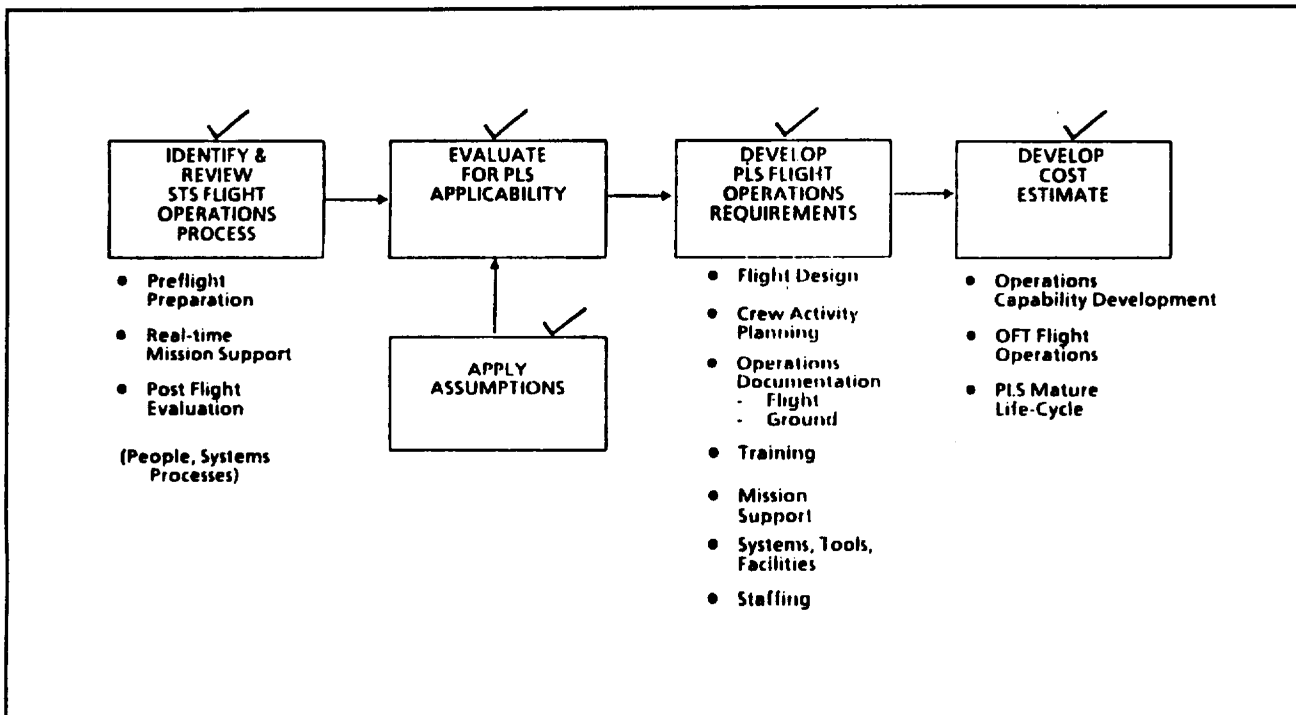
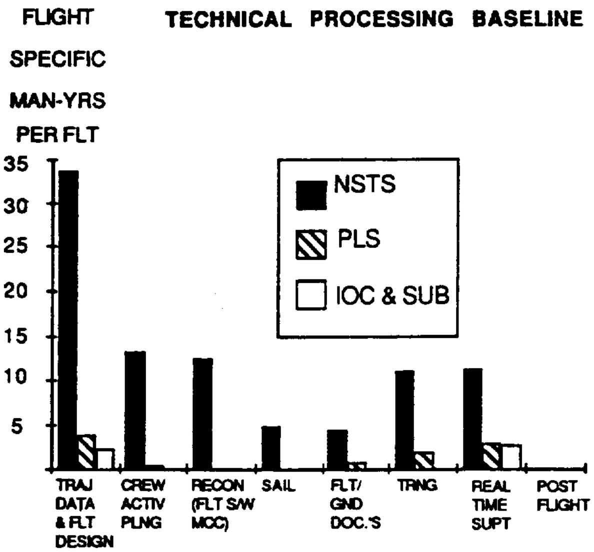


Figure 4-22. PLS Flight Operations Study Approach.

Mission and flight operations timelines were developed using the NASA JSC flight design and crew activity plan software tools. These data were used in assessing the overall requirements associated with crew planning, mission support, and flight data recording and processing. One of the key items that must be addressed in later studies is the use of existing NSTS and SSF mission support during flight for PLS. This can reduce overall sustaining costs and management systems that are necessary to maintain a full time team for PLS. The flight design trajectory



ALL UNITS IN MAN-YEARS

COMPARISON OF STS TO PLS REQUIREMENTS	STS	PLS	PLS IOC2 & SUBS
TRAJ DATA & FLT DESIGN	33.69	4.00	2.5
CREW ACTIVITY PLANNING	13.40	.47	
RECON (FLT S/W, MCC)	12.57	N/A	
SAIL	4.93	N/A	
FLT/GND DOC.'S	4.68	1.00	
TRAINING	11.13	2.00	
REAL TIME SUPPORT	11.48	3.00	2.8
POST FLIGHT	TBD		

SUMMARY PER FLIGHT	STS	PLS
FLIGHT SPECIFIC	91.88	10.47
ALL FLIGHT	154.8	7.5
NON-FLIGHT	77.4	10.47
TOTAL	324.08	28.44

Figure 4-23. NSTS Contractor vs PLS Flight Operations Support.

tape was used for the DRM-1 timeline. The first DRM-1 results indicated that a long mission first day (18.5 hours for crew) could exist if the launch vehicle inserted the PLS in a 50 by 100 NM orbit with a phase angle of 40°. Since that run, a reduction in phase angle to determine an optimum mission (approximately a 15 hour day) was evaluated. Also, in cooperation with the Martin Titan Commercial vehicle team, the latest Titan envelope was defined: it indicates that a Titan can provide a 85 by 90 NM insertion orbit and a capability for a lift weight of more than 45,000 lbs. Using these data, new timelines can be generated to assess first day/hours more accurately.

Detail analyses of each function in Figure 4-24 was accomplished to "size" the manpower requirements for PLS. It was also used to determine what requirements were needed to support PLS. The overall mission planning requirements for STS (Figure 4-25) were tailored to meet the operations flight test program, and finally to meet a fully mature PLS operations program. These data were then used to determine manpower and resources needed to meet PLS mission planning requirements. One major item to note is that the payload driven mission requirements for STS are not required for PLS. The schedule depicted in Figure 4-26 identifies the final size of a mission plan cycle of seven months.

Post flight evaluation for the NSTS is performed on each flight to evaluate actual performance vs. predicted performance to verify margins and refine analytical tools. The PLS post flight evaluation is required for OFT and then the effort can be reduced

OPERATIONS FUNCTION	OFT 1	OFT 2	OFT 3	OFT 4	PLS-1	PLS-2
5.0 MISSION SUPPORT	1022 MHRS	1022 MHRS	1022 MHRS	1022 MHRS	1022 MHRS	586 MHRS
10.1 OPERATIONS PROCESS REQUIREMENTS	6.83 MYRS	-	-	-	1.5 MYR UPDATE	UPDATE A/R
10.2 FLIGHT GROUNDRULES & CONSTRAINTS	4200 MHRS	1000 MHRS	500 MHRS	500 MHRS	4200 MHRS	UPDATE A/R
10.3 INITIALIZATION PLANS & SCHEDULES	4 MYRS	3.25 MYRS	2 MYRS	2 MYRS	3.25 MYRS	
10.4 FLIGHT DESIGN AND DATA	21.6 MYRS = 4800 HRS TRAINING	11.3 MYRS	11.3 MYRS	11.3 MYR		
10.5 CREW ACTIVITY PLAN	944 M					
10.6 MISSION PRODUCTS & DOCUMENTS						
10.7 CREW AND FLIGHT CONTROLLER TRAINING						
10.8 PRELAUNCH REAL TIME SUPPORT						

Figure 4-24. Flight and Missions Operations Resource Data Developed Through Functional Flow Analyses.

once the operational data bases are complete. The primary function for PLS post flight analysis will be to resolve anomalies and in-flight accuracies. Figure 4-27 illustrates this philosophy.

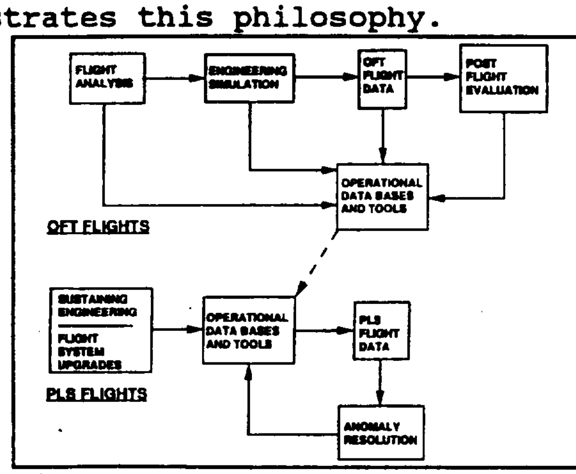


Figure 4-27. PLS Post Flight Evaluation.

4.3.2 Procedures

Procedures identified for the PLS flight and mission segment of this project were classified into three basic categories:

- Program Management and Control Procedures
- Mission Support Procedures
- Flight Support Procedures

Program management and control procedures. The initial Program Management and Control documents were identified and a development schedule prepared which started the process 30 months before the first operational flight test (OFT) flight date and called for completion at 13 months before the flight. This set of documents cascades down from the project "Baseline Operations Plan" which establishes the roles and responsibilities of the PLS flight operations team.

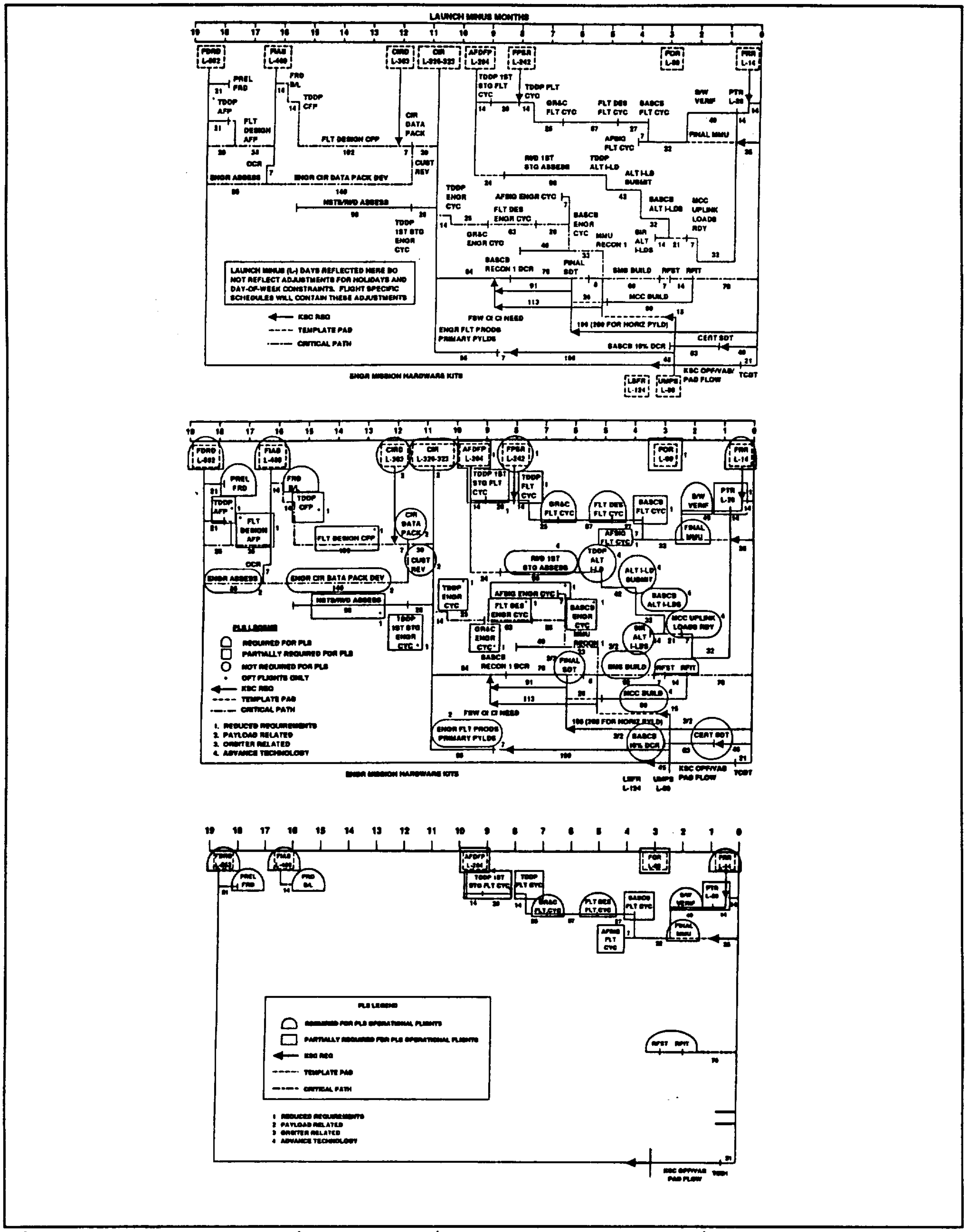


Figure 4-25. Generic STS Flight Cycle Production Templates Were Adapted to the PLS DRM Process.

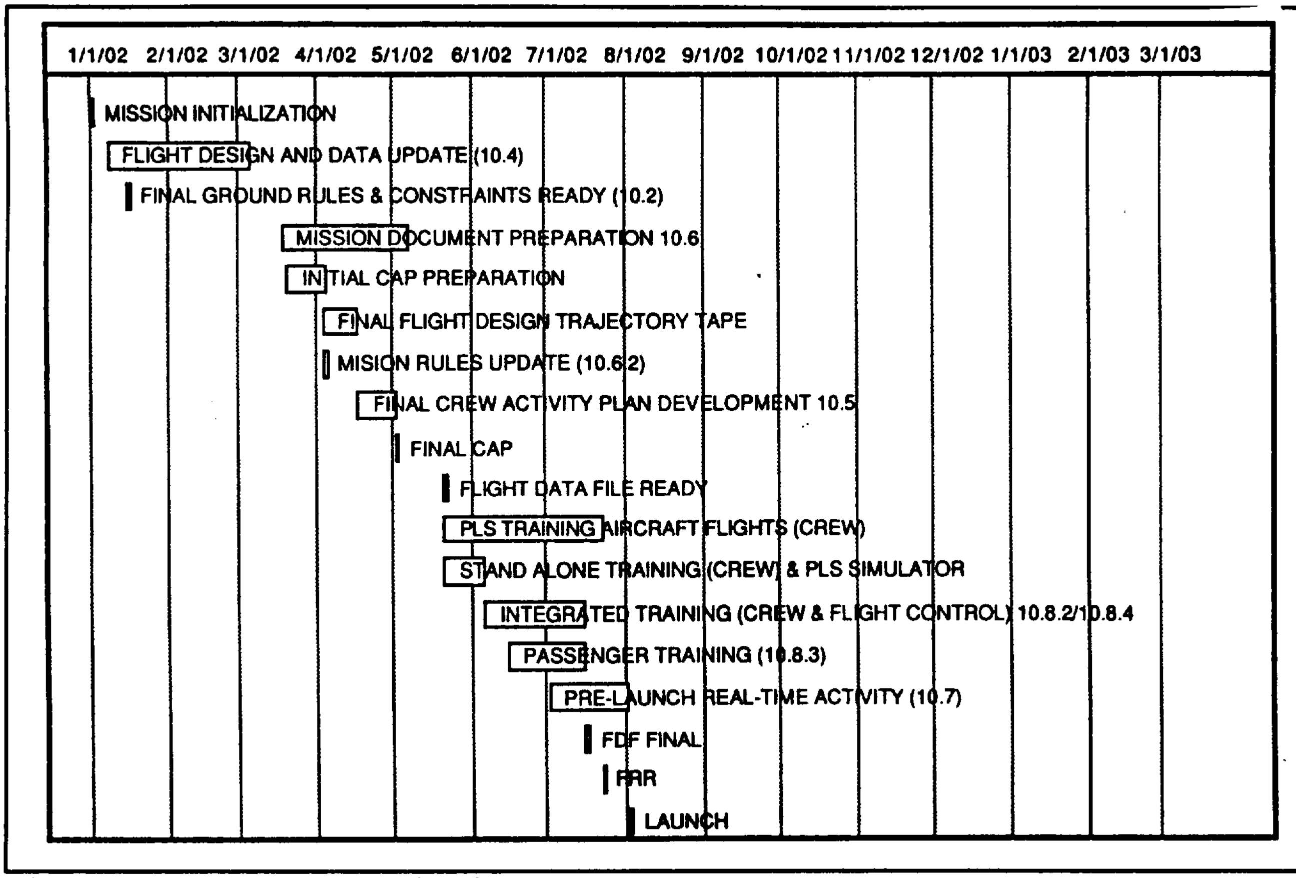


Figure 4-26. PLS-1 Mission Plan.

<u>Mission support procedures.</u> The PLS Mission Support Procedures are a living set of documents that provide the guidelines for conducting PLS missions. The basic document set consists of:

- Groundrules and Constraints
- PLS Operations Data Books
- Control Center Console Hndbks
- PLS Readiness Plans
- Mission Rules
- PLS Systems Hndbks
- PLS Training Plans

These documents are reviewed on a flight-to-flight basis to verify any changes to the mission requirements or PLS flight or ground systems which have been addressed and incorporated into the mission operations baseline.

<u>Flight support procedures.</u> The flight support procedures for PLS missions are contained in the PLS flight data file. This set of documents was derived from the NSTS flight data file inventory list. The procedures required to execute the PLS design reference mission have been identified from the NSTS list and the initial development support estimated for the first flight and recurring maintenance for the subsequent flights.

Documentation requirements. The flight operations and missions data/documentation required for PLS consists of the following plan and documents:

- Baseline operations plan
- Flight design hndbks
- Flight procedures mgmt plan
- Ground procedures mgmt plan
- Operational data mgmt
- Flight data file production hndbks
- Configuration mgmt plan
- Crew activity planning hndbks
- Control center readiness plan
- Software mgmt plan
- Training records mgmt

Our estimates used the STS Flight and Mission support data and networks to formulate the PLS estimates. The major reduction in manpower requirements was for Flight Design re-configuration criteria. The STS flies each flight in a unique configuration. This requires extensive modification to the flight software to support the payload community. PLS does not have this requirement for the reference mission (to SSF and back). By standardizing flight and ground procedures, a verified baseline can be used to initiate each flight with a minimum of changes to cover I-loads, and day of flight data. The use of automated tools for modification and production of flight design and crew activities reduces the overall support costs considerably. The establishment of an overall configuration management function that control all PLS program activities reduces the current conflicts in data that STS often experiences. Establishment of common data bases across PLS will also reduce support costs experienced in the areas of maintenance, logistics, launch and recovery, and flight operations.

We have identified a set of minimum flight documents which include! ascent checklist, post-insertion checklist (includes rendezvous and docking/berthing), deactivation/activation checklist, entry checklist, emergency landing, crew activity plan, Que cards, malfunction procedures, star charts, contingency abort procedures, contingency de-orbit preparations, systems data book, and medical checklist. The mission documentation will consist of mission rules, console handbooks, PLS operational data, PLS systems handbook, PLS training plan, and flight readiness plan. This set of data becomes the baseline set that each mission is built on. Modifications are required only when the performance of the PLS is changed. Changes to accommodate the specific mission are considered minimal compared to the STS (see Figure 4-23). As illustrated in the figure, the STS requires much more flight design data due to its unique flight to flight mission requirements and the reconfigurations necessary to support the payloads.

The PLS design reference mission (DRM-1) analyses covered; 1) a three-day nominal mission, 2) a two-day alternate mission, 3) use of KSC as prime site for return (with alternate return sites at Edwards, Northrop, Guam, and Hawaii), 4) launch window and vehicle assessment (using Martin Titan Commercial data for performance parameters), and 5) determination of rendezvous sequence and entry g-load assessment.

4.3.3 Mission operations assessment

A function-by-function analysis of the NSTS flight operations process has been performed. A cradle-to-grave approach was taken, from mission authorization through post-flight evaluation. NSTS contractor support levels were determined. PLS operational assumptions were applied, and the NSTS processes were adapted for both the PLS Orbital Flight Test (OFT) and mature operations environments. We have estimated the OFT Flight Design template to be 13 months and the mature operations template to be six months.

NSTS Mission Operations orchestrates all facilities, equipment, personnel, and processes required to accomplish pre-mission planning and real-time support. Among the pre-mission planning tasks are trajectory analysis, consumables analysis, mission procedures, flight rules, ground and flight software upgrades, data base management, and flight crew and flight controller training. The major real-time mission support tasks are actual troubleshooting, procedures and timeline support and updates, in-flight maintenance support, medical support, and search and rescue coordination. We found that all of these processes are necessary for PLS operations. However, the resources devoted to most of these items can be significantly reduced in a mature PLS program because of the relatively limited scope of its mission.

The development of operations requirements from a flight and missions standpoint that impact design include: 1) use of embedded diagnostics that annunciate faults to crew (and ground if requested), 2) a method to monitor fuel and consumables for loads assessments, 3) standardization of resource monitors that provide operational information in actual values (in pounds, time, distance, etc.), 4) minimizing flight-to-flight re-configurations requirement to reduce manpower and tasks (flight software/I-loads), 5) optimized designs to ease upgradability in hardware, software, and testing, 6) maximize performance with robust system designs to handle established margins (based on alternate DRM's that do not impact PLS envelope).

Figure 4-28 illustrates the flight and mission support operations development schedule that is required for the 1st PLS launch. This schedule represents the "operations capability development (OCD)" and "operational flight test (OFT)" program activities. Each element has been costed separately in order to determine projected costs for DDT&E WBS elements. The template for OFT has been set at 13 months, and for PLS mature operations at 6 months.

4.3.4 Key Mission Operations Assumptions

Key to the flight design effort is the understanding of boost vehicle performance characteristics. The Titan IV was used as the test case for our analyses. Newer flight performance data have been developed by Martin Marrietta for the Titan IV upgrade that indicates that we now can have more margin in the PLS to support flight and rendezvous operations to SSF. By using GPS rendezvous

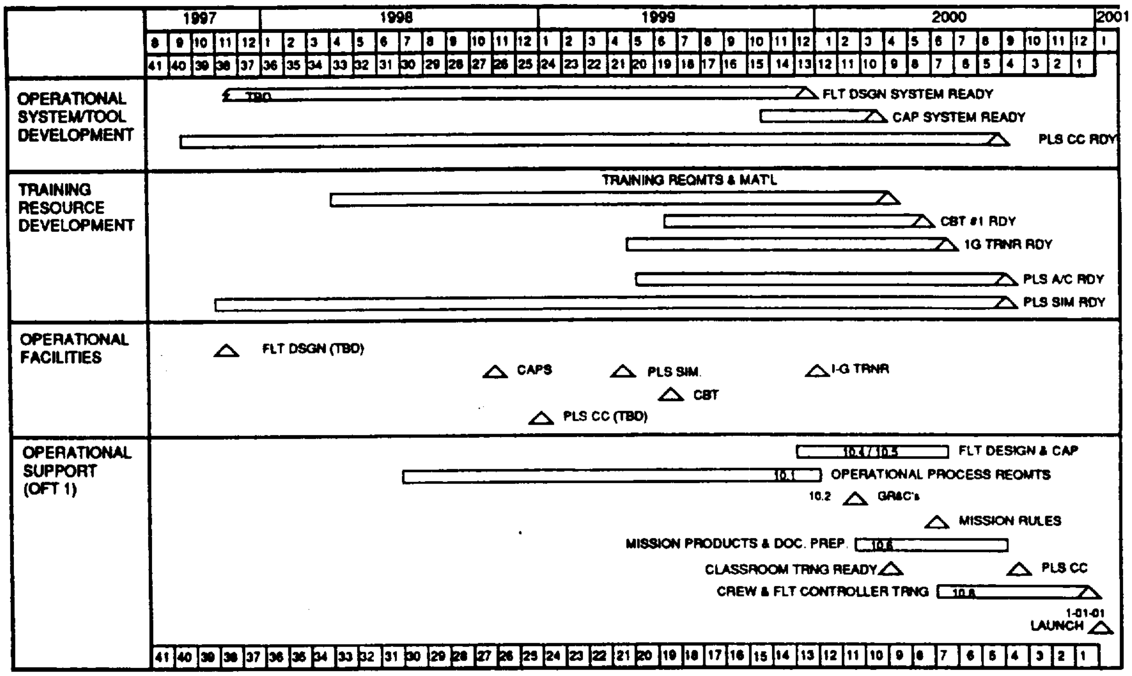


Figure 4-28. Flight Mission Operations Planning Schedule.

and tracking capability, we gain approximately three hours of time to make rendezvous. This projects a day of launch crew day of approximately 15 hours versus the projected time of 18 hours for the initial rendezvous sequence. Continued studies on the Titan IV to reduce launch processing time are underway and will be used in new assessments of the PLS and the Advanced Launch System (ALS) booster requirements.

PLS operations will differ greatly from the NSTS. Based on these differences, we have formulated seven key assumptions. They are:

1. Payload-specific activities are eliminated because the DRM-1 payload always consists of 8 passengers and associated baggage.

2. In-depth flight design analysis will be accomplished during OFT. Testing will verify DRM-1 capability and define operational limitations. Only minor variations due to SSF orbital parameters will be allowed during mature operations.

3. There will be no flight-specific onboard avionics reconfiguration because they will be baselined during OFT.

4. There will be no flight-specific ground software reconfiguration because it will be baselined during OFT.

5. The DRM-1 timeline is to be standardized, with only minor variations in time allowed due to SSF orbital parameters. There will be no variations in content.

6. Standard Groundrules & Constraints and Flight Rules are to be developed during OFT. There will be no effort to optimize each flight to obtain the last "inch, pound, and mile" of performance.

7. Because of the above items, the Flight Data File and all training can be minimized, simplified, and standardized.

4.3.5 PLS Mission Operations

PLS Flight Operations includes launch and maneuvering into the SSF orbit, attached operations, and return to Earth. Launch includes liftoff, ascent, orbit, and rendezvous and docking with SSF. Attached operations include crew transfer, system statusing, vehicle loading, and crew briefings. Return includes undocking, separation from SSF, de-orbit preparations, entry, and landing.

DRM-1 will be either a two or three day mission, with launch and nominal landing at KSC. Rendezvous and docking with SSF will be accomplished on the first day, at about nine hours after launch, with the crew sleep period to begin three hours later. Note that the greater navigational accuracy offered by the Global Positioning System is necessary to achieve this goal.

4.3.6 Flight Operations goals

One of the major goals of the PLS is to design for flight operations. Just as the vehicle itself will be designed for ease of maintenance, the PLS infrastructure must be designed to facilitate the operations. In order to accomplish this, it is necessary to integrate operations with SSF, NSTS, and the ELV while minimizing those interfaces where practical; and by maximizing the use of pre-planned procedures and tools. Flight Crew and Flight Controller procedures should be standardized to the maximum practical extent. By this we can simplify and minimize the human element in Mission Operations activities.

Expert systems will be developed and used extensively in preflight analysis and in the Mission Control Center. Initial qualification training and techniques and procedures development for Flight Crews and Flight Controllers require high fidelity integrated simulations starting at least 6 months prior to the first OFT flight. However, once trained and certified, crews can be assigned for specified operational tours, and then will form a cadre of instructors to train their replacements. This approach stabilizes the workforce, thus minimizing training requirements during mature PLS operations while maximizing the experience base.

4.3.7 Detailed Support Estimates

The flight operations detailed support estimates for the PLS study were developed for the operational capability development project phase, for the OFT flight phase, and for the PLS mature operations phase of the project. Figure 4-29 identifies the schedule for developing the functional plans and documents needed to support flight and mission operations development efforts.

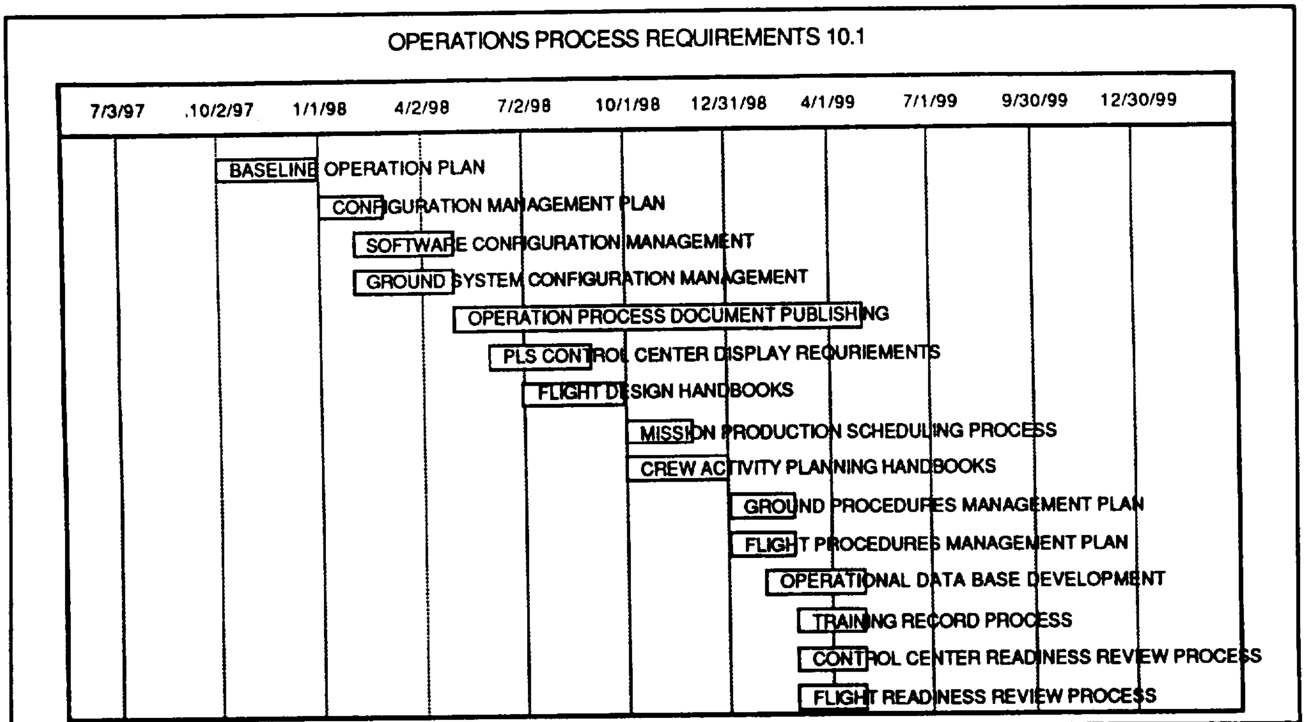


Figure 4-29. Program Control Documentation Development.

4.3.8 Flight Operations Capability Development

During the flight operations capability development phase, support estimates were prepared to cover the labor and materials associated with the design and development of the following flight operations systems:

- Flight Design System
- PLS Dynamic Simulator
- PLS Scheduling System
- PLS 1-G Trainer
- PLS Training Aircraft
- Crew Activity Planning System
- PLS Control Center

The development schedules were prepared using system software design, test, and integration as the pacing item for the ADPE based systems. The 1-G trainer and PLS training aircraft development focused on long-lead items and modifications cycle, respectively, to determine the development schedule.

The required operational facilities were identified to complete the flight operations capability development. The facilities required are:

- Flight Design System Facility
- PLS Computer-based Trng Facility
- PLS Training Aircraft Facility
- Crew Activity Planning System Facility
- PLS Control Center
- PLS Simulator Facility
- PLS 1-G Trainer Facility

All facilities are required to support computer based systems except for the support to the PLS training aircraft. The facility estimates were based on square footage and environmental requirements.

OFT Flight Operations. The preparation cycle for the PLS OFT flights is estimated to be performed over a 13-month template. The support estimates addressed the following functions:

- Flight Design & Trajectory Data
- Flight and Mission Documentation
- PLS Aircraft Training
- PLS 1-G Training
- Crew Activity Planning
- Computer-based Training
- PLS Simulation Training
- PLS Real Time Support

These activities for the OFT flights were estimated at the levels to support a project development phase and provided full time mission support during the OFT flights. It was also assumed that the OFT phase would validate the operations concept and support the creation of the flight operations data base.

PLS Mature Operations. The preparation cycle for the PLS mature operations flights is estimated to be performed over a 6-month template. The mature operations preparation cycle has been streamlined in terms of function durations and support levels. Reuse of flight and ground products is anticipated with only minor updates required on flight-to-flight basis. The support estimates for PLS flight operations functions were developed and submitted for inclusion and basis of estimate for the PLS life cycle cost.

Estimating Flight and Mission Resource and Manpower Requirements. In order to establish credible requirements, our flight and mission analyses were accomplished within the guidelines established in the functional flow block diagram for flight operations (Figure 4-30). We have developed estimates of manpower, software, facilities, and support networks based on functional requirements. These analyses indicated that we required 13 months (serial time) to develop and operational flight test (OFT) template and 6 months to maintain a PLS template during mature operations.

DRM-1 was assessed from a nominal and alternate mission requirements. This was accomplished using the PLS guidelines and assumptions. Titan IV performance was used to determine the PLS flight time to Space Station and return. Figure 4-31 illustrates the timeline that is developed for the data set. The revised orbit data indicate a reduction of three hours rendezvous time using improved booster data.

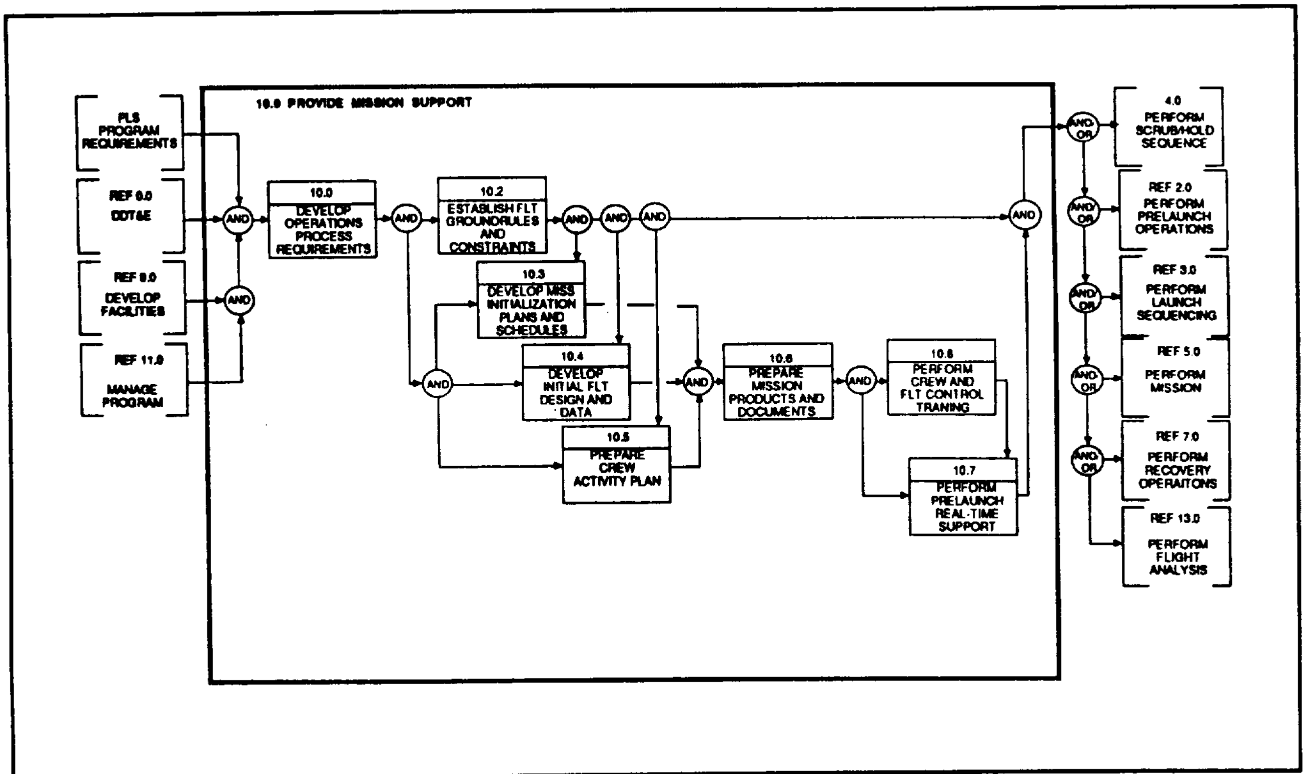


Figure 4-30. PLS Flight Operations Process.

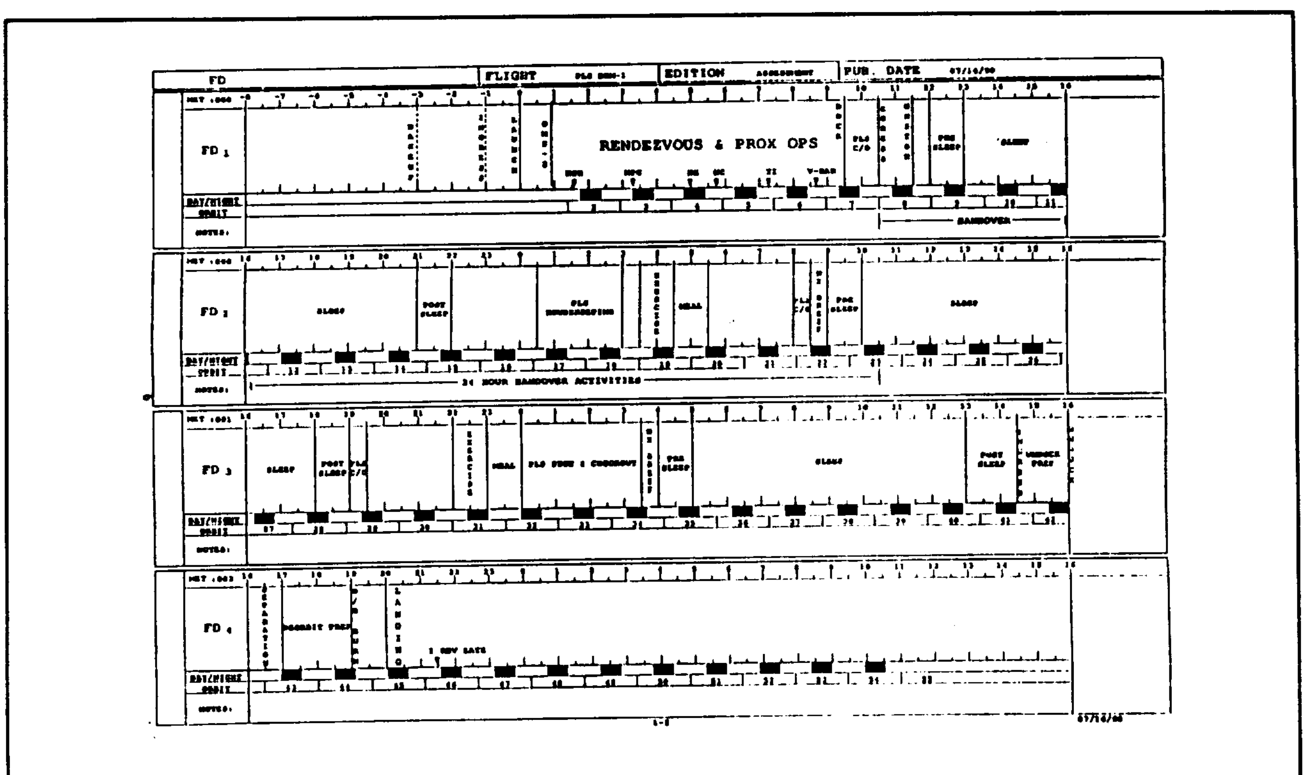


Figure 4-31. 72-hour Mission Timeline.

One of the major reductions in PLS flight and mission operations documentation development is the use of standard mission data sets for training and flight. This can be accomplished once the crew training has been completed during operational flight test program. A major reduction in PLS versus STS manpower is possible since the PLS has a mission specific task of going to SSF and not the unique tasks associated with new missions and payloads every flight.

Our mission control operations support concept is summarized in Figure 4-32. From this philosophy we have derived the processing documentation requirements for each mission support position. This concept reduces the support costs that impact real-time operations support activities.

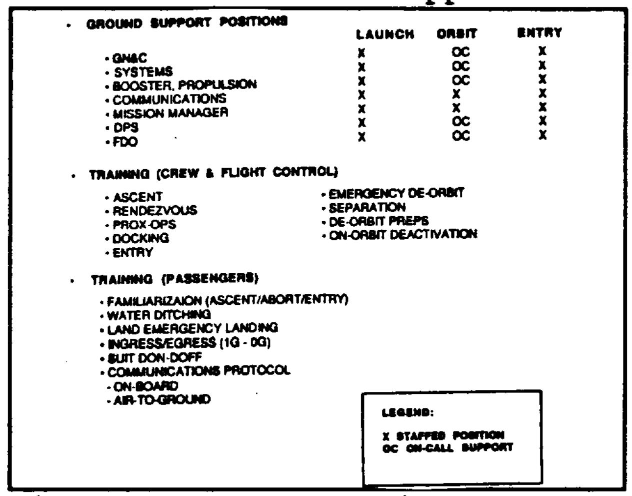


Figure 4-32. Operations Concept Data.

Post flight cost elements include the associated resources needed for: operational capability development, operational flight test, and mature operations. The basic software and hardware system development along with the development labor and facility costs were addressed. Other factors were system lease/maintenance costs, spares, maintenance and operations (M&O), and training of the operations personnel. Common support and non-productive costs were addressed for all program phases.

4.3.9 Implementation

PLS flight operations implementation and assumptions were derived using the NSTS process as a starting point and evaluating each shuttle flight operations function for PLS applicability. The PLS flight operations concept features a management and organization approach which has a dedicated flight manager assigned to each specific PLS flight. It is the flight managers' responsibility to initiate and monitor each mission in terms of baseline requirements and work authorizations for each functional support area.

Real-time support requirements for the PLS in the area of mission support are identified in Figure 4-33. The on call positions are for those periods that the PLS is at the Space Station. During flight operations, each position will be supported full-time.

Within each functional area there will be a lead individual assigned to supervise the activities and product deliveries for each specific flight in production. Functional "teams" will be

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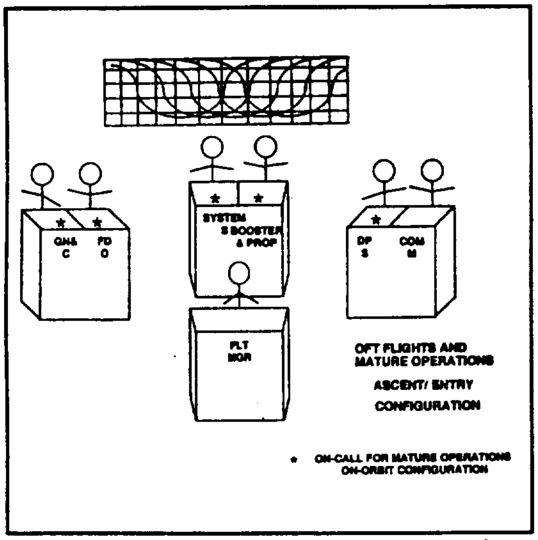


Figure 4-33. PLS Real Time Support Requirements.

assigned to flights on a rotating basis. This is also true for the flight crew and mission support personnel. The functional areas include:

- Flight Design
- Crew Activity Planning
- Flight/Mission Documentation
- Training
- Real Time Support
- Facility M&O Support
- Common Support
- Sustaining Engineering

This approach allows the optimum support level to support the flight rate over the life of the project. It is expected that increased efficiency and experience gained will allow incremental steps in the flight rate to be supported with the baseline support levels.

4.3.10 Staffing Levels

PLS flight operations production templates were developed for the OFT and mature operation flight cycles. Based on the function duration as reflected in the template and the man-year estimates to perform the tasks, staffing levels were derived and provided to the cost analysis team as the basis-of-estimate for the life cycle costs. NSTS resource allocations were used as references to determine the ratio of engineers to technicians for the PLS staffing. Except for the real-time support to the OFT flights and specified control center position support to mature flight operations the PLS support was conducted on a one shift per day, five day per week schedule.

The PLS flight operation staffing levels are summarized in Reference 4-1. It explains the rationale for PLS staffing estimates, and provides a recommended NASA role consistent with PLS operating concepts. Table 4-4 identifies the headcount estimates based on the functional tasks associated with flight and mission operations.

The staffing levels for PLS flight operations have been estimated using a bottom-up approach based on a function-by-function analysis. The specific functions were deemed applicable to the PLS DRM-1 scenario. As alternative DRMs are addressed a review of the functions should be accomplished to identify adjustments necessitated by additional mission requirements.

Table 4-4. Headcount Estimate for Flight Planning and Mission Control.

Functional breakdown	10.1	10.2	10.3	10.4	10.5	10.6	10.7	Miss Ops	**	10.8	Total
						(474 Hrs)					
Engineer			11.50	2.50	0.17	2.00	0.25			5.94	
Technician			0.50	0.83	0.06	0.67	0.08			1.98	
Management			0.15	0.25	0.02	0.20	0.03			0.60	
(T) Schedule			8W	11W	3W	8W	5W	5W		11W	
Max in Flow * For headcount derivation purposed assume 10.1 & 10.2 absorbed by other personnel since these Ops costs estimated to occur every four years			2	2	1	2	1	1		2	
Total Hours			4042	6730	470	5396	673			16108	
Hrs avail ~ 1st Shift			320	440	120	320	200			400	
Headcount Activity			12.6	15.3	4	16.9	3.4			35.4	
Headcount Required			26	31	4	34	4	9	24	73	
Headcount Breakdown											
E (.697)			18	22	2	24	2	7	24	51	150
T (.232)			6	7	1	8	1			17	40
M (.070)			2	2	1	2	1	2		5	15
											205

* 8 Launches Per Year

** Mission Control Center Mission Support Headcount

4.3.11 Trade Study - Automation of Flight/Mission Planning (Trade T7)

The mission planning trades centered on the options associated with the ground network that interfaces with the flight and crew planning activities. The primary objective was to trade use of increasing levels of automation in the mission planning activities. We evaluated the STS operational resources and planned automation efforts that would have PLS application. Figure 4-34 identifies the STS/PLS resource assessments that were analyzed.

The flight design and crew planning activity were two candidates for automation. This is due to the similarity in the software packages and the need to verify the data between each plan. The interface between these packages mainly deals with trajectory data and which occurs by transmittal of a magnetic tape referred to as a "Super Tape". If these systems were to reside on a common host computer systems, certain benefits could be realized in terms of data management, maintenance and software lease costs. Figure 4-35 illustrates the existing and candidate configuration for modifying the current automated mission planning process.

The automation of the PLS flight and crew activity planning data is highly recommended. However, the cost associated with such an effort is considered to be high (approximately $5M for development and integration). Further study of this area is required since the use of commercial packages was not evaluated.

STS RESOURCE	FUNCTION	PLS REQUIREMENT	RATIONALE
FLIGHT DESIGN SYSTEM *	FLIGHT DESIGN: · ASCENT · ORBIT · RENDEZVOUS · DESCENT · CONSUMABLES	PLS ADAPTATION - · RMS · EXP. POINTINGS · VIEW RQMTS (EARTH)	BASIC PLS FLIGHT DESIGN CAPABILTY REQUIRED
CREW ACTIVITY PLANNING SYSTEM *	DEVELOP MISSION TIMELINE (CREW ACTIVITY PLAN)	PLS ADAPTATION - SMALL WORKSTATION SUBSET	MEET PLS FLIGHT RATE RQMTS
SOFTWARE DEVELOPMENT FACILITY (SDF)	PRODUCE CORE FLIGHT SW CONFIGURATION ITEMS	PLS AVIONICS LAB	EVOLVE INITIAL PLS FLT S/W
SOFTWARE PRODUCTION FACILITY (SPF)	RECONFIGURE CORE SOFTWARE ON FLIGHT SPECIFIC BASIS	N/A	USE PLS AVIONICS LAB
SHUTTLE AVIONICS INTEGRATION LABORATORY (SAIL)	VERIFICATION OF FLIGHT SOFTWARE LOADS	N/A	USE PLS AVIONICS LAB

* CANDIDATE SYSTEMS

Figure 4-34. STS/PLS Resources Assessment.

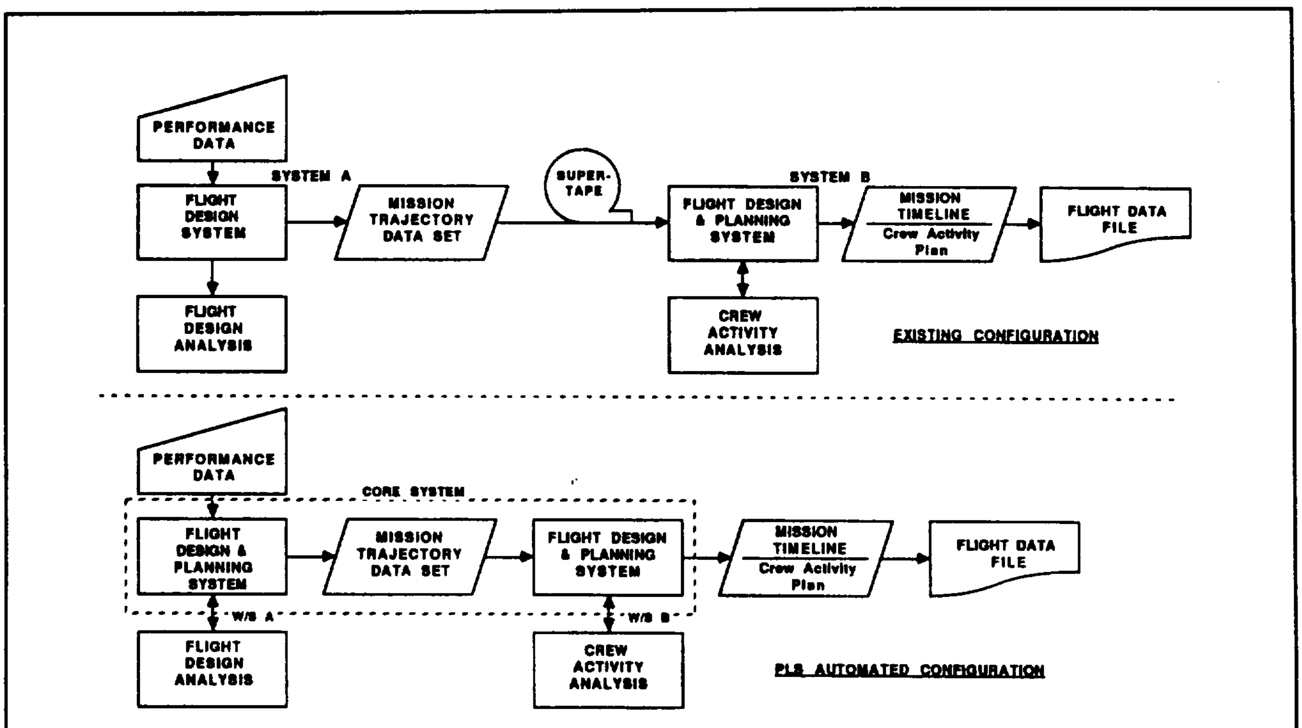


Figure 4-35. Mission Planning Automation.

Although one of the major factors in this study that dealt with subsystem design of the PLS on-board equipment was not addressed here, it was addressed by the flight and crew planning teams during the timeline analyses. These data were used in the development of guidelines that define operational constraints against the flight architecture. It is our plan to document these requirements during the continuing studies currently planned. Traceability of requirements in this area should lead to a robust system with good margins to minimize flight software reconfigurations for each flight.

4.4 SUPPORTABILITY

The primary objectives for the logistics analysis activity has been to provide supportability data (spares, repair, manuals, maintenance training, depot support, transportation, warehousing, and support equipment) to the design and costing teams. These data included spares quantities, GSE listings, off-equipment training/certification requirements, repair estimates, depot support personnel requirements, and predicted repair quantities based on our spares and repair models.

4.4.1 Task Analysis

The logistics support concept has been determined by first examining the factors that drive logistics. Off-line maintenance drives the logistics support requirement from a maintenance aspect. Three types of resources are required; 1) spare vehicle LRU's and/or maintenance consumables, 2) support equipment, and 3) technicians. There is also a fourth but intangible resource consumed and that is time. In determining support resources: 1) warehousing, 2) support equipment spares, 3) test equipment for support equipment, 4) personnel training, and 5) operations/-maintenance instruction, times must be evaluated for turnaround of repair resources. Figure 4-36 illustrates this baseline off-line maintenance support concept. Spares, support equipment, and manuals are the significant drivers on life cycle costs. These data represent a major support base of material that can be analyzed by varying missions, flights, vehicles, and maintainability features.

During our analysis we have examined the logistics program costs for the following eight items:

Depot Support Equipment	Organization Maintenance
Depot Maintenance	Depot Manuals
Organizational training	Depot Training
Warehousing	Management
Packaging, Handling, Storage,	and Transportation

The factors that we used in the assessment included:

Vehicle Description -	crew size, weights
Operations Description -	number of vehicles, operating hours/years, power on times
R/M Factors -	reliability, MTBR, MTBM, MH/MA, sufficiency levels

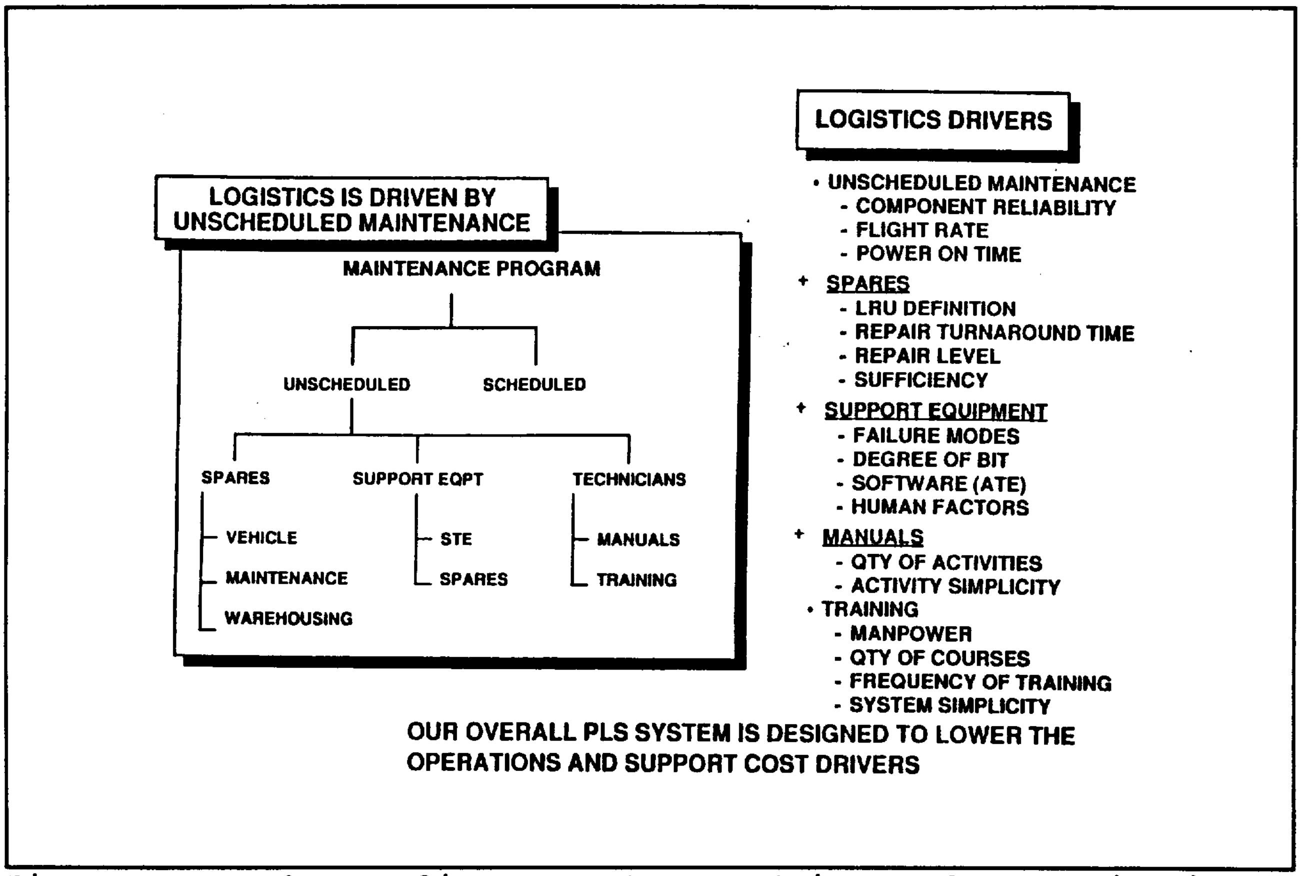


Figure 4-36. The Baseline Support Concept is Based on Examination of Support Drivers.

Depot Factors - turnaround times, mean time to repair

Logistics Factors - transportation, LRU types, Manuals

From these data, spares and repair costs were developed in detail.

One of the driving factors used in our analysis is the estimated failures and removal requirements after each flight. Our estimates also include ground power-on time since that was the major contributor to unscheduled maintenance on the Shuttle orbiter. In our current processing cycle, projected PLS ground power-on times are brief enough that they do not contribute significantly to the PLS spares estimates. This is mainly due to the simplicity of PLS design and the short turnaround requirements at the horizontal processing facility. Figure 4-37 illustrates the variableness of the ground power-on-time on PLS. As illustrated, ground power is not significant on our predictions for spares requirements with a 50 to 100 hour value. We used a baseline value of 200 hours ground power-on time. This provides a margin to the spares estimates since we do not have a firm subsystem and assembly design.

The use of the Rockwell developed MAtrix model for determining spares has provided us with a more realistic approach than what has

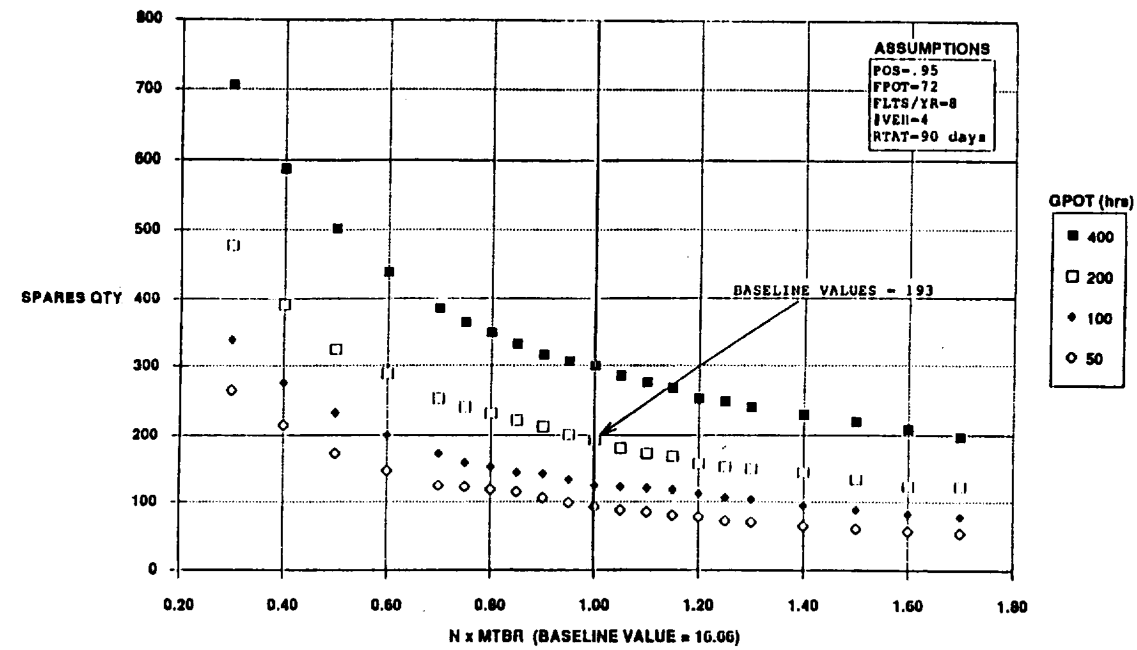


Figure 4-37. Predicted PLS Spares as Functions of MTBR and Ground Power-on Time.

been used historically for space vehicle applications. Spares predictions using MTBF in lieu of MTBR increases the spares forecast considerably (up to 1.5 times over real value). The use of traceable failure data to similar systems, with a space rating applied, assures us that the values are within a high confidence limit (greater than 85 percent).

The Logistics function has been an active member of the concurrent engineering team during the PLS conceptual design analyses activity. Contributions have been made to: 1) influence design by development of requirements and examination of subsystem configurations, 2) improvement in spares forecasting using MAtrix modeling, 3) identification of logistics drivers, and 4) defining off-vehicle ground operations/processing through support system assessment using various analytical approaches.

Our support concept is directly traceable to program goals of reduced operations costs and ground operations simplicity. To facilitate achievement of the program objectives, logistics support requirements need to be developed/imposed on the system design early.

An implied support infrastructure implies that: 1) a maintenance and operations program will require minimal activity on the vehicle, 2) automation of current processes reduces maintenance time and the associated administrative activities, and 3) it takes advantage of multipurpose ground support and test equipment to reduce the range and depth of required support.

A reduced initial support investment implies that: 1) the system will utilize existing assets, where practical, and 2) the burden of building a large depot repair capability and or spares stock be eliminated by employing repair warrantee concepts.

A reduced maintenance demand implies that: 1) we will depart from an operational concept of re-certifying every system of the vehicle prior to each flight, and 2) take advantage of proven hardware with known reliability performance to reduce maintenance requirements.

No launch delays for normal maintenance implies that: 1) establishment of repair time requirements, regardless of repair location will decrease risks associated with launch schedules/windows, and 2) vehicle systems and subsystems will be designed to ensure achievement of repair time requirements.

4.4.2 Procedures Definition

Some of the key findings in the technical documentation/manual development area for logistics have been: 1) maintenance documentation costs can be most effectively reduced by introducing automation when the systems developed and produced, 2) maintenance technical manuals are dynamic documents which require timely coordination to ensure current hardware and component configuration are available to user, 3) estimated cost of changes to technical manuals can be reduced 70 percent if baseline data are digitized, and 4) costs of technical manual development can re reduced if design engineering and manufacturing data are digitized and in reusable formats. (DoD reports the cost of manuals at $600 to $1200 per page without automation.)

It is recommended that the PLS program formulate a strategy for mandating use of neutral digital exchange standards and technologies whenever data and documentation are developed/produced for the project. The program should avoid program unique hardware and software solutions for documentation production, shortage, and distribution where possible. By focusing maintenance documentation development through a thorough task analysis process and by optimizing repair level analysis techniques, major reductions in costs can be realized.

4.4.3 Detailed Support Estimates

Our primary goal has been to provide supportability data requirements (spares, repair, manpower, transportation, warehousing, training, etc.) that can be derived for logistics support and cost those to the appropriate WBS level. The level of

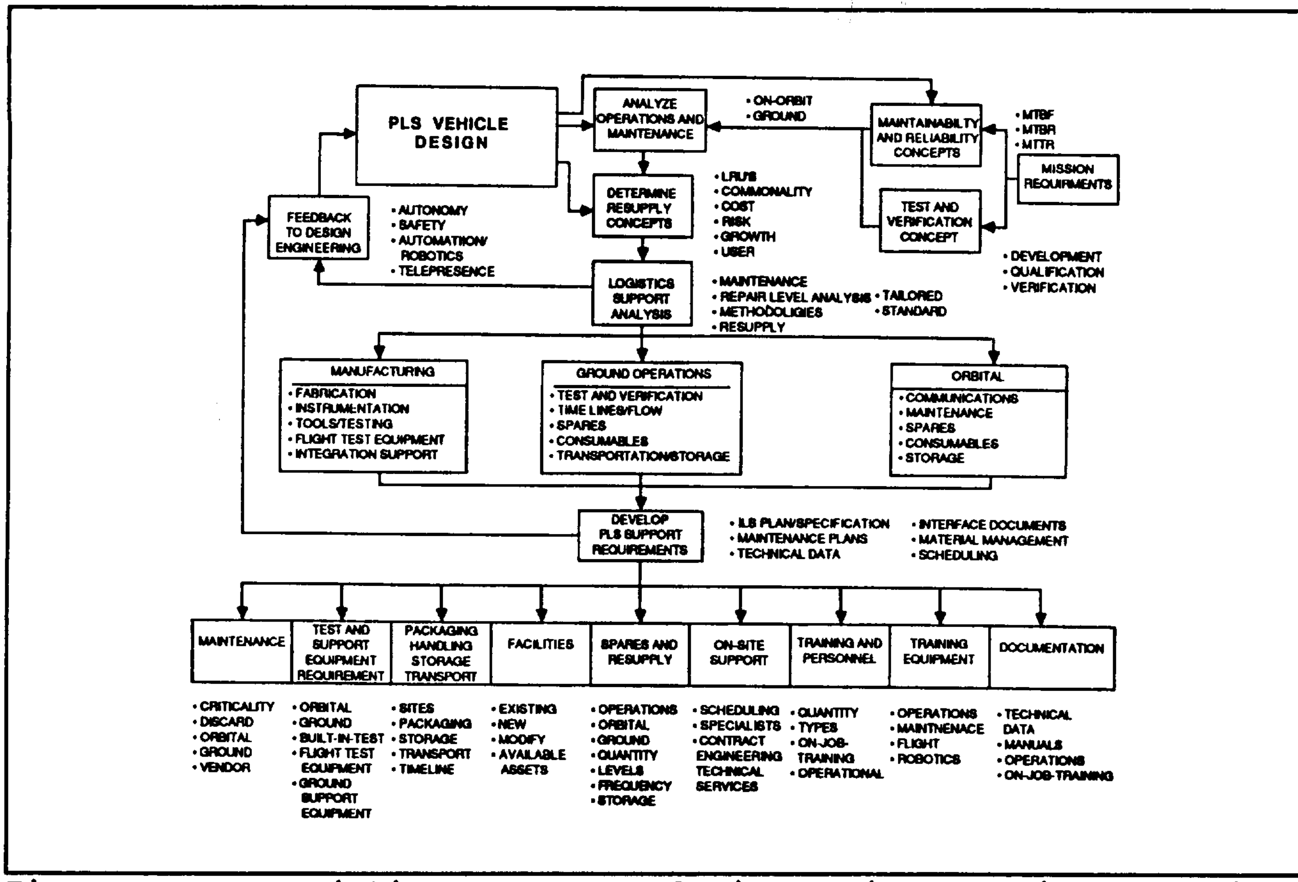


Figure 4-38. Logistics Support Analysis Requires Detailed Planning.

detail varies by subsystem; therefore, we started with a top down analysis and later defined those elements that required more detailed data to define the support parameters properly. Figure 4-38 depicts the types of data that end up in a fully operational support system. We have provided a spares and reliability/maintainability model that captures their intent. The development of detailed supportability estimates requires that the logistics function be an integral part of the design team. The specific task activity was to define support requirements for all phases of operations including, 1) unscheduled maintenance, 2) spares quantification, 3) ground support equipment, 4) task times, and 5) staffing levels. Logistics also provides support to ground operations, maintainability, and cost functions.

The data synthesis included identification and definition of:

- Initial Spares Lay-in
- Off-equipment Repair
- On/off Equipment Consumables
- Recurring and non-recurring GSE
- Recurring and non-recurring MSE
- Recurring and non-recurring ATE
- Recurring and non-recurring Off-Equipment Manuals
- Recurring and non-recurring Warehousing
- Recurring and non-recurring Transportation

• Recurring and non-recurring Off and On Equipment Training

Detailed technical data and analyses based on the reliability estimates and support requirements have been provided during the study (quarterly reports and cost reviews). Specifically preliminary spares, technician, technical data, training, and ground support equipment assessments were performed. These documents provide a representative example of the reports that were generated on each supportability aspect to date.

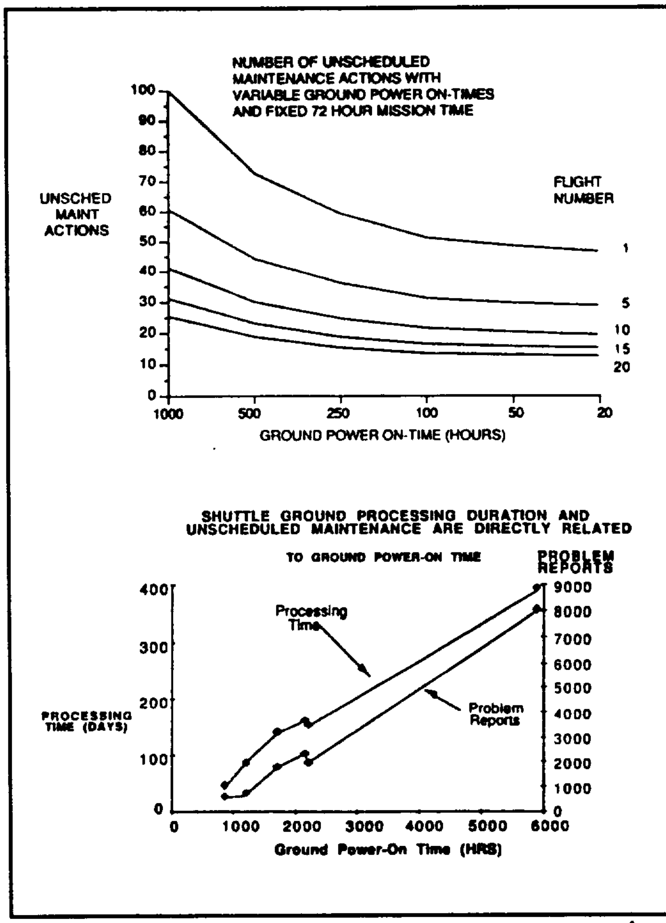


Figure 4-39. Ground Power-on Time Drives Expected Failures.

Detailed assessments of spares requirements and repair actions have been evaluated for infant mortality, ground power-on-time, and mission duration. Figure 4-39 illustrates the expected failures of PLS systems due to ground power-on time Upper diagrams) and a comparison to the STS-based experience. It becomes evident that the low times shown for PLS (based on the 14 day HPF time) do not compare to STS. The STS, once again, has a unique mission requirement and the amount of configuration changes, payload integration changes, and an overall requirement to re-certify all systems prior to flight drives the ground power-on-time higher than a normal airline would.

An evaluation of predicted failures over a number of flights is shown in Figure 4-40. The main issue with the PLS is to determine what systems are going to cause maintenance impacts and then determine how to make a design change recommendation that can reduce the failure values, or identify a method to support the system effectively.

In the detailed analysis effort considerations have been given to supportability. For instance, the dominant vehicle design theme has been to "design for accessibility". This reduces maintenance times, GSE requirements, induced failures (due to design complexity), and reduces manpower/personnel requirements. Use of "built-in test (BIT) and health monitoring" adds to the benefits attributed by accessibility by further reducing the skills and training requirements for fault isolation. Imposing a processing characteristic of "no post-flight decertification" of the total PLS reduces the requirement for technical data, spares, maintenance consumables, GSE, and significantly reduces processing turnaround times.

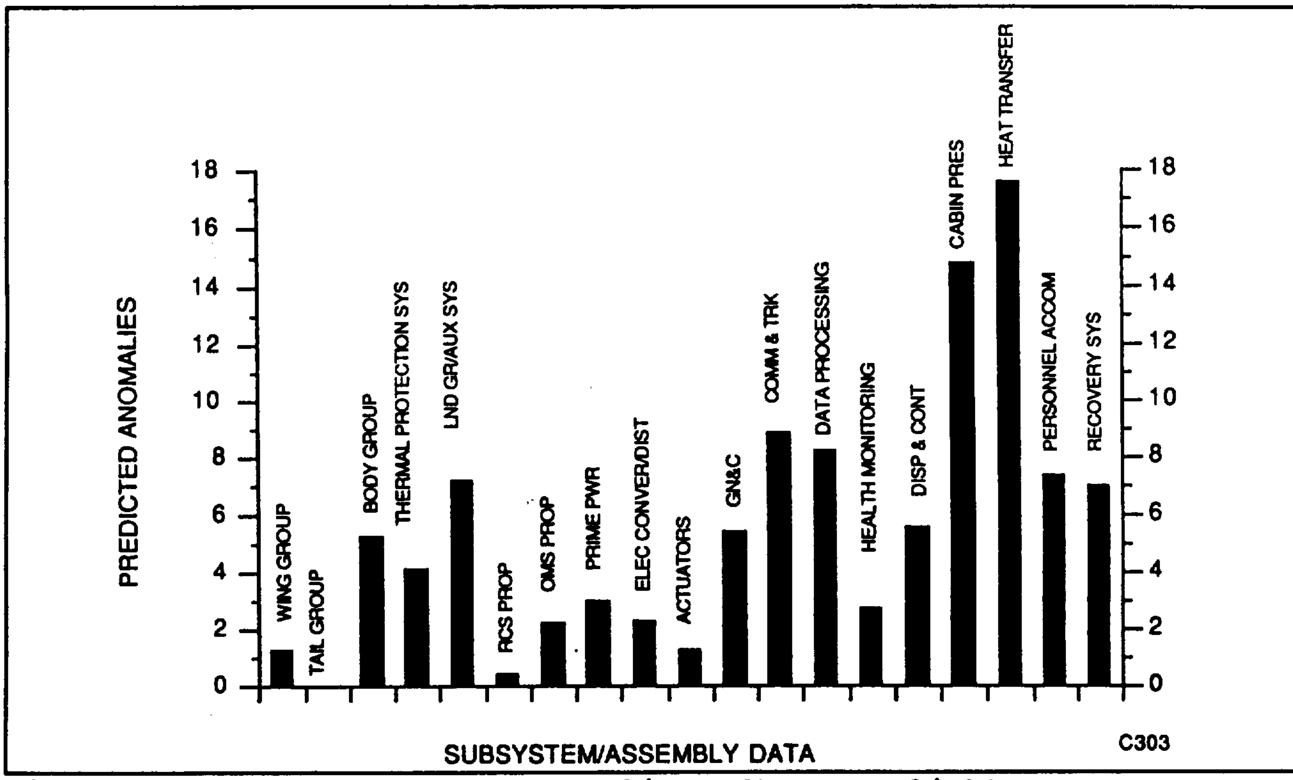


Figure 4-40. Expected PLS Anomalies After 29 Flights.

4.4.5 Implementation

Management of the logistics organization would basically remain within the KSC infrastructure and support organization. This is mainly due to the methods for handling logistics over the life of any program. The key to economic support policies is to merge all functions that are common into one organization and then use the resources that are available on an as-required basis. The documentation of spares and repair data is accomplished during DDT&E with a minimum amount of changes that would not be covered under the standard change control process.

Analysis of the spares requirements for PLS reveals that the mean-time-between-removal (MTBR) factor drives the spares acquisition process if the value is off by more than 15 percent per vehicle quantity. Since the spares data use actual aircraft failure rate information (modified for space applications), we believe that our values have a high correlation ratio to actual data and that the MTBR would not vary over 10 percent.

Figure 4-41 indicates that the quantity of vehicles (2.5, 3, and 4) each have a different impact on the spares requirements. The reason that the four vehicle spares quantity is lower than the three vehicle spares quantity is due to the safety spares calculation. With a four vehicle inventory, safety spares becomes insignificant and therefore they are not required to assure a PLS flight. This also indicates that the recommended initial spares and recurring spares for operations support are not too sensitive to turnaround times and that the flight rate can be supported by the recommended spares. The final values of spares can not be determined until the design has been completed at critical design review, however, the LRU values contain the support materials to effect a repair as predicted. The shop replaceable units (SRU's) are a part of the overall value.

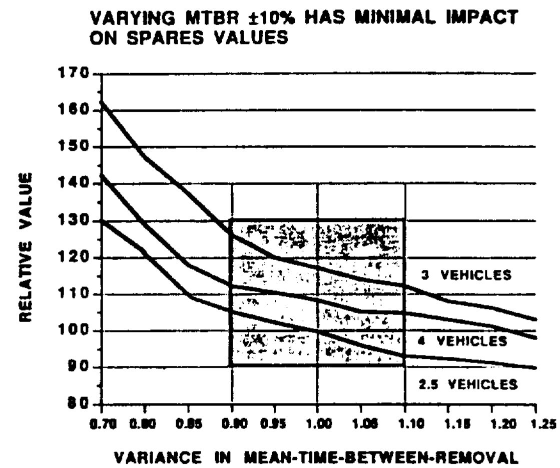


Figure 4-41. Variance of PLS Spares Requirements.

Table 4-5. Traceable Design Guidelines Lower Life Cycle Costs and Increase Availability.

Design Feature	Quantitative Goal	Rationale
Mean-time-to-repair	3 Hours	Directly Related to Launch
Maximum Maintenance Time	6 hours	Directly Related to Launch
Percent Fault Detection		
Critical 1 Failures	100%	Assures Attainment of Safety, Availability, and Life Cycle Cost Goals
All Other Failures	95%	
Percent Fault Isolation	95% of Detectable Failures to 1 LRU Within 1/4 hour	Supports MTTR Objectives and Assures Attainment of Availability & LCC Goals
False Alarm Rate	1.5%	Support 98.5% Overall Availability Goals

Table 4-5 identifies the design guidelines that were imposed on the design team during the study. This helped us to assure that the PLS vehicle design could be assessed against some primary operations support drivers. One of the key requirements for fault isolation leads the design team to embed diagnostic capability into each subsystem. It also meets the autonomy guidelines for the PLS. The false alarm rate value assures us that the design will not have ambiguous test networks that require multiple test paths in order to determine the subsystem failure. The STS has experienced a false alarm rate of approximately 30%. This type of value leads to excessive test and analysis time and can actually increase maintenance due to the unnecessary removal of good assemblies.

4.4.6 Staffing Levels

The logistics manpower has been included in the ground operations estimates. Manpower and resources necessary to perform DDT&E functions are included (as support personnel) in the overall manpower estimates. With the PLS using aircraft type subsystems the support packages for these items should be readily available. This would mean a minimal amount of effort would be required to acquire the data in the standard DoD formats. With the assumption that the PLS program is to use MIL-STD-1388-1 & 2 as the developing document for the support packages the task of converting the data is not costly.

4.4.7 Trade Study - Automated Processing Data Management (Trade T6)

This trade determined the automated processing data management techniques that can be implemented in the repair of a space vehicle such as PLS. Briefly, the study identified a cost effective strategy for introducing automated technical documentation products into the ground processing environment.

Five factors were addressed in the study: 1) changes at KSC, 2) maintenance documentation attributes, 3) levels of automation at DoD applications, 4) cost, and 5) recommended strategies.

The principal finding was that introducing an integrated automated data management process into the ground processing environment is cost effective. However, even larger costs savings can be achieved if data standards are incorporated early in the technical documentation development process. This includes the development of a management information system that handles all project data from the Phase A inception through the fielding and operations activity.

The primary recommendation is that the PLS program develop a technical documentation acquisition strategy and an automated processing data management system strategy that are compatible with each other and complies with national standards for neutral data exchange. Further work is needed, in order to develop specific recommendations for incorporating the findings of this study into a more detailed acquisition strategy for the PLS Program.

Maintenance documentation for the PLS is the key area of study because maintenance documentation represents a significant portion of the technical documentation that will be delivered. In addition, investigating this subset of technical documentation in some depth will provide results that will be applicable to other subsets of technical documentation that will be delivered for the PLS program. Application of automation to maintenance documentation has also been the subject of significant efforts by the Department of Defense (DoD) and DoD contractors in the area of military aircraft, vehicles, and ground systems. As a result, the PLS program can benefit by incorporating the results of these efforts into an automated technical documentation strategy.

Introducing increased amounts of automation in the ground processing environment should be cost effective. The Shuttle Ground Operations Efficiencies/Technologies Study (Reference 4-4) incorporated extensive analysis resources in identifying issues, conducting trade studies, and identifying high payoff recommendations. In their list of "Shuttle Lessons Learned (Applicable to Future Vehicles)", the following findings apply: "Analysis indicates the greatest improvements in current operations can be gained via redesign of Shuttle program Data Management System (SPDMS) to conform to TMIS and associated systems. Potential Savings -- $3.0B plus increases of up to 30 percent in launch rate (based on FY85 rate of eight per year)". In other words, if the Shuttle program Data Management System was automated and integrated incorporating neutral data file and transmission standards instead of the isolated mainly paper-based system that currently exists, then the quoted savings could be realized. (TMIS is a NASA information system that incorporates these features.)

Similar findings were contained in a technical memorandum entitled "Reducing Launch Operations Costs" that was delivered to Congress by the Office of Technology Assessment in September 1988 (Reference 4-5). It contained an estimate that "an integrated paper-less information management system could reduce the time spent in launch operations by one-half". Also included were recommendations to minimize custom hardware and software and to standardize the architecture of onboard and ground systems including code. These recommendations were specifically identified for inclusion in a future launch system.

As a result of the above and other studies, many of the important data bases at Kennedy Space Center are in the process of being automated. This effort, the Shuttle Program Data Management System, has the goal of connecting many currently existing but isolated data bases. The above effort is important to the PLS program because the final system implemented for the Space Shuttle program will likely be a prototype of a similar system into which PLS technical data will flow. So, while the Space Shuttle program must incur extra costs to transform information into a form that can be readily used in a digital network, the PLS program can avoid this costly step. If technical data developed during the PLS program is delivered in a digitized format that can be readily used in a digital network, then baselined technical information would be readily available in an integrated automated information management system.

The PLS program should adopt a technical documentation acquisition and ground processing information system strategy that consists of four principal ingredients:

(1) Insert automation requirements early in the system acquisition process.

(2) Insure data are digitally reusable and transferable.

(3) Avoid custom hardware and software. Insist on compatibility with national standards.

(4) Insure technical information developed during the program is directly useable by an integrated automated information management system during ground processing.

4.5 SYSTEMS ANALYSIS

4.5.1 Fleet Sizing

Fleet sizing for the preferred three vehicle PLS fleet assumed: 1) a SSF crew stay of 180 days and a flight interval of 60 days, 2) initial SSF operations is supported by the Shuttle every 90 days, 3) complete crew change out would exceed Shuttle capability (eight per mission), 4) SSF operability requires crew overlap, and 5) Lunar/Mars personnel transferred on dedicated flight until crew is complete.

The SSF crew size of 24 is expected between 2007-2020. The duration of stay is 180 days with a total personnel exchange of 48 during a given year. This equates to 6 flights with a launch interval of 60 days to transport 8 SSF personnel each flight. The flight crew is a separate quantity of people. This also equates to a total of 141 PLS flights through the year 2020.

4.5.2 System Attrition

In-flight aborts of the PLS vehicle do not necessarily denote loss of vehicle e.g.: mission abort. Catastrophic vehicle loss represents a small fraction of the total number of aborts, and approaches zero. However, a number of PLS failures, occurring singularly or in combination with others, may cause mission abort. That is, one or more required functions may not be available for use when needed. Should this happen, the PLS simply would safely return to earth for subsequent repair and reuse.

PLS loss also may occur as a result of catastrophic failure of the Titan IV Launch Vehicle. The most recent projections for Titan IV Launch Reliability range between 0.96 and 0.98. The PLS should be capable of escaping from all but a no-warning Titan IV explosion occurring on the launch pad or in flight. A Titan IV no-warning explosion is considered to be a very unlikely event. Accordingly, it is unlikely that a PLS loss will occur at a rate greater than 1 per 1,000 launches.

4.5.3 Estimated Repair Turnaround

The estimated repair turnaround analysis was performed in detail. Upon review, it was determined that the repair turnaround time (RTAT) was not a significant driver to the PLS program. This is mainly due to the low flight rate, the number of supporting vehicles to meet flight rate, and the number of system components that require repair during any given repair period. This is also due to the improved reliability of system components, and the method used to derive actual LRU failures that require intermediate or depot support.

4.5.4 Vehicle Impacts Due to Flight Rate

The current quantity of recommended vehicles to support the PLS program allows for attrition and resiliency. The need for additional vehicles to support other missions was not evaluated during this reporting period. However, the use of a dedicated vehicle to support a unique mission scenario could be handled with the resources currently planned for PLS (if no attrition occurs).

4.5.5 Facility Capabilities and Resiliency

The current facilities planned for PLS would allow for growth to six vehicles (with shared work bays). The current plan is to use two bays to support turnaround operations and to use the remaining two bays for maintenance and staging.

5.0 HARDWARE/SOFTWARE DESIGN SPECIFICATIONS/DESCRIPTIONS

The philosophy for PLS subsystem design is to support the requirements and objectives of the manufacturing and system operations functions. Traditional spacecraft design criteria, such as weight and performance, do not have the same relative importance on the PLS program as do life cycle cost, ease of manufacture and minimum ground operations. As a result, the subsystem concepts selected justifiably depart from conventional spacecraft design in some cases in order to achieve the desired or mandated subsystem characteristic. All subsystem concept selections meet the technology readiness guidelines of the PLS program.

5.1 CONFIGURATION AND SUBSYSTEMS DESCRIPTION

The Personnel Launch System defined in this report performs a single reference mission: to transfer eight Space Station personnel to and from Space Station Freedom. Two PLS crew members operate the spacecraft. The design of the vehicle and its subsystem selections and arrangements are driven by the low cost of ownership requirements.

5.1.1 PLS Vehicle General Arrangement

The vehicle is sized to fit within the payload envelope of the STS Orbiter (with fins folded) to comply with a non-groundruled STS delivery option. The vehicle will transport a crew of two and eight passengers to Space Station Freedom in the DRM-1, three day mission. The internal arrangement of the subsystem components and personnel integrates the requirements of subsystem function, structural load path efficiency and allowable center of mass range. With the preferred set of subsystem concepts, the vehicle has a dry weight of 17,335 pounds, a landed weight of 20,705 pounds and a launch weight (with booster adapter and escape system) of 41,420 pounds.

Figure 5-1 presents the general arrangement drawing of the PLS vehicle. The locations of the major subsystem components are shown along with the significant dimensional data for the vehicle. More detailed descriptions of the separate subsystems are found in subsequent sections of this document.

5.1.2 Mass Properties

The mass properties of the major vehicle elements and subsystem components are given in Table 5-1. The weight data are derived from exact data available for off-the-shelf components to approximations developed from historical parametric relationships based on physical characterization, such as component area or volume or performance capabilities such as power output or input.

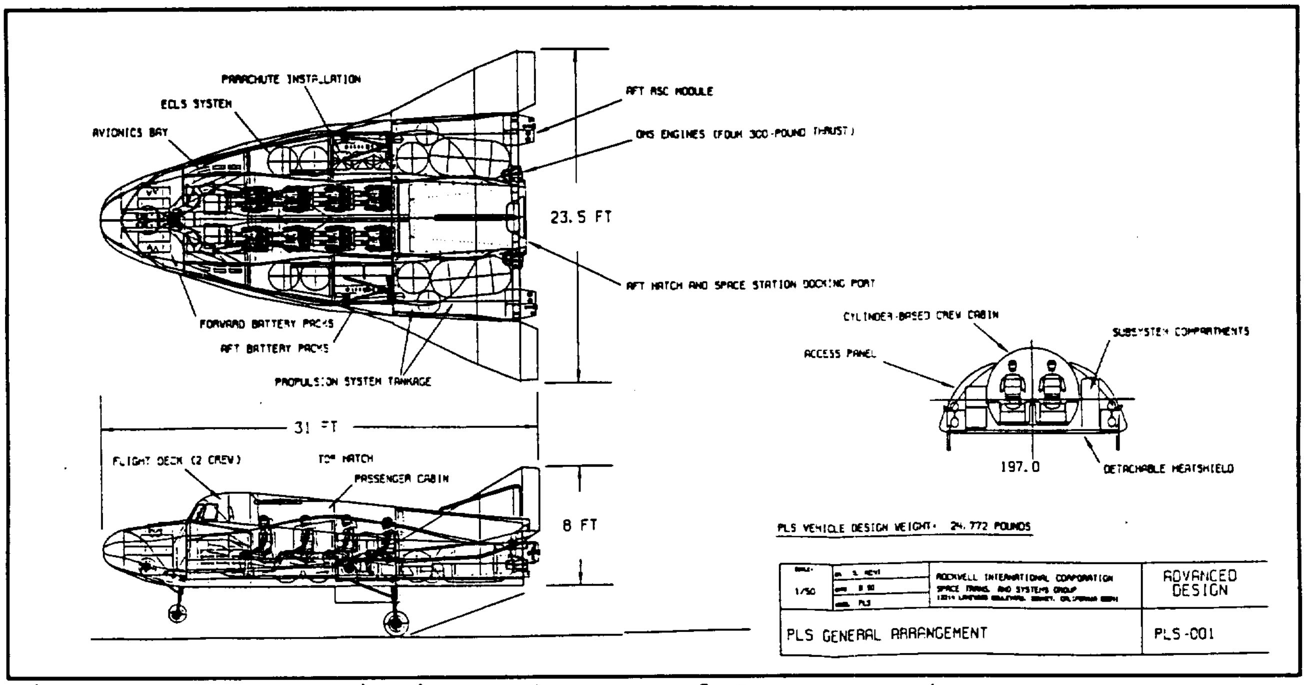


Figure 5-1. PLS Lifting Body General Arrangement

Table 5-1. Mass Properties

	DESIGN WEIGHT	
WING GROUP	1,739	
TAIL GROUP	62	
BODY GROUP	2,907	- LARGE TRANSFER TUNNEL
THERMAL PROTECTION SYSTEM	1,555	- SIZED BY AEROTHERMAL ANALYSIS
LANDING GEAR	829	
INTEGRATED PROPULSION	1,138	- JP4/H2O2 CONCEPT
PRIME POWER	2,720	- RECHARGEABLE Ag-Zn BATTERY PACKS
ELECTRICAL DISTRIBUTION	999	
ACTUATORS	123	- ALL EMA
AVIONICS	956	- ADOPTS ASCM TECHNOLOGY
ENVIRONMENTAL CONTROL	1,478	- ACHIEVED WITHOUT RADIATOR
PERSONNEL ACCOMODATIONS	1,195	
RECOVERY & AUXILIARY SYSTEM	1,634	
DRY WEIGHT (LBS)	17,335	
PERSONNEL & PROVISIONS	2,415	
RESIDUALS	955	
LANDED WEIGHT (LBS)	20,705	
ADAPTER & LES	9,778	
PROPELLANTS & CONSUMABLES	4,067	
LAUNCH ESCAPE WEIGHT (LBS)	34,550	
ALS ADAPTER	6,870	
LAUNCH WEIGHT (LBS)	41,420	

The placement of the subsystems within the PLS vehicle are done so as to achieve the required center of mass location of 53% to 56% of vehicle body length at both the full and residual consumables conditions.

5.1.3 Launch Escape System and Booster Adapter

A launch escape system is provided to achieve assured crew safety over as much of the mission profile as possible. The system provides the capability to separate from the launch vehicle when on the launch pad or during ascent flight. The preferred system consists of three solid rocket motors attached externally to the conical booster adapter as shown in Figure 5-2.

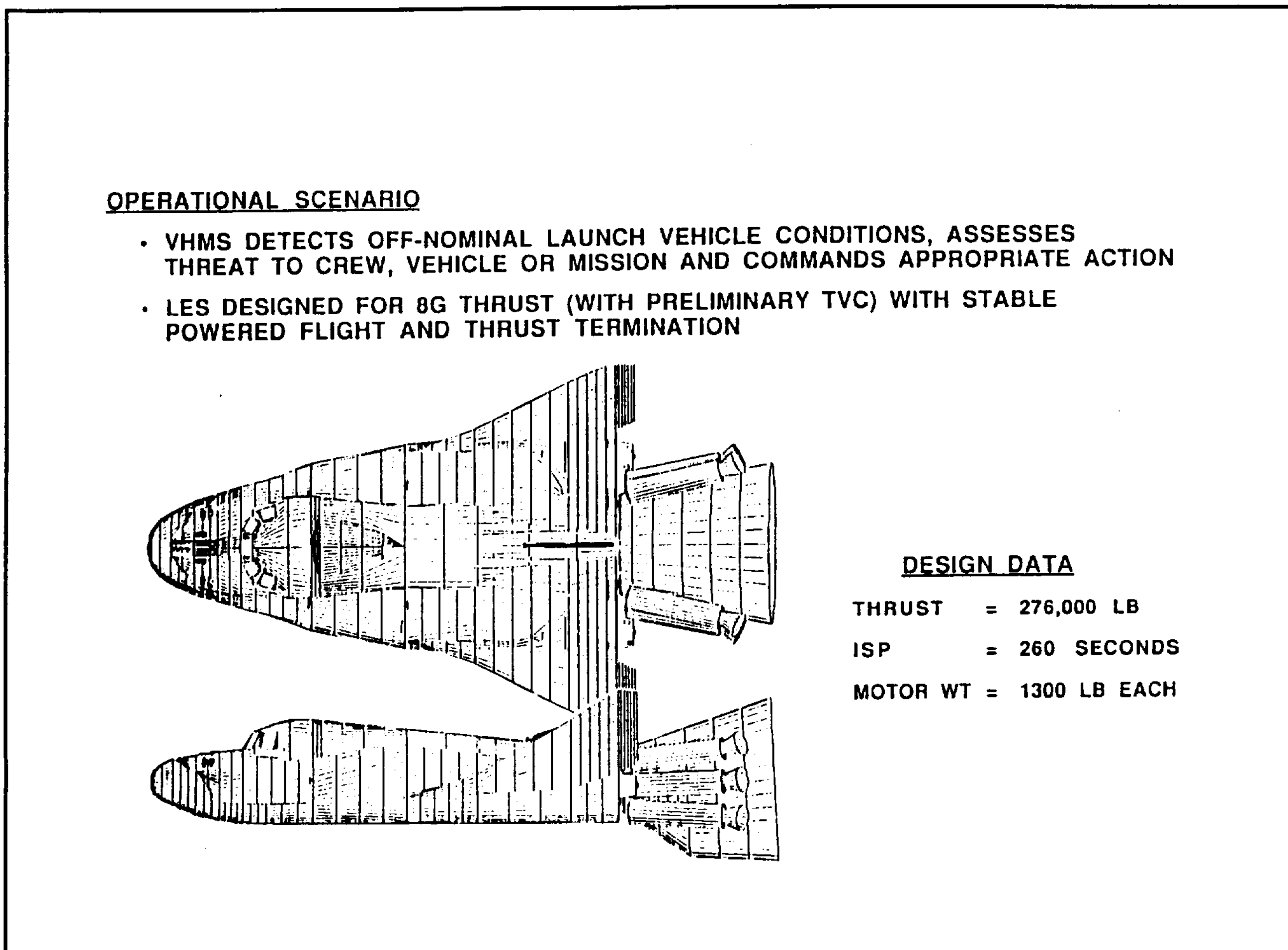


Figure 5-2. Launch Escape System

5.1.4 Wing, Center Fin and Control Surfaces

To satisfy the requirement for compatibility with the STS payload bay and for aft compartment maintenance access, the PLS wing fins must have a folding feature. The folding fins also enable transport in C-5 or C-17 cargo aircraft (Figure 5-3). The auto-flight guidance avionics system provides redundant control of the flight control surface actuators, with manual over-ride

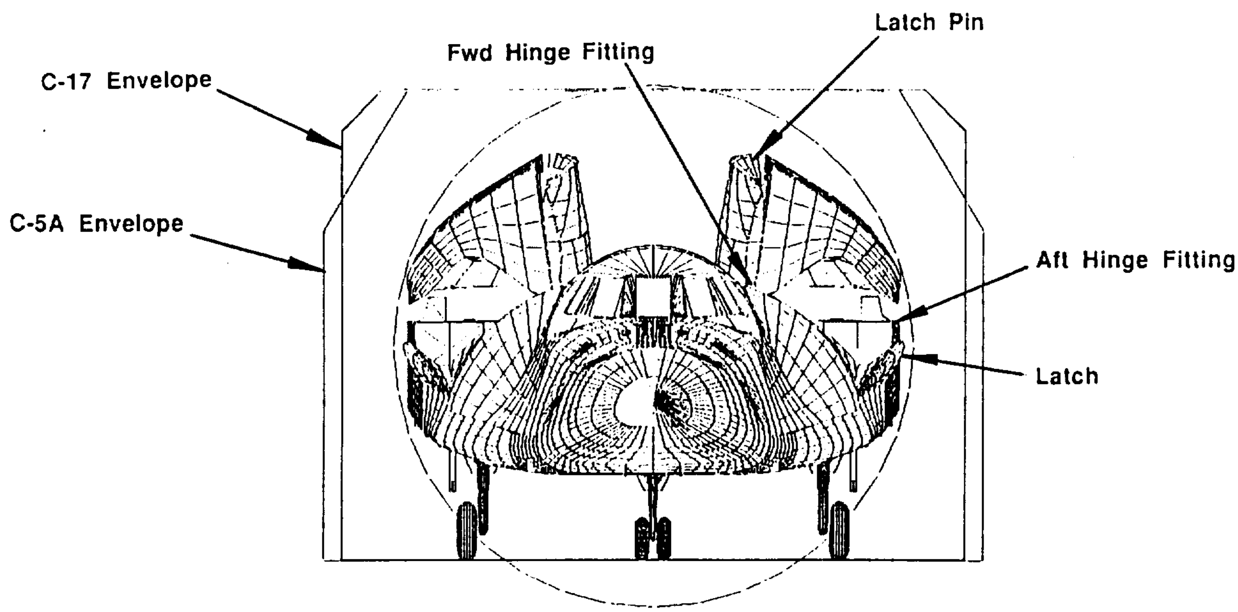


Figure 5-3. The Folding Wing Feature Permits Transport in Large Cargo Aircraft.

capability. The wing folding actuators are manually operated only: they use external power on the ground and internal power on-orbit. Only the autocontrolled actuators are paired in single redundancy with self monitoring/self healing control circuitry.

All actuators are low power electro/mechanical systems. The auto-flight guidance BITE test includes actuator BITE interface with the position indicators on the flight display panel. Position status data are also sent to the control master communication system computer for ground communication and command feedback, as well as to the onboard maintenance system (OMS) computer for maintenance update.

Structural and Mechanical Concept. The fin folding requirement is expanded in the PLS design to a geometry which adds fin root bending strength, more subsystem access and the possibility of improved aerodynamics. Rather than hinge the fin near the fin root along a line parallel to the vehicle centerline, the movable section of the fin also includes part of the fuselage upper/outboard skin (as shown in Figure 5-4,) , so that in rotating the fin to a stowed position a sizable subsystem access area is also opened. The hingeline of this concept runs outboard, aft and down rather than just aft so that 1) a triangular access area is revealed, 2) the fin folds to a compact, inboard and aft position

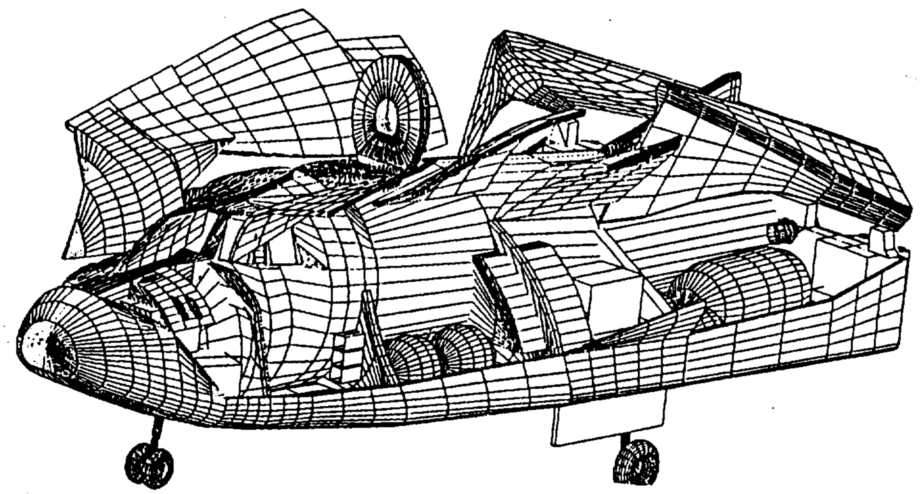


Figure 5-4. Folding Fin Arrangement

and 3) the upper elevon and pitch RCS thruster locations are not disturbed.

Two of the major frame extensions (from cabin pressure vessel) act as the carry-through structure for the fin. The aft carry-through frame (aft bulkhead) reacts the fin rear spar loads through the fin aft pivot fitting (Figure 5-4). The forward carry-through frame transfers the fin front spar loads to the cabin compartment via the fin latch. The forward-inboard fin pivot, which defines the diagonal fin hingeline, is located inboard at the frame/cabin attachment. Since the high fin root bending moment is reacted by the latch on one side and a large root fillet radius on the other, the design has better section properties than the short coupled, latch-and-hinge alternative. This can either lead to reduced weight or increased fin stiffness.

The design also has non-structural benefits. The large fillet radius in the fin root area may reduce interference drag between the fin and the adjacent upper fuselage surface. Because the motion of the fin and fuselage at joining is nearly translational, a simple labyrinth seal can be used rather than the more complicated rotational seal which would be required with a rotating-type fin folding concept.

The wing fin structure is a conventional dual-spar, multirib design using the same graphite polyimide materials as that of the heatshield since the TPS materials and method of attachment for the fin is the same. The difference in the wing application is that the lower wing surface uses the honeycomb material for moldline stability while the upper wing cover can be a single skin (multi-ply) for the upper cover based on TPS installation requirements and blast pressure reaction. The single skin is easier to form into the airfoil curvature and easier to install using blind fasteners. The two structural surfaces are separated by a conventional arrangement of bonded-in spars and frames of the same graphite polyimide material. For the ribs and spars, graphite polyimide sine-wave corrugations or a truss configuration for thermal stress reduction is assumed.

The static load conditions assumed for the wing design task are: liftoff, abort, Max q-alpha and Max q-beta, descent loads and TAEM maneuvering. The dynamic load conditions are: vibro-acoustic pressures, blast overpressures (transient excitation) and buffet (potential excitation due to unsteady aerodynamic flow). Aero-thermal temperatures are applied to the structure for ascent and descent heating environments. Figure 5-6 defines the governing blast overpressure characteristics.

The governing load case for the PLS wing fin design is the 10 psi blast overpressure occurring from a catastrophic booster explosion. This pressure will create a bending moment on each fin of approximately 1.4 million in-pounds. This moment must be reacted by the wing structure and transferred to the PLS cabin structure without failure if the vehicle is to be controlled for subsequent parachute deployment and recovery.

The cross-section, shown in Figure 5-7, is used to estimate a nominal stress level in the wing skins in reacting the 10 psi overpressure load. Given the average 4.8-inch structural depth and imposed bending moment, the stress level in the wing covers is estimated to be 28,000 psi which is significantly below the material limit of 44,000 psi at non-reentry temperatures.

The blast load reacted by the fin must be transferred to the cabin through the hinge and latch structural attachments. The fin folding concepts is such that the least loaded attachment is the latch. It is designed to react only lateral and vertical loads. The forward/inboard hinge is the highest loaded point because it transfers lateral and vertical loads as well as reacting all of the bending moment transferred forward by the fin through the torque box nature of the fin root design. The aft hinge point is nearly as highly loaded as the forward hinge and also reacts lateral and vertical wing loads.

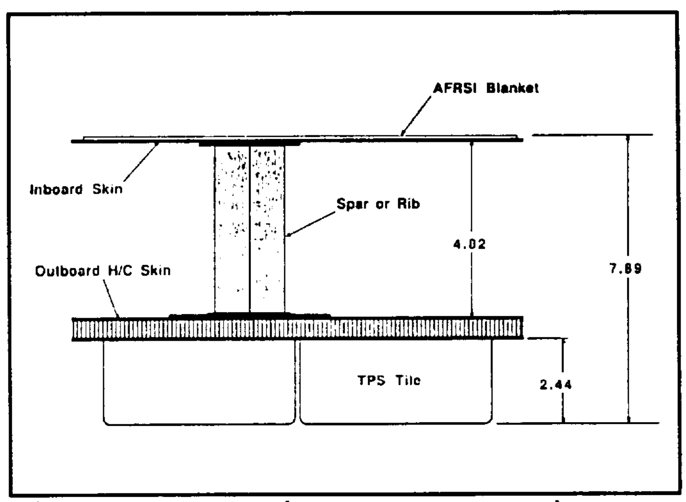


Figure 5-7. Wing Cross-section

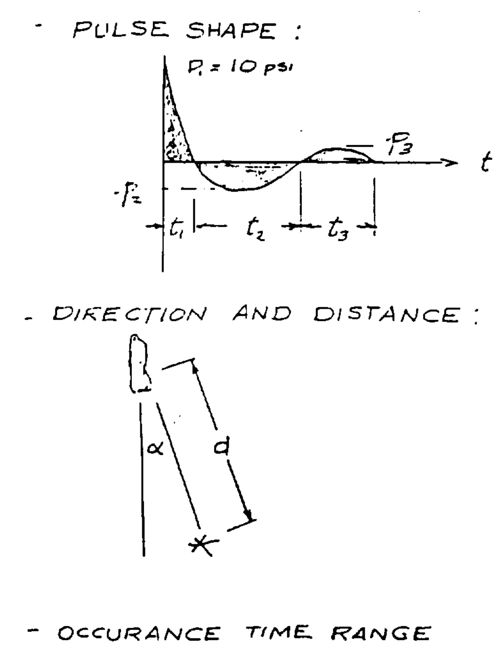


Figure 5-6. Explosive Overpressure Characteristics

The hinge or latch pin diameters were calculated based on the reactions at each of the three attach points. Assuming steel material and a factor of safety of 1.4, the diameters of the three attach points sized by the blast overpressure load case are 2.38, 2.12 and 1.0 inches for the forward hinge, aft hinge and latch pin diameters, respectively.

To summarize, based on the dimensions and thermal environment of the selected representative point on the fin surface, it is concluded that there is adequate depth between the wing moldline surfaces to accommodate the ceramic tile thickness required and enough remaining depth between the structural skins to react the extreme fin bending loads.

5.1.5 Primary Structure

The body group is defined as the crew cabin/primary structure including the extension frames which support the lower heatshield and define the subsystem bays plus the lower heatshield structure itself (Figure 5-8). A representative section of the crew cabin and a model of the heatshield structure were structurally analyzed for loads cases defined by PLS mission phases.

The crew cabin is based on a 76-inch diameter cylinder which is a fundamentally efficient pressure vessel shape. The cylinder is stiffened by 3-inch deep ring frames spaced every 17 inches and by six longerons. The basic cylinder shape is modified by the necessity of a flat floor in the cabin and a moldline slope of six

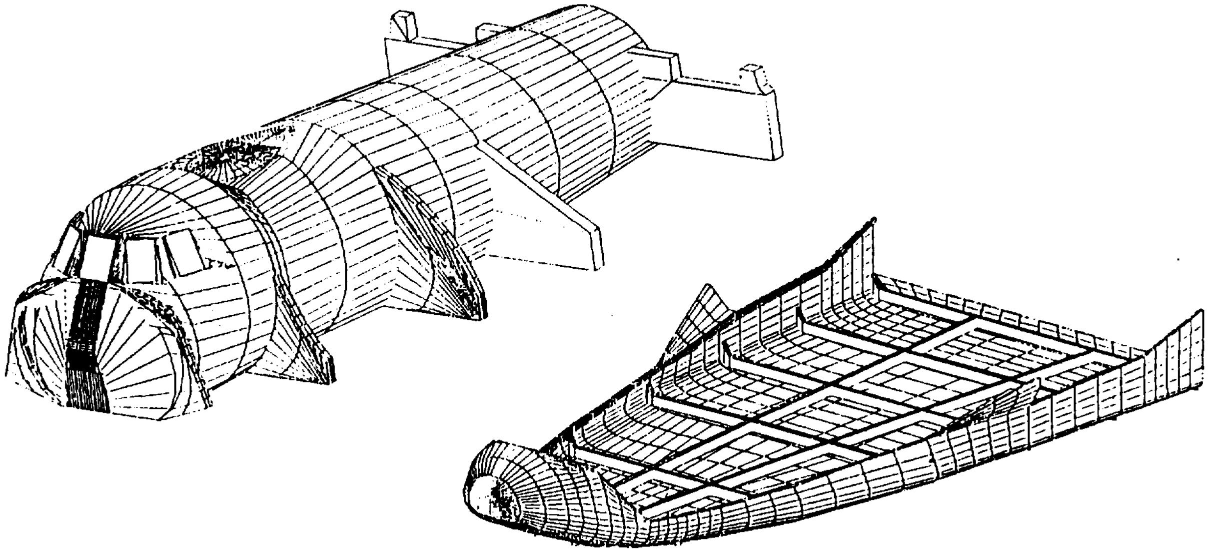


Figure 5-8. Primary Structures Are Crew Module and Heatshield

degrees going aft. The material (2219 aluminum) and welded construction of the crew cabin are very conservative with well established design life characteristics. The extension frames are placed to correspond with the location of the personnel seating structure inside the cabin. In this way both structural design elements work together as an efficient, integrated unit. The extension frames attach and react airloads from the lower heatshield and define the size of the side access panels.

The heatshield employs a direct-bond TPS tile concept that was investigated with a prototype demonstration in the CAST program sponsored by LaRC in the early 1980's. Directly bonding the tiles to a graphite polyimide honeycomb structure with a similar coefficient of thermal expansion yields weight savings of up to 30% compared the present Orbiter technique. Cost savings accrue from fewer manufacturing processes (including the elimination of the strain isolation pad) and larger tile sizes.

A typical PLS vehicle cross-section is shown in Figure 5-9 to identify the key structural features of the body group. The cylinder-based crew cabin dominates the cross-section with only a small portion at the top actually forming the outer moldline of the vehicle.

The beam elements of the design efficiently perform three functions. The portion of the cross beam outside the cabin reacts the aerodynamic heatshield loads as a bending moment across the vehicle. The carrythrough portion of this beam also provides the

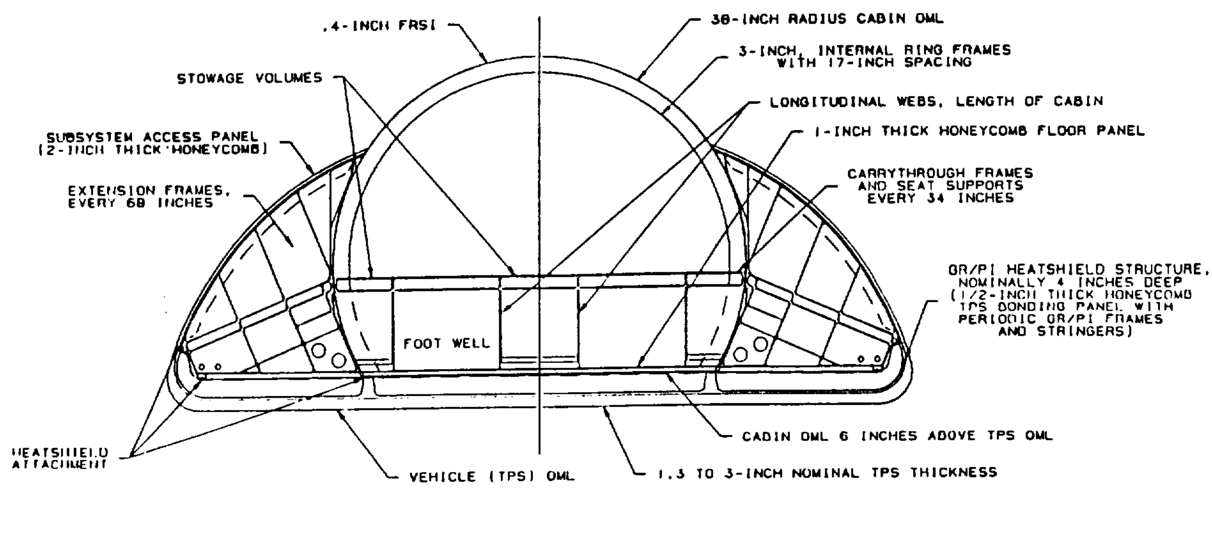


Figure 5-9. Typical Cross-section Identifies Key Structural Features

primary attach point for the passenger seats to react the high launch and abort loads of the personnel. Finally, the carrythrough portion works with the longitudinal dual keel to maintain the flatness of the floor under the one atmosphere internal pressure for launch (and abort).

Pressure Vessel/Crew Cabin Design. Because the vehicle structure is necessarily associated with nearly every other subsystem, its fundamental design approach affects the installation, operation and maintainability of every other system. Since the level of subsystem access was so important to the outcome of the efficient operations and low cost goals of the PLS program, the crew cabin design was given the highest priority as the primary structural element of the vehicle.

To approach the airline level of subsystem access in a vehicle as small as the PLS is challenging if standard spacecraft structural arrangements are adopted. Maintainability approaches are usually at a lower level of design such as a particular access door design, LRU rack or quick disconnect. To achieve believable levels of access at the vehicle level requires an accessible, maintainable design in the overall vehicle design, not just at the detail level.

The airlines perform major maintenance on an aircraft by first removing all access panels to completely reveal the subsystems. The PLS structural concept allows the same level of access to validate our claim of airline-like servicing during turnaround operations.

One structural approach to be taken is to use a near-moldline conformal pressure shell which is both primary structure and most of the crew cabin. As a result, the subsystems reside within the pressurized portion of the shell. Access to these systems is therefore from the inside, which suggests maintenance complexity and interference as well as concealment by seats, passenger stowage provisions and non-structural interior panels. The broad, flat PLS lower structure (as is conforms to the lower moldline) is susceptible to extreme deformation when pressurized. This effect can be reduced by using vertical tension ties installed across the vehicle interior, but these further hamper access to subsystems. From a manufacturing standpoint, the complex shape of the shell as a tight, welded pressure vessel, would not be easy to build. Non-pressurized cutouts in the shell would be required for the landing gear and the propulsion system which would involve complicated bulkheads, subsystem penetrations, and weight penalties.

Another structural approach is to employ a floating conformal pressure vessel inside the outer primary structure. This design has the same access-through-primary structure concerns as the former. In addition, this concept reduces the amount of volume usability by having this extra structural element. Additional concerns are the loss of interior volume and difficult inspectability created by the redundant structures (a compartment permanently within a structural shell), doors through two structures which may have relative motion, membranes which are difficult to penetrate for access and curved walls which reduce volume utilization. The principal advantage of the floating structure is that it is easier to thermally isolate.

With the evaluation of the two structural concepts described above providing guidance, the approach taken on the PLS structural arrangement is to incorporate features which support the low operations and manufacturing costs. The process begins by listing the characteristics of the structure which are preferred by the different disciplines. The list illustrates that the disciplines often have divergent desires. The objective of the Design function is to accomplish the best compromise possible in achieving the overall PLS program goals. These are a few of the characteristics desired by different disciplines:

Operations	"Inside-Out" vehicle (easy access) Non-structural doors Easily inspectable primary structure Rugged structure (no GSE protection)
Manufacturing	Flat or single-degree-of-curvature structure Minimize parts count and subassemblies Minimize materials requiring hand labor Specify conventional materials and joints
Structure	Simple shapes with few cutouts Advanced, high performance materials

The following paragraphs describe the PLS structural concept which addresses the design requirements of the separate interests involved. The specific design features which achieve the desired vehicle characteristics are:

1. As many subsystems as possible are kept outside of the pressurized volume. This eliminates either doors through primary structure or undesirable access from within the interior. This avoids the safety issue of the crew occupying the same volume as possibly hazardous subsystems. It also allows a simpler (non-OML) shape for the crew compartment.

2. Constant sections, straight elements and single degree of curvature are incorporated in the design of the primary structure. These features reduce manufacturing expenses by requiring simpler tooling, lower cost materials and fewer fabrication processes.

3. The largest access doors the vehicle geometry will allow are provided to provide access provisions approaching conventional aircraft.

4. A separate lower surface heatshield structure of graphite polyimide material and direct bonded ceramic TPS tile is used to allow complete inspection of the primary structure and to implement a thermal protection concept with demonstrated thermal and structural advantages.

Rather than inherit the weight, volume and access problems associated with the conformal configurations, the PLS design combines the functions of primary vehicle structure and pressure vessel into a single structure. The operations issues created by having most of the subsystems located within the body are avoided by the having the pressurized compartment sized only to meet the crew space requirements. This concept permits a much simpler compartment structure since a moldline no longer needs to be adhered to. By using constant, circular cross-sections in the crew cabin, both the design and manufacturing functions are supported. Since the subsystems are located outside of the pressure vessel, penetrations through primary structure are largely avoided. If a requirement for micrometeoroid protection arises,, a double shell concept can be designed much more easily without having to incorporate several doors or complex curvature panels.

The thermal advantage of a floating cabin structure (one with few heat shorts) is captured with a separate, airload-carrying heatshield suspended from the vehicle primary structure. This approach offers the option of using a higher temperature material than the conventional aluminum alloy of the primary structure, since the method of attachment allows differential expansion. The use of direct-bond tiles to a graphite polyimide heatshield has been demonstrated with an STS Orbiter body flap segment for the CAST program of the early 1980s. Since the composite material has a practical use temperature of 550 degrees F, thinner TPS tiles can be used and the pressure vessel can be slightly larger for given moldline constraints. Since the tiles and the composite have similar coefficients of thermal expansion, the direct bonding technique can be used to eliminate the strain isolation pads and carrier plates of other systems. The belly heatshield has stiffening features such as frames and stringers since it reacts the reentry airloads and transfers them to major frames extending from the crew cabin primary structure. Although doors in the heatshield for the landing gear cannot be avoided, the few heatshield attachments to the compartment and major frames are accessible from above, making the heatshield a fairly clean, monolithic structure.

The remaining moldline surface contours not defined by the compartment top or the monolithic belly heatshield, are created by large, secondary-structure doors supported by the major frames extending from the crew cabin. Located under these doors are virtually all of the vehicle subsystems. They are attached either to the exterior wall of the pressurized compartment or from the forward or aft sides of the major frames.

Since the vehicle is required to have a forward c.g. location for aerodynamic stability, the main landing gear is located forward and attached to the substantial forward wing carry-through frame. This not only provides a more desirable landing gear placement, but creates more volume in the rear for the propulsion system hardware. The gear is enclosed in a non-pressurized box which extends forward to the next major frame.

The structural efficiency of the flat cabin floor (whether honeycomb or machined plate) is compromised by the attachment of conventional aircraft seats with seat legs. Additionally each seat must withstand high bending loads between the seat cushion area and the seats attachment to the floor structure. The PLS design eliminates seat legs and attaches the forward edge of each seat to internal sections of the major frames. This accomplishes three design objectives; it simplifies the design of the cabin floor, it eliminates the weight of seat legs, it provides a longer, more tailorable seat vertical stroke for crash load attenuation, it allows the seat bottom to move upwards to permit a seat position for deconditioned passengers, it produces a very short load path between the seat passenger load and its introduction into a dual center keel structure.

The configuration selected employs a simple, cylinder-based crew cabin as the primary structure. This pressure vessel provides the volume necessary for adequate habitability for the short PLS mission. The remaining volume between the crew cabin and the outer moldline is devoted to subsystem installation. Because the crew cabin is the primary structure, large access panels can be provided above the external subsystem bays to maximize both manufacturing access and ground turnaround access. The heatshield, suspended from the primary structure by the extension frames is thermally isolated from the crew cabin. Prior to heatshield installation, the exposed structure provides subsystem access during

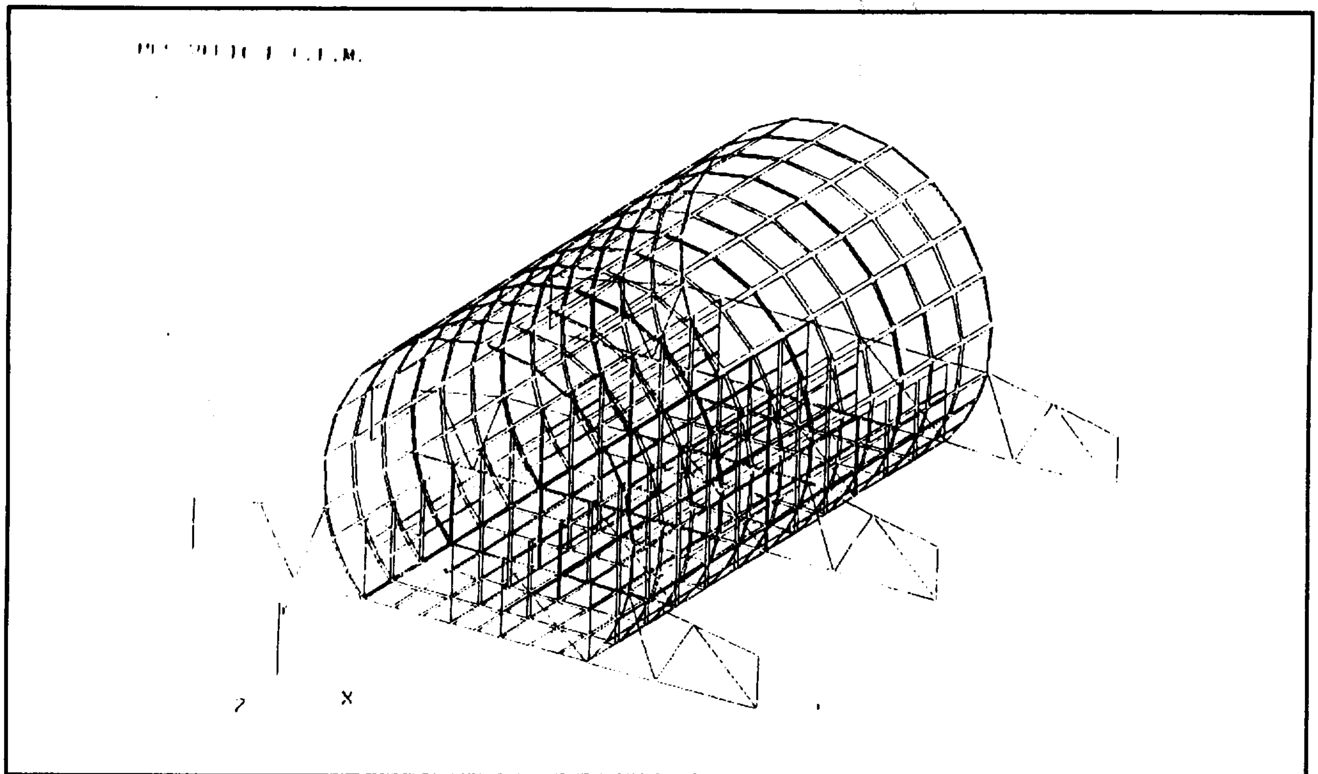


Figure 5-10. NASTRAN Finite Element Model of Crew Cabin

manufacturing and, if removed, permits complete pressure vessel structural inspection during major maintenance programs.

Pressure Vessel/Crew Cabin Analysis. A 3-D NASTRAN finite element model of the PLS vehicle was formulated with a representative stiffness for both in-plane and out of plane behavior. The model consists of 2-D plate elements for the pressure vessel shell, floor and longeron webs. Beam elements are used to model the lateral frames, external trusses that transfer the heatshield bending loads, and the pressure vessel ring frames. Figure 5-10 shows the finite element model for the PLS vehicle.

The pressure vessel is a 0.05 inch thick aluminum shell stiffened by 3 inch deep rings spaced every 17 inches. The cabin floor structure is a 1 inch thick aluminum honeycomb panel attached to the dual keel longeron elements and to the interior carrythrough portion of the exterior frames. The lateral frames that hold the seats for the crew are aluminum truss structures that provide stiffness against the loads carried through the structure by the external truss system.

The vehicle was analyzed for four loading conditions:

1.	Cabin Pressurization:	14.7 psi internal pressure.
2.	Overpressure Blast:	10 psi external pressure.
3.	Re-entry loads:	From the heatshield FEM results.
4.	Thrust loads:	8 G thrust environment.

The finite element model was loaded with 14.7 psi uniform internal pressure simulating the cabin pressurization loading condition. The resulting maximum deflections are:

1.	Floor	0.22 inches
2.	Longerons	0.18 inches
3.	Shell	0.07 inches
4.	Ring Frames	0.015 inches

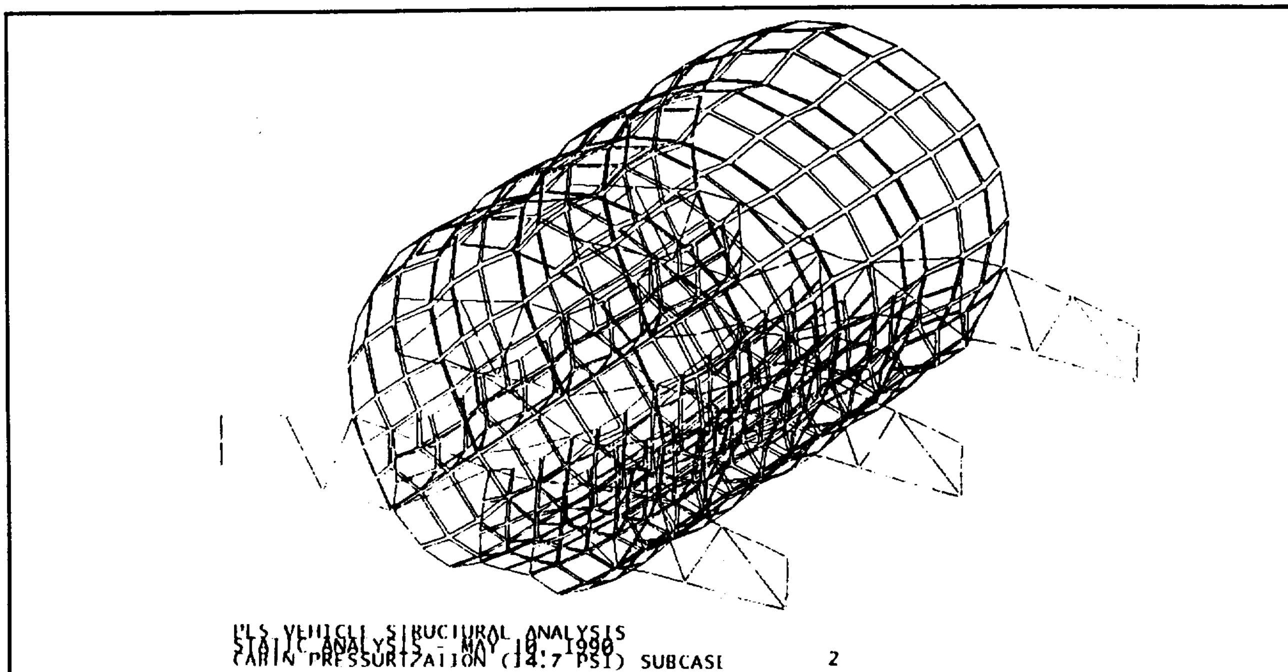


Figure 5-11. NASTRAN Model Analysis - Cabin Pressurization (Deformed Shape)

Figures 5-11 and 5-12 present the deformed shape and the displacement contours for the cabin pressurization load case. Note in Figure 5-11 the effect of the rings on the deformed pattern. Figure 5-13 presents the stress contours for internal pressure load case.

The principal stresses are as follows:

1.	Floor	8000 psi
2.	Longerons	13,000 psi
3.	Shell	10,000 psi
4.	Rings	13,000 psi

For the blast overpressure load case, a dynamic forcing function is modeled as a 10 psi uniform external static load. A dynamic load factor of 2.0 was used for this load case. Structural deformations due to the blast overpressure are as follows:

1.	Floor	negligible
2.	Longerons	negligible
3.	Shell	0.1 inches
4.	Rings	0.012 inches

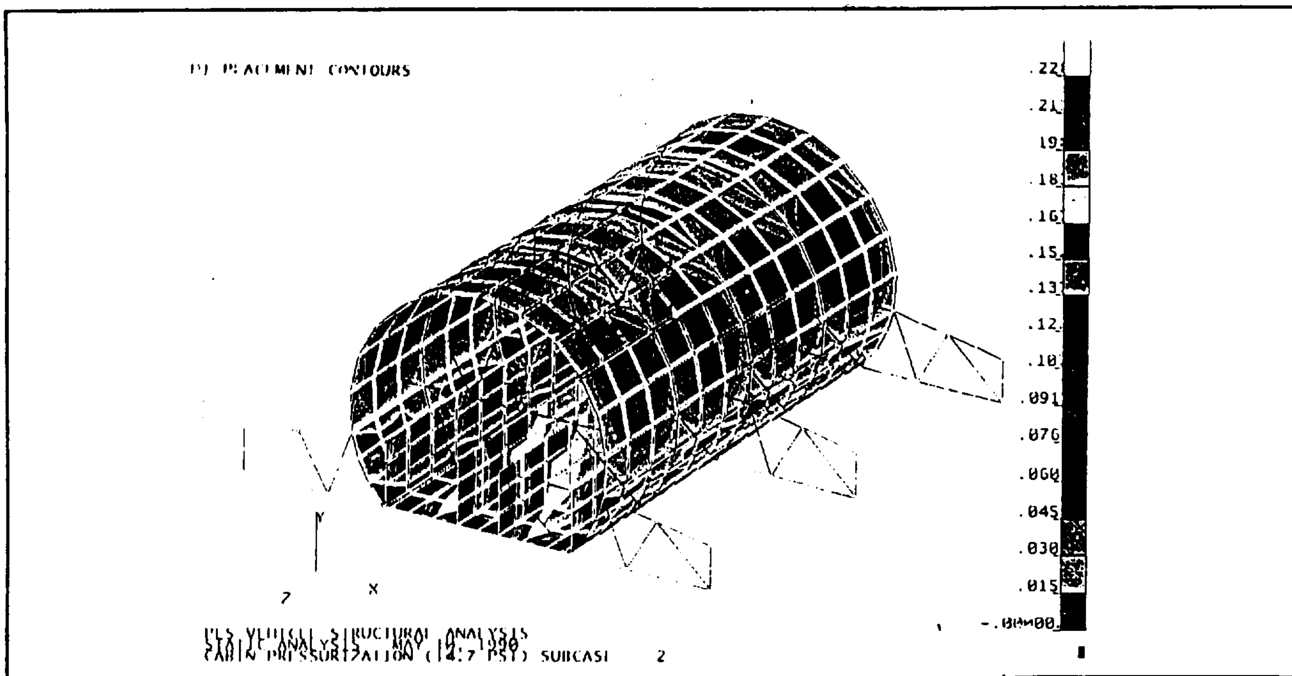


Figure 5-12. **NASTRAN Model - Cabin Pressurization (Displacement Contours)**

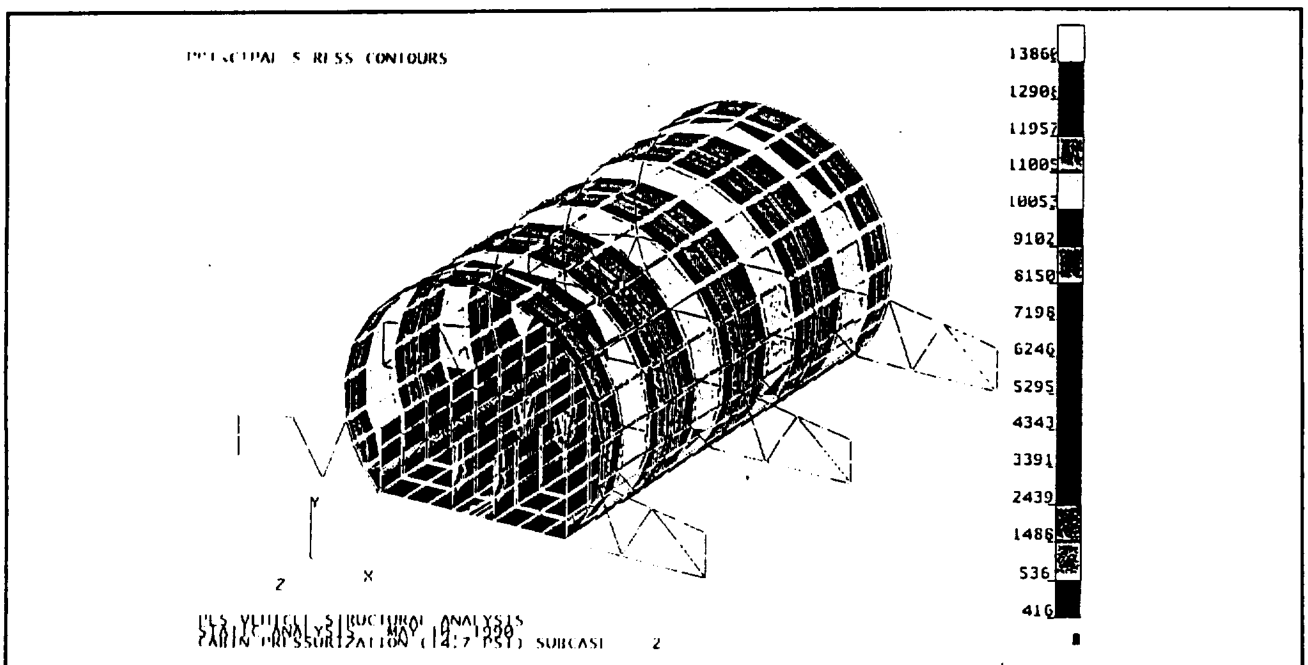


Figure 5-13. **NASTRAN Model - Crew Cabin - Internal Pressure Loads**

The deformed shape and displacement contours for the overpressure load are presented in Figures 5-14 and 5-15, respectively. The stresses on the shell can reach levels as high as 3400 psi in compression (Figure 5-16). However, note that due to the external loading on the shell, stability considerations will probably govern the failure mode of this load case. Preliminary hand calculations yield possible buckling of the shell in the areas

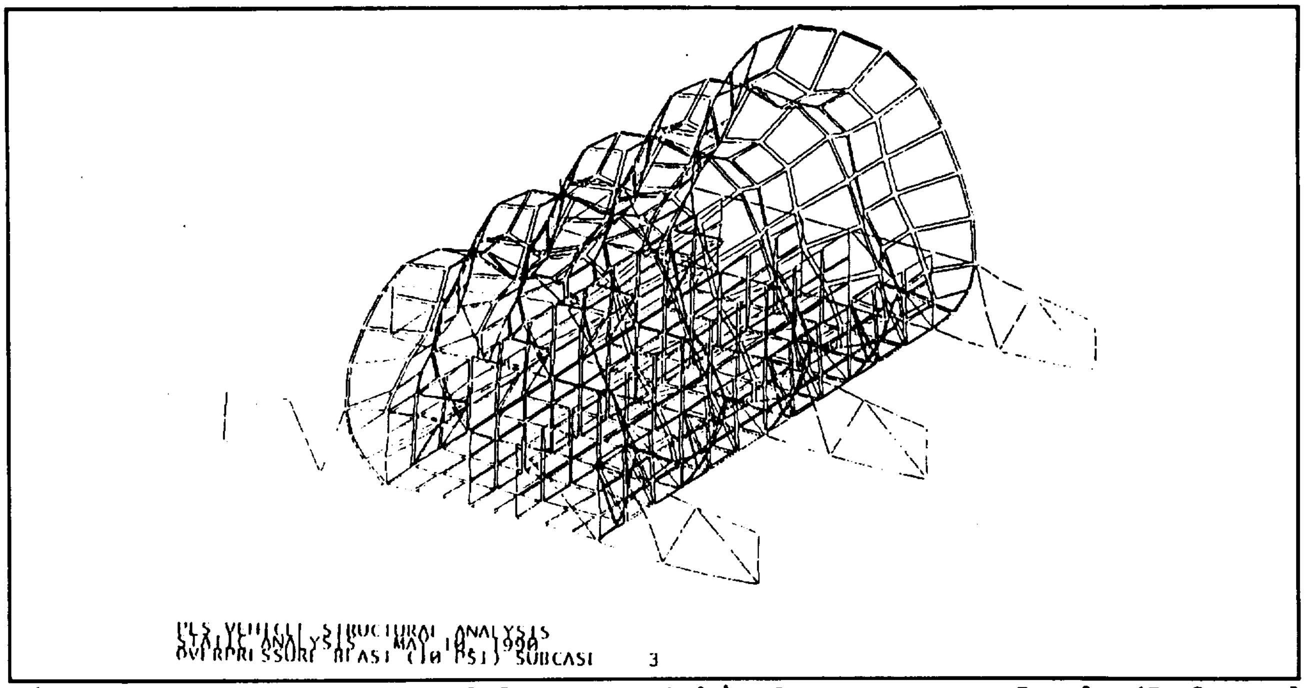


Figure 5-14. NASTRAN Model - Crew Cabin Overpressure Loads (Deformed Shape)

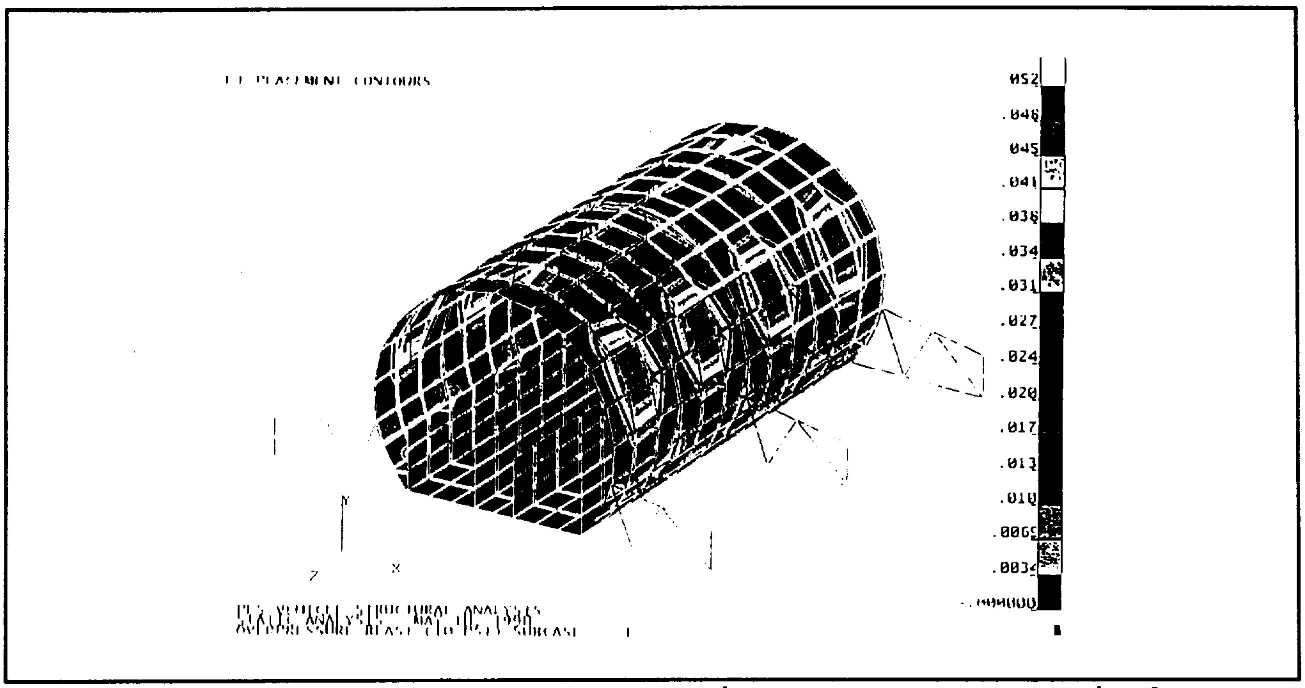


Figure 5-15. NASTRAN Model - Crew Cabin Overpressure - (Displacement Contours)

between the rings. For more exact results, the buckling eigenvalue problem has to be taken into consideration.

For the re-entry load case, the heatshield reactions (from the preceding heatshield analysis) are imposed on the extension frames of the vehicle. Truss tip deflections as high as 0.12 inches are

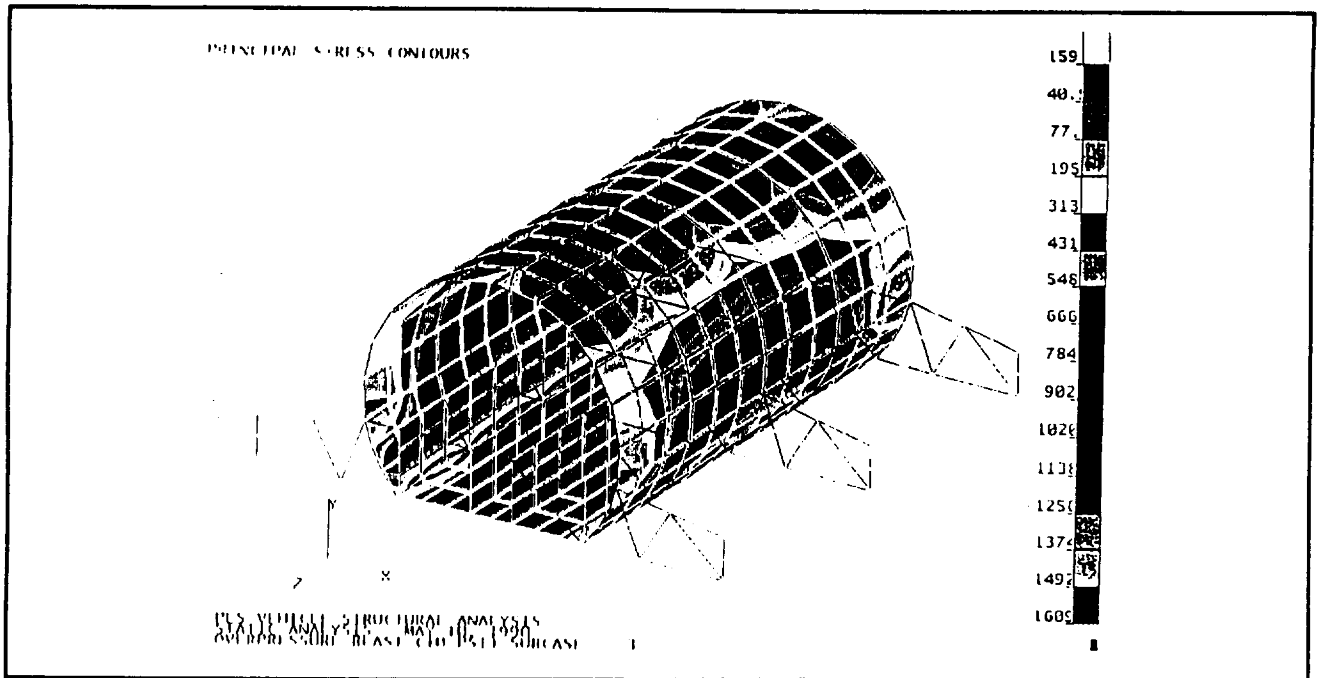


Figure 5-16. NASTRAN Model - Crew Cabin Overpressure - Compression

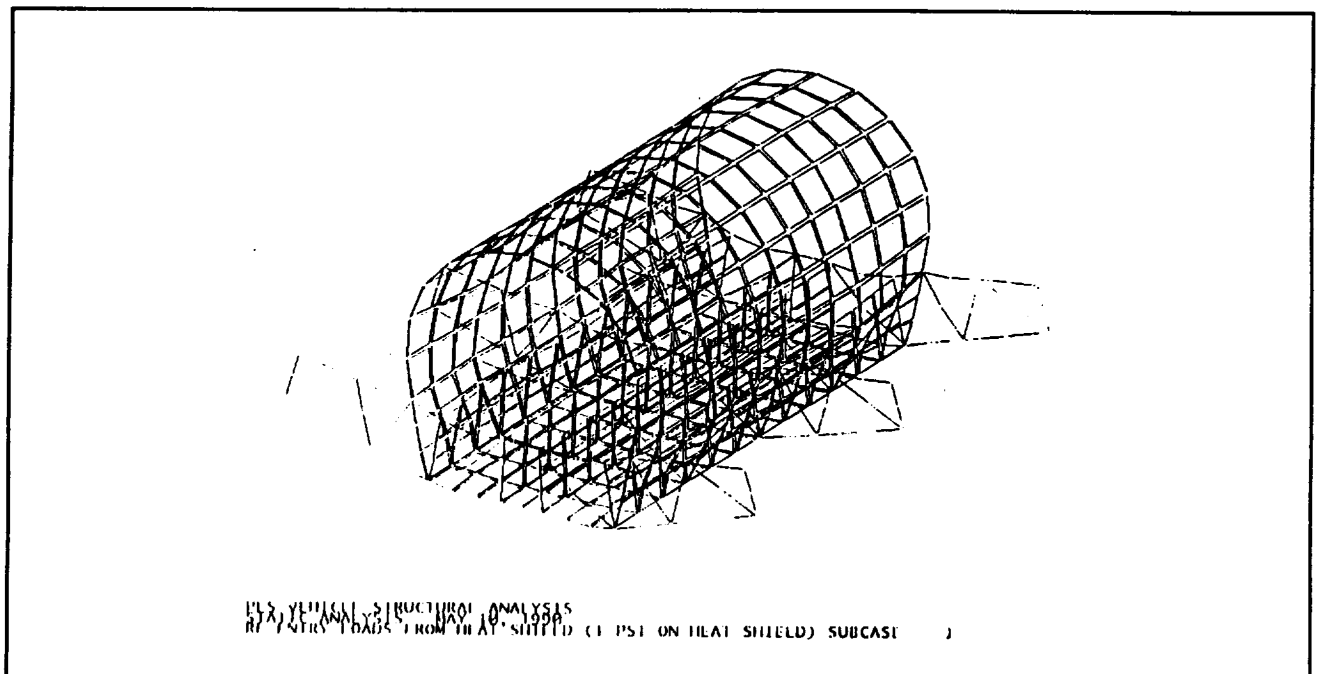


Figure 5-17. NASTRAN Model - Crew Cabin Reentry Loads (Deformed Shape)

observed. The remaining structural deflections are small.

Figures 5-17 and 5-18 present the deformed shape and the displacement contours for the vehicle, respectively. Note the interaction between the external truss assembly and the internal load carrying members. Stress contours are shown on Figure 5-19.

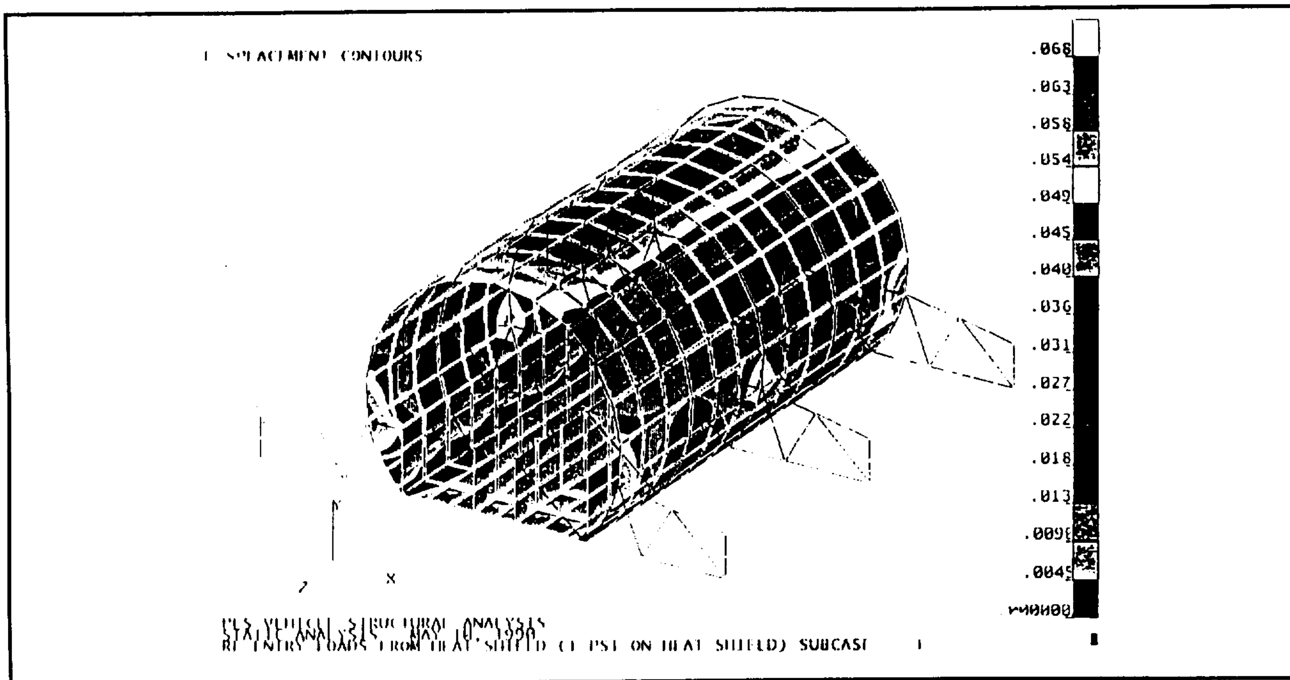


Figure 5-18. NASTRAN Model - Crew Cabin Reentry Loads (Displacement Contours)

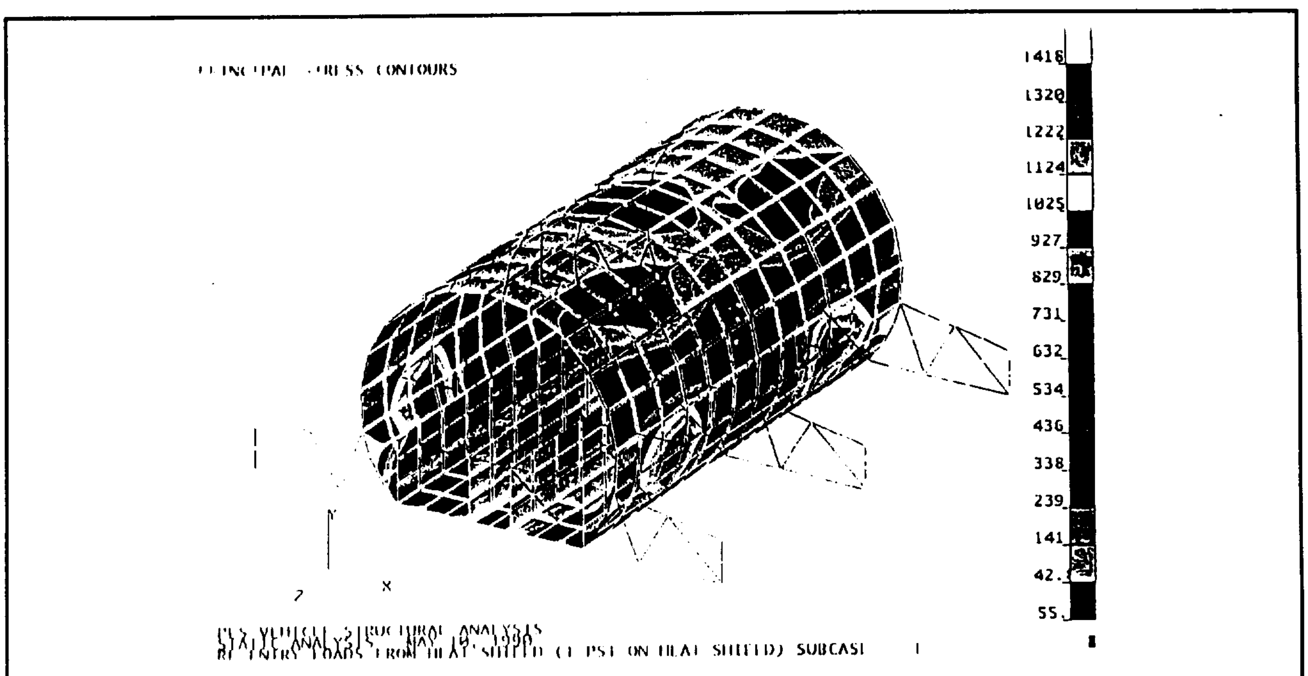


Figure 5-19. NASTRAN Model - Crew Cabin Reentry Loads (Stress Contours)

For the 8 G abort thrust load case, due to the difference between the locus of the centroid and the c.g. of the vehicle, the thrust load imposes a bending moment on the structure with the primary load carrying members being the longerons. A stability analysis will eventually be required since lateral buckling of the longerons is possible. Generally, the deflections from the abort thrust are small. Stresses reach a maximum of about 5000 psi at the fixed (aft) end.

This preliminary analysis of the primary structure (lower heatshield and pressure vessel) shows that the proposed approach of separating the structural functions of the airload-bearing heatshield and the core cabin structure is fundamentally sound. A complete vehicle, detailed stress analysis will be required in a later design once a more detailed structural definition is available.

Suspended Heatshield Design. The structural concept for the suspended lower heatshield is half-inch thick graphite/polyimide honeycomb. It is stiffened with upstanding honeycomb frames and longerons located to coincide with the heatshield attachments to the pressure vessel. This concept is one of many which could be employed on the PLS suspended heatshield structure. The following discussion addresses two of these alternates and compares the performance and manufacturing characteristics of each relative to the composite honeycomb.

Skin-Stringer Heatshield Option. This alternate is offered because of a concern over the bondline strength of the graphite/polyimide face sheets and the honeycomb core due to the high operational temperature of the materials and the lack of experience with the nominal graphite/polyimide honeycomb design.

This concept consists of a thick graphite/polyimide layup for the lower tile bonding surface with a number of stiffeners to achieve the required resistance to deflection. Since the heatshield design is driven by high stiffness requirements, the thick composite layup must be substantial to produce the same resistance to bending as the highly efficient reference honeycomb. Specifically, the layup must be about 0.39 inches thick to match the half-inch thick honeycomb. This value translates to a 300% increase in total heatshield weight with the same number and location of stiffeners as the reference. Because this large thickness raises concerns over the creation of voids in the layup, a modified thickness would be on the order .09 inches. This dimension, however, would mean that substantially more stiffeners would be required to meet overall stiffness. Rather than stiffening frames occurring every 68 inches as on the reference, the single, .09 skin of the alternate would need stiffeners roughly every 18 inches in both lateral and fore/aft directions.

Since the tooling for the stiffeners of either concept requires the design and fabrication separate tooling elements to support the upright stiffeners, the alternate concept needs on the order of 60 times as many tooling pieces. With the increase in the number of stiffeners, the final heatshield has roughly 20 times the number of inside bays or pockets as the reference. Therefore, the number of hand-bagged internal insulation packages to install on the heatshield is also 20 times as many.

The added weight, increased part count for both tooling and the final heatshield and the much larger insulation installation costs of the alternate skin-stringer design, leads to the contention that the reference honeycomb design is preferred from both manufacturing costs and performance considerations.

Isogrid Heatshield Option. The isogrid alternative to the reference resulted from the desire to install tiles on carrier plates for subsequent fastener attachment to the nodes of the isogrid. Open isogrid also suggests the possibility of subsystem access. However, it can be shown that the ideal isogrid size for the loads on the PLS heatshield is too small to provide a practical level of access.

A few concepts for making isogrid structure from composite materials have been put forth, but from a technology readiness standpoint, this isogrid is assumed to be produced by machining a thick aluminum plate. The isogrid is most competitive with, or superior to other structural concepts when the loading is complex and especially when substantial torsional loads are applied. Since PLS heatshield is not primary structure in the reference configuration, only bending, small shear and no torsional loads are imposed on the heatshield. The isogrid is not loaded to is full potential and is therefore less efficient than more simply-machined shapes like a waffle pattern. Because the aluminum material has a lower service temperature limit than the reference graphite/polyimide, the amount of TPS must increase correspondingly. Although the mechanically-fastened tile carrier plate concept is easier to remove and replace than the reference bonded tile design, the added initial cost over the entire heatshield is not believed to be offset by the potential savings in a localized repair situation.

The same conclusion is reached with the isogrid option as with the skin-stringer heatshield design: the reference composite honeycomb is recommended as the preferred concept.

The selected PLS heatshield is a graphite polyimide composite structure stiffened by an array of lateral and longitudinal graphite stiffeners. Graphite polyimide honeycomb is used to separate the composite facesheets which increases the overall stiffness and stability of the cross sections. Thermal tiles are bonded to the heatshield in order to insulate the structure against 2000 degreesF re-entry temperatures. The significant thermal gradient provided by the tiling system results in temperatures not greater than 600 degreesF at the tile-to-composite skin interface.

The driving requirement for the heatshield analysis is a direct function of the state of stress within the thermal tiles since the ceramic tiles are inherently brittle. Analytical and experimental methods have shown that if the flat supporting structure deforms such that the local radius of curvature is less 300 inches, the resulting stress in the tile will exceed the ultimate and the tile will probably crack. Therefore, the governing requirement for the heat shield design is to maintain radii of curvature above 300 inches throughout the structure. (Note: This design requirement refers to a flat surface tile installation. The more fundamental requirement is to limit the strain exposure on all tiles. This, then, also applies to tiles which are initially machined and installed on curved surfaces.)

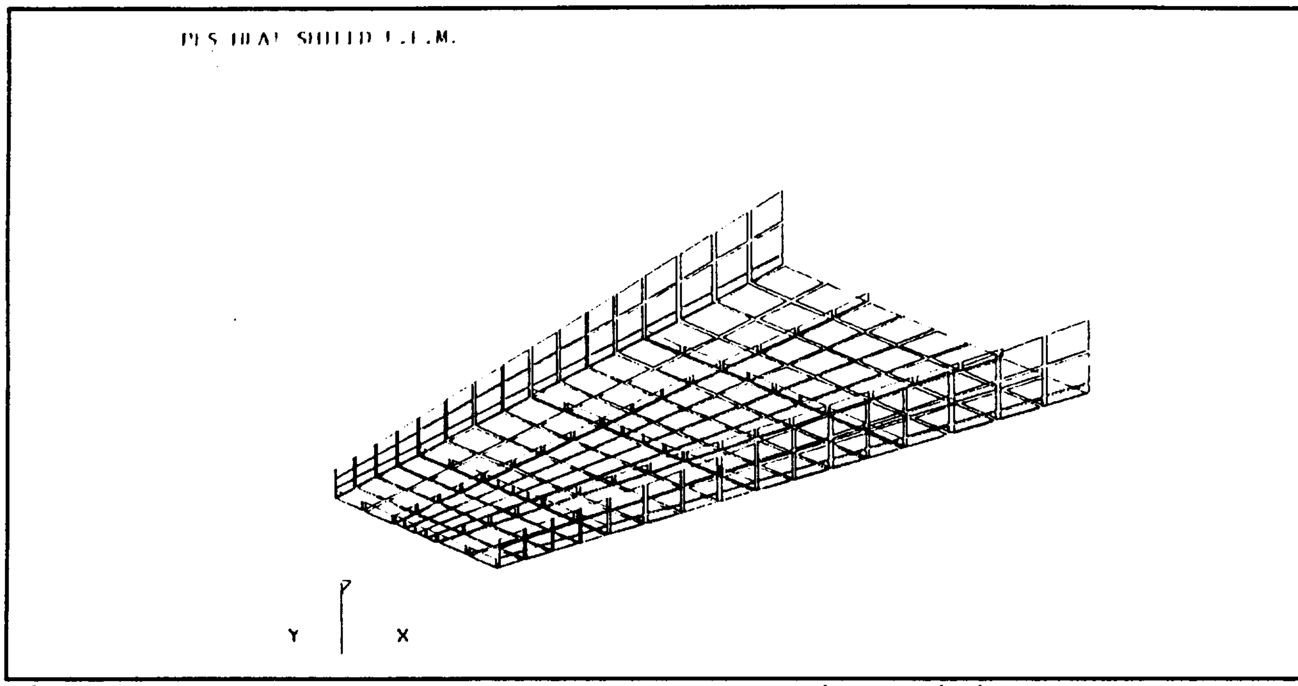


Figure 5-20. NASTRAN Model - Lower Heatshield Finite Element Model

Figure 5-20 presents the finite element model formulated to analyze the heatshield. The geometry has been idealized by linearizing the heatshield's curved taper. All other simplifying assumptions are made such that the heatshield's structural behavior is not altered. The boundary conditions are the joint connections that attach the heatshield to the vehicle interface (extension frames). The plates have been modeled such that the facesheets provide stiffness against membrane and bending loads while the honeycomb provides stiffness against shear loads. Half-inch polyimide honeycomb along with 8-ply, symmetric graphite facesheets were used in the analysis.

The loading conditions are as follows:

1. One psi uniformly, distributed re-entry pressure load.

2. 600 degreeF uniform re-entry temperature distribution with the provision that in-plane expansion of the heatshield is allowed. (Due to the joint designs, the resulting internal loads are negligible).

The 1 psi load case is due to the 3 G re-entry acceleration loading. The actual loads have been approximated as a uniformly distributed static load. Due to the nature of the load versus time, the static approximation is adequate. A factor of safety of 1.4 is applied to the results from this load case.

Subdividing the heatshield into four bays, each bay separated by a lateral stiffener and numbered 1 to 4, Bay #1 being the forward bay, the deflections due to the 1 psi loading are as follows:

Bay #1 - 0.005 inches
Bay #2 - 0.15 inches
Bay #3 - 0.50 inches
Bay #4 - 0.979 inches

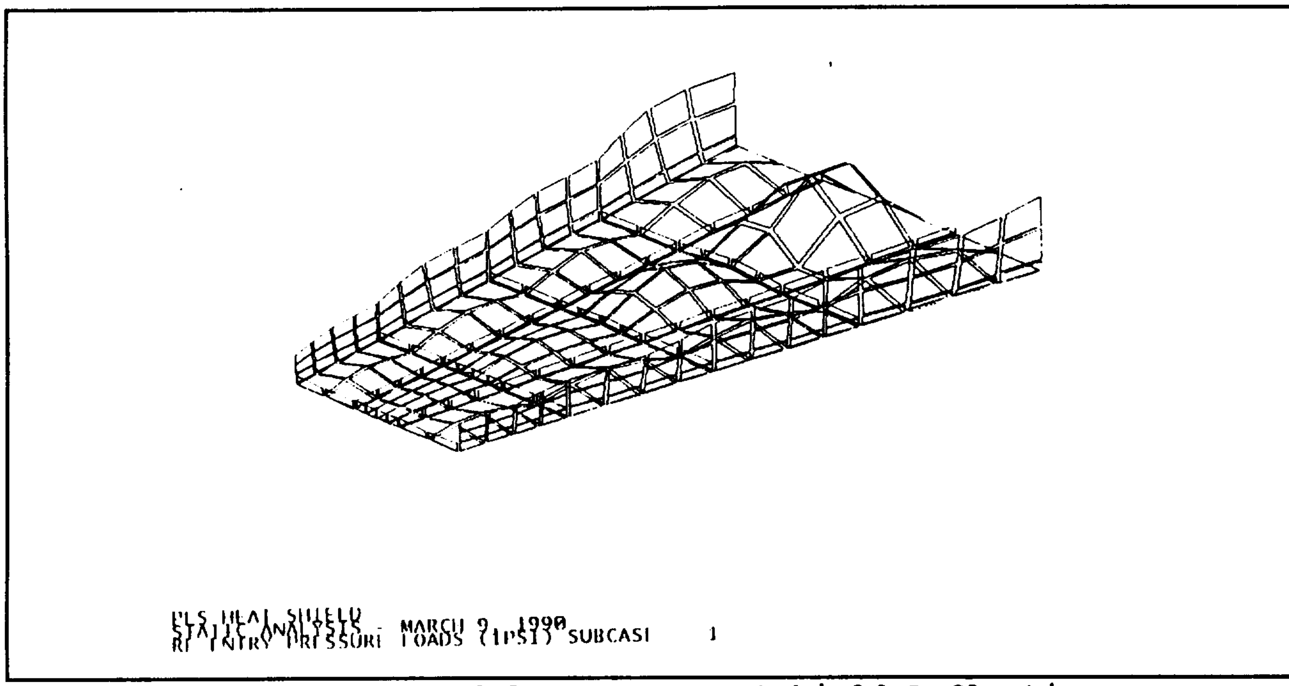


Figure 5-21. NASTRAN Model - Lower Heatshield Deflections

The deflected pattern and a detailed displacement contour are presented in Figures 5-21 and 5-22, respectively. The principal normal stresses vary in the following manner:

Bay #1 - 500 psi
Bay #2 - 1700 psi
Bay #3 - 4000 psi
Bay #4 - 5300 psi

Figures 5-23 and 5-24 present the normal and shear stresses transformed to the principal axes. It is apparent that the heat shield does possess adequate strength; however, the stiffness is not adequate as seen from the relatively large deflections occurring in Bay #4. From plate theory, using moment-curvature

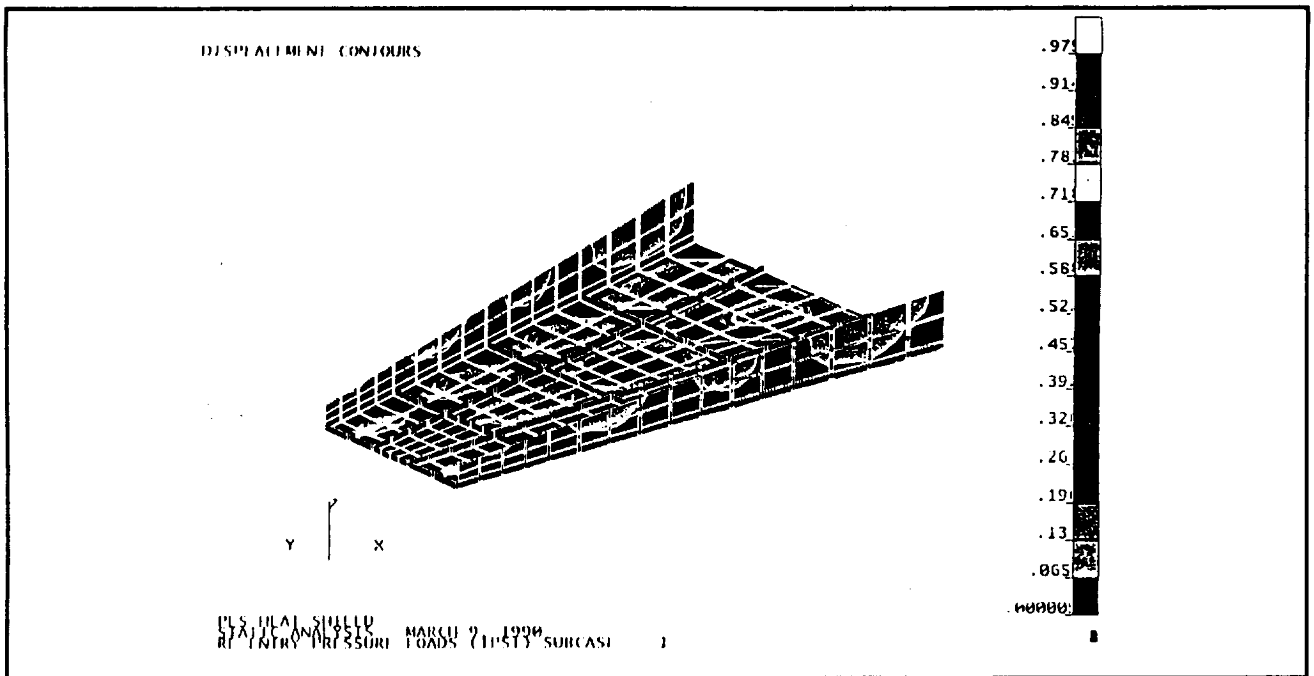


Figure 5-22. NASTRAN Model - Lower Heatshield - Displacement Contours

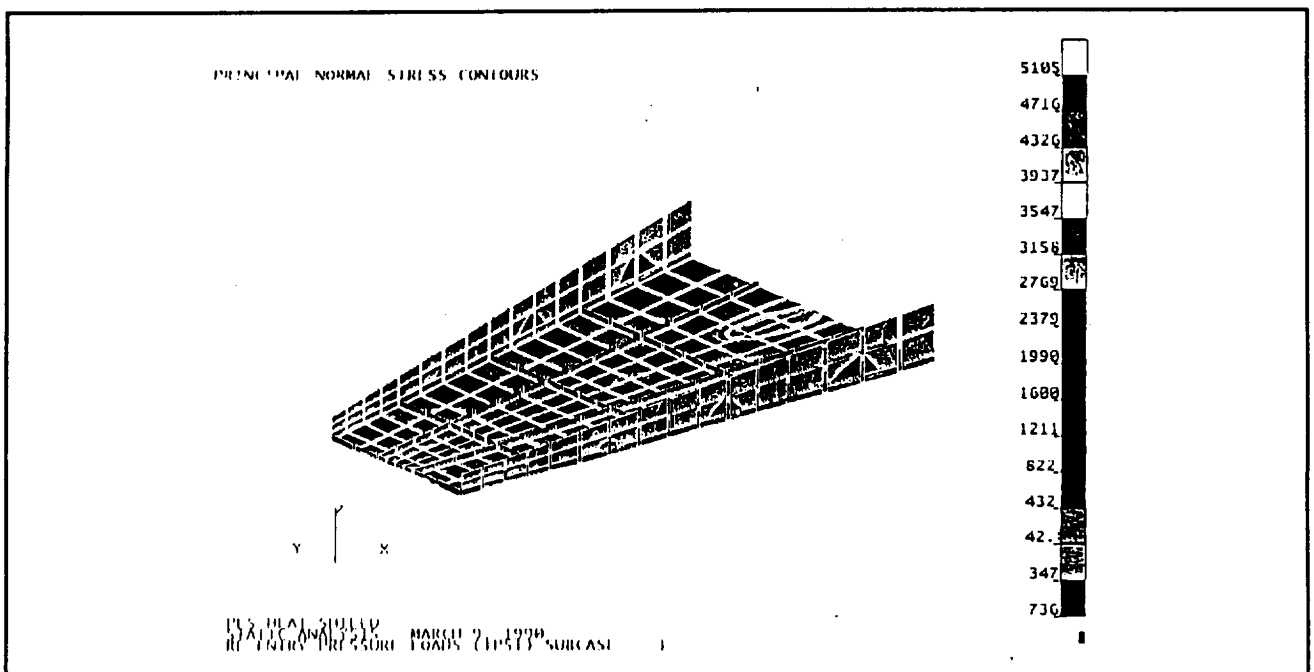


Figure 5-23. NASTRAN Model - Lower Heatshield - Normal Stresses

relations, the mid-span radius of curvature at bay #4 is approximately 400 inches which meets the 300 inch minimum criteria. However, the results can be improved by adding another lateral stiffener in the middle of Bay #4.

Figure 5-25 shows the heatshield finite element model with the additional stiffener. The new deflected pattern and displacement

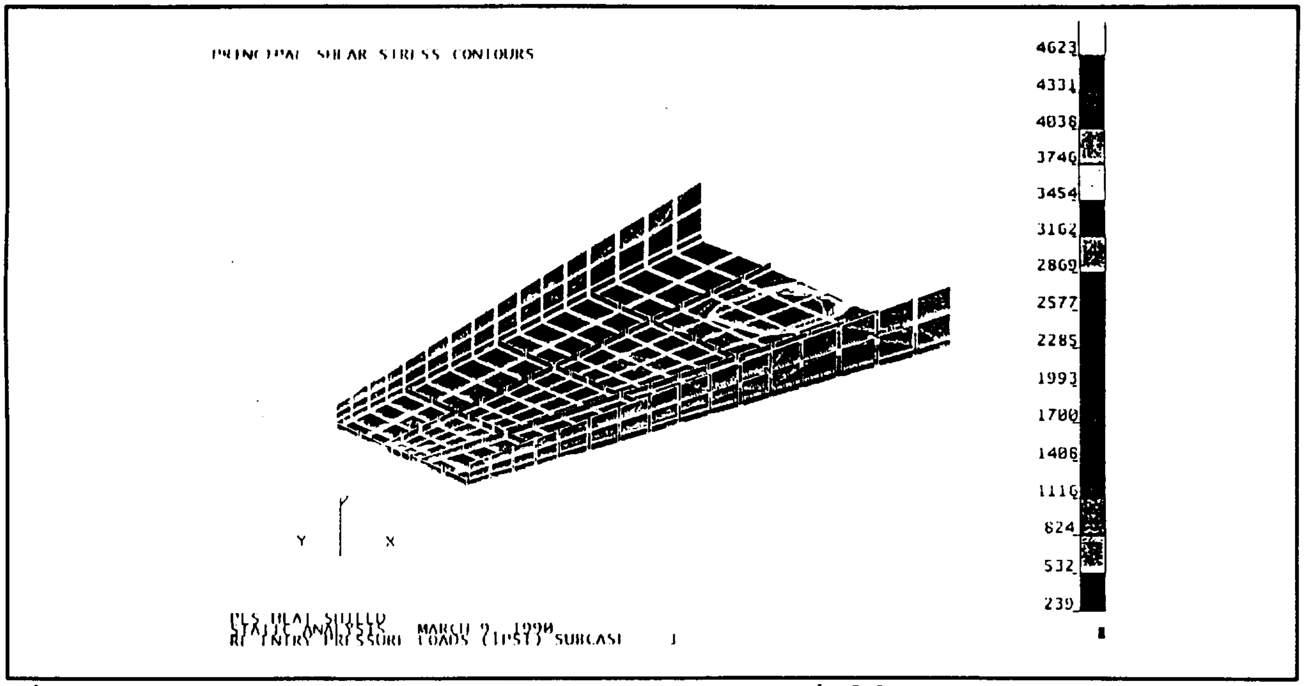


Figure 5-24. NASTRAN Model - Lower Heatshield - Shear Stresses

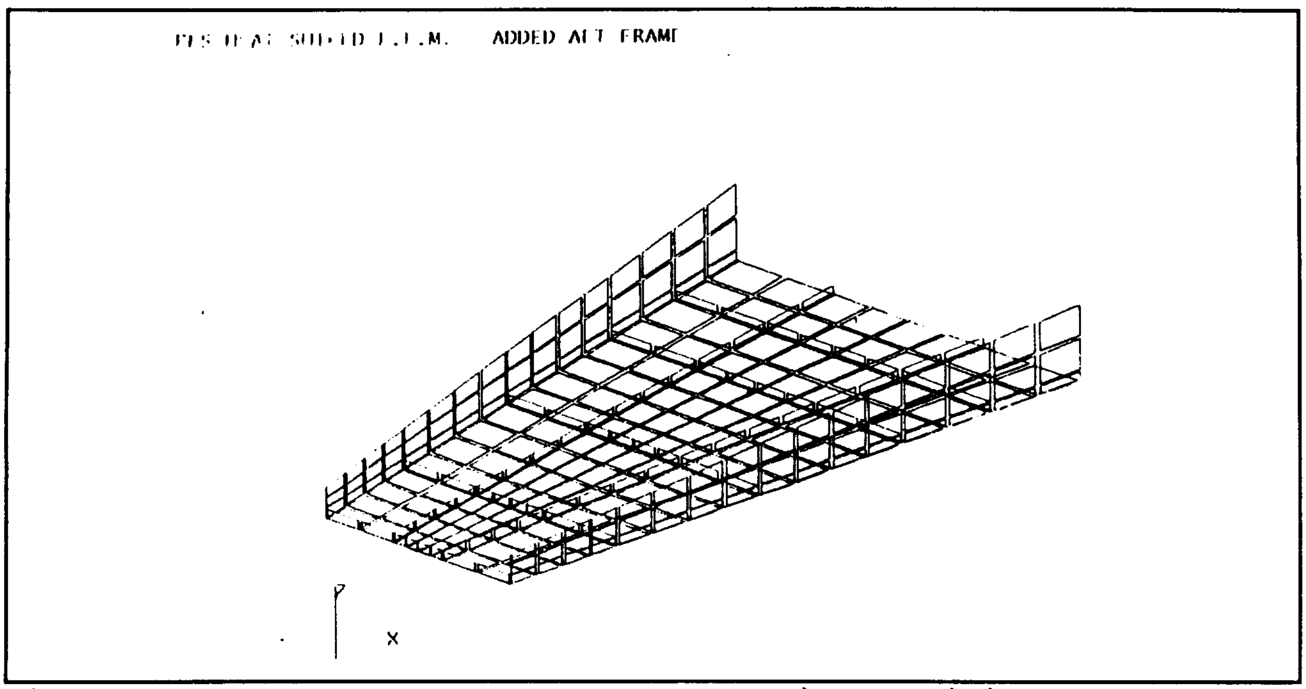


Figure 5-25. NASTRAN Model - Lower Heatshield - Finite Element Model with Additional Stiffener

contours are presented in Figures 5-26 and 5-27, respectively. The maximum deflections in the aft bay are reduced to 0.484 inches. The corresponding radius of curvature is 1100 inches which provides a large margin against tile fracture.

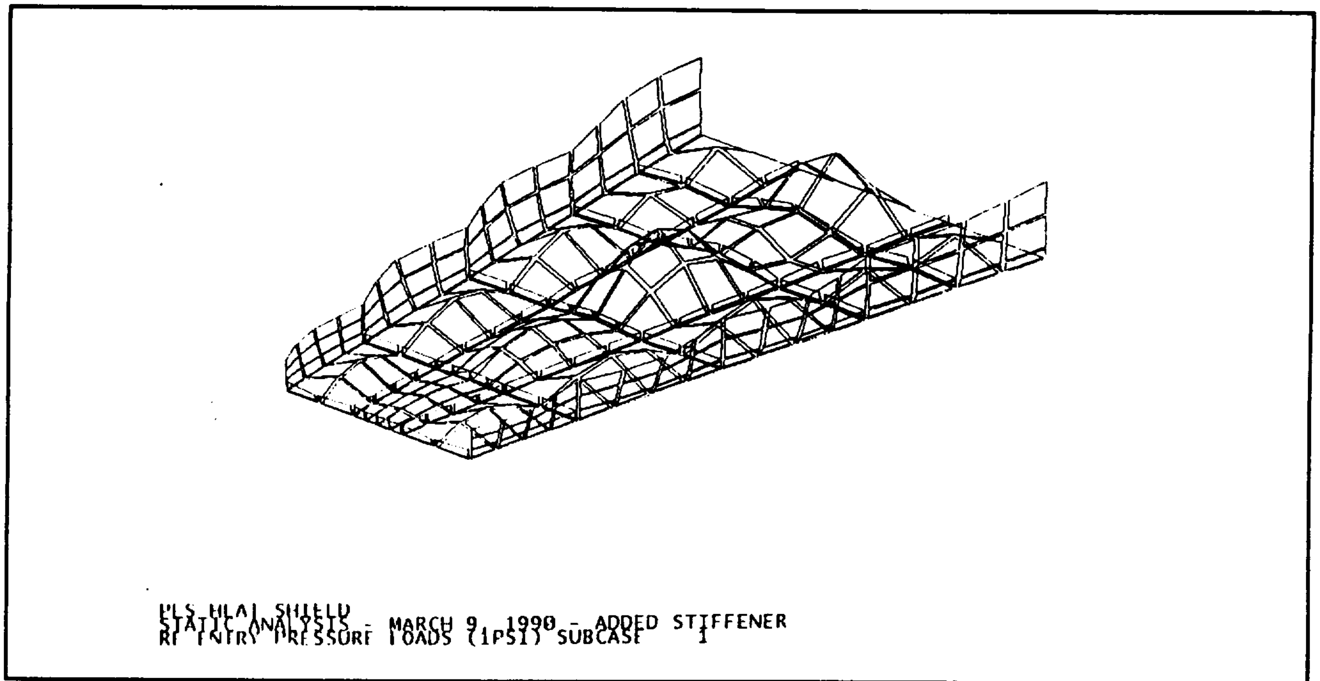


Figure 5-26. NASTRAN Model - Lower Heatshield Deflections

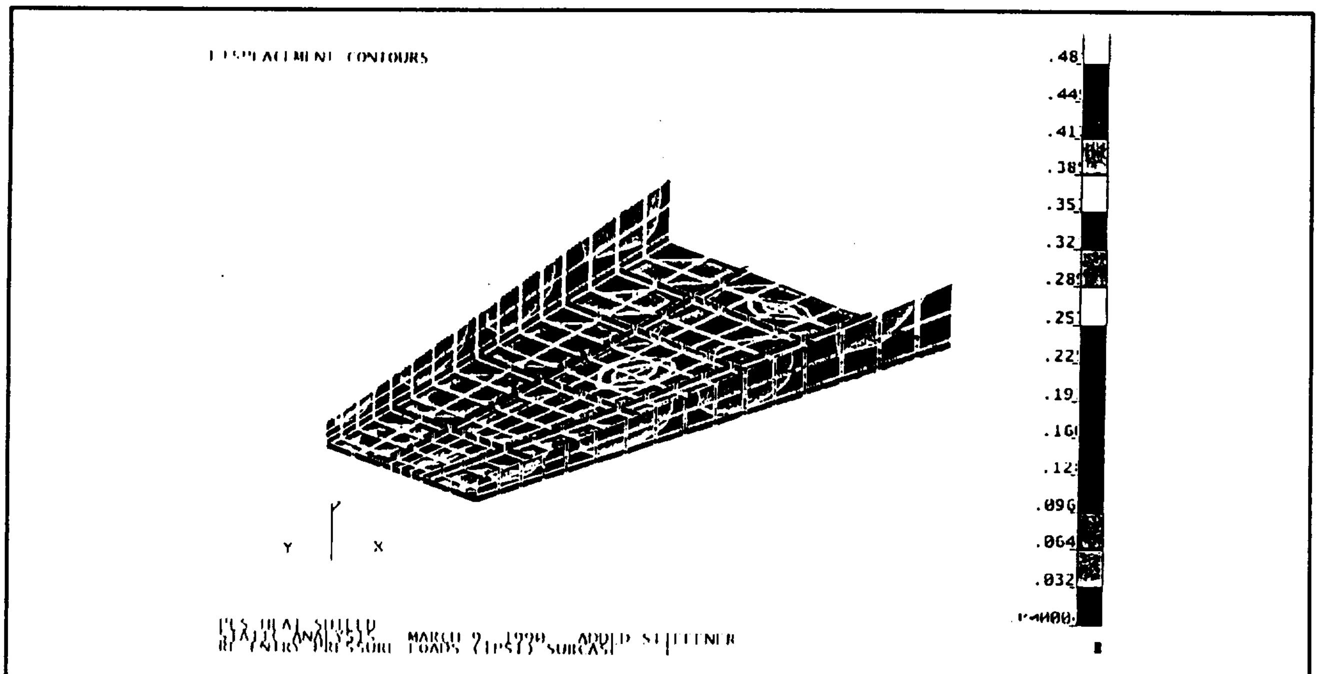


Figure 5-27. NASTRAN Model - Lower Heatshield Displacement Contours

For the temperature distribution load case, the model was allowed in-plane expansion, due to the nature of the joints connecting the heatshield to the vehicle. With this assumption, the contribution of this load case to the radius of curvature is zero. However, if the joints do not allow the free expansion of heatshield, internal loads will develop which will further warp the heatshield. Figure 5-28 shows the free expansion of the heatshield.

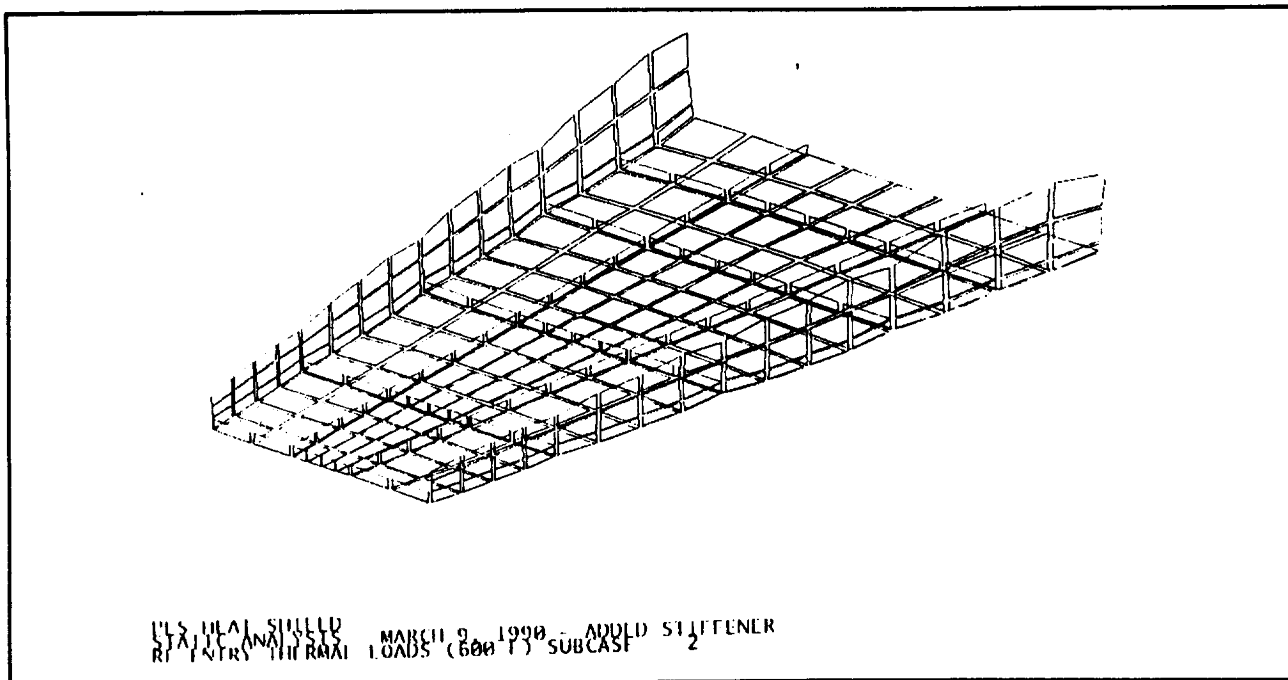


Figure 5-28. NASTRAN Model - Lower Heatshield (In Plane Free Expansion)

5.1.6 Landing Gear

The nose gear and main gear are electrically deployed and stowed by redundant low power electromechanical actuators. These actuators are driven either by redundant auto-control circuitry of the auto-flight guidance computer or by manual switch activation. In an emergency case, the nose gear and the main gear can be mechanically deployed to "free fall". Gear-Stowed and Gear-Deployed annunciators are located on the flight panel. Gear status data are also sent to the onboard maintenance system (OMS) computer for processing and to the central master communication system computers for telemetry to ground operations.

The landing gear for PLS is based on current fighter technology concepts. The nose gear is a conventional dual-wheel design that pivots aft to deploy. The main gear is a single wheel concept for each side, also with a down-and-aft deployment motion. Both the nose and main gear use electric deployment with a pyro backup. Both designs also use electric braking hardware with the nose gear using electric steering and the main gear employing an antiskid feature.

The nose gear is mounted to the forward cabin bulkhead with a strut rake of eight degrees. The main gear is mounted to the forward (front spar) wing carry-through frame.

To reduce the gear slapdown tendency of the PLS vehicle, the main landing gear was moved forward nearly 10 inches from its initial position on the "reference" configuration. Although from a c.g. relationship standpoint this location is still too far aft, to

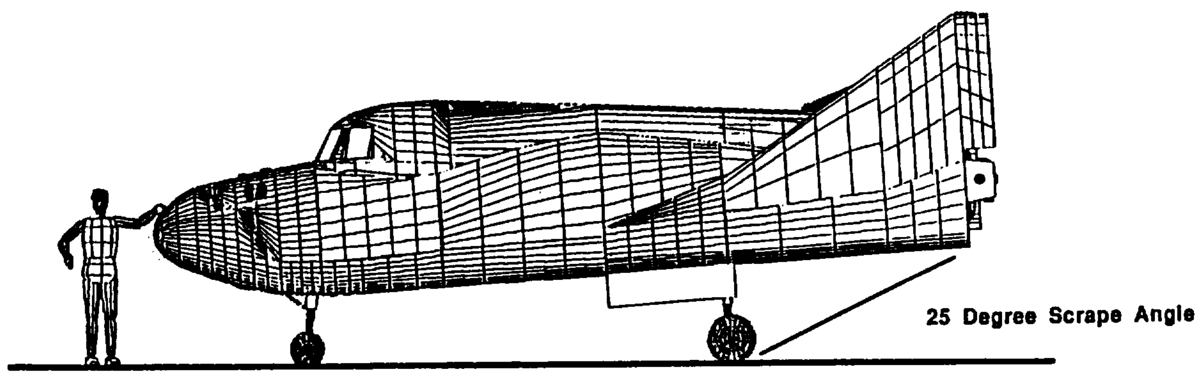


Figure 5-29. Tail Scrape Clearance at Landing

move it any farther forward would have detrimental effects on the primary structure, fin and gear load paths, passenger seat pitch and wing folding design. Also, as the main gear is moved farther forward, it must lengthen to achieve a satisfactory tail ground clearance. (The current tail clearance angle is 25 degrees (see Figure 5-29) which is more than the minimum recommended value of 17 degrees).

Lengthening the landing gear would have serious stability consequences in addition to increasing the weight of the landing gear subsystem. A vehicle overturning calculation was performed on the preferred PLS landing gear geometry, using the following equation; $A=\arctan(2hL/W(L-d))$, where A is the overturning angle, W is the track width of the main gear, d is the location of the c.g. ahead of the main gear, h is the location of the c.g. above groundlevel and L is the vehicle wheelbase. The results are an angle, A of 50 degrees at touchdown and 44 degrees during rollout, both of which are acceptable values.

5.1.7 Thermal Protection/Thermal Control System

The trajectory used to evaluate the PLS vehicle re-entry heating is the same as that used for the Rockwell ACRV lifting body vehicle. The PLS and the ACRV lifting body configurations are nearly identical except for two features. The forward half of the AMLS/PLS has a trapezoidal-shaped cross-section whereas the ACRV is more rectangular in shape. The other difference is that the lower trailing edge of the ACRV body is flared upwards whereas the AMLS/PLS vehicle is straight in this area.

The entry trajectory is based on a vehicle reentering with a maximum L/D of 1.3 (Figure 5-30). This trajectory was modified to include a time-to-touchdown segment to provide a complete thermal analysis since the original trajectory terminated at an 85,000 foot altitude. The Space Shuttle STS-5, End-of-Mission flight segment was used to complete the trajectory.

Aerothermal Analysis. The heating rates were computed for only the convective component assuming real gas, equilibrium chemistry. Since the AMLS/PLS vehicle will be reentering the atmosphere from Earth orbit, the hot-gas radiation to the vehicle surface will be a negligible contributor to the total heating and therefore, it was not calculated in this analysis. All heat fluxes reported here are radiation equilibrium values. Heating rates at four constant wall temperatures were calculated and are available for review.

Figure 5-31 shows the locations where the reentry heating was evaluated. These points include the nose stagnation point, points along the lower and upper body centerline, points along the vehicle shoulder and side, a point on the fin leading edge and points on the lower and upper wing surfaces. Figure 5-31 also shows the maximum heating rates the surface locations experience during reentry. Figure 5-32 presents the total heating history of a selected body point (e.g. the nose stagnation point).

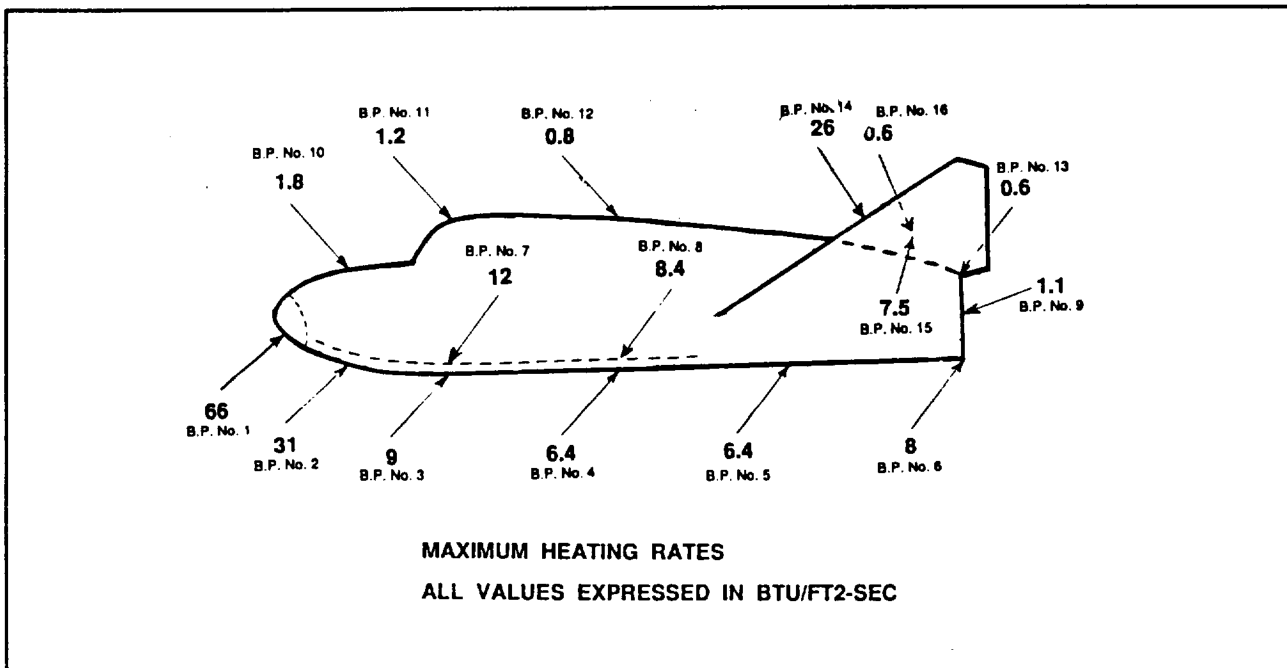


Figure 5-31. Locations For Entry Heating Analyses

The type of thermal protection material needed for the vehicle can be roughly estimated by assuming the surface temperatures to be equal to the radiative equilibrium wall temperatures calculated from the radiative equilibrium heat fluxes. For example, the heat

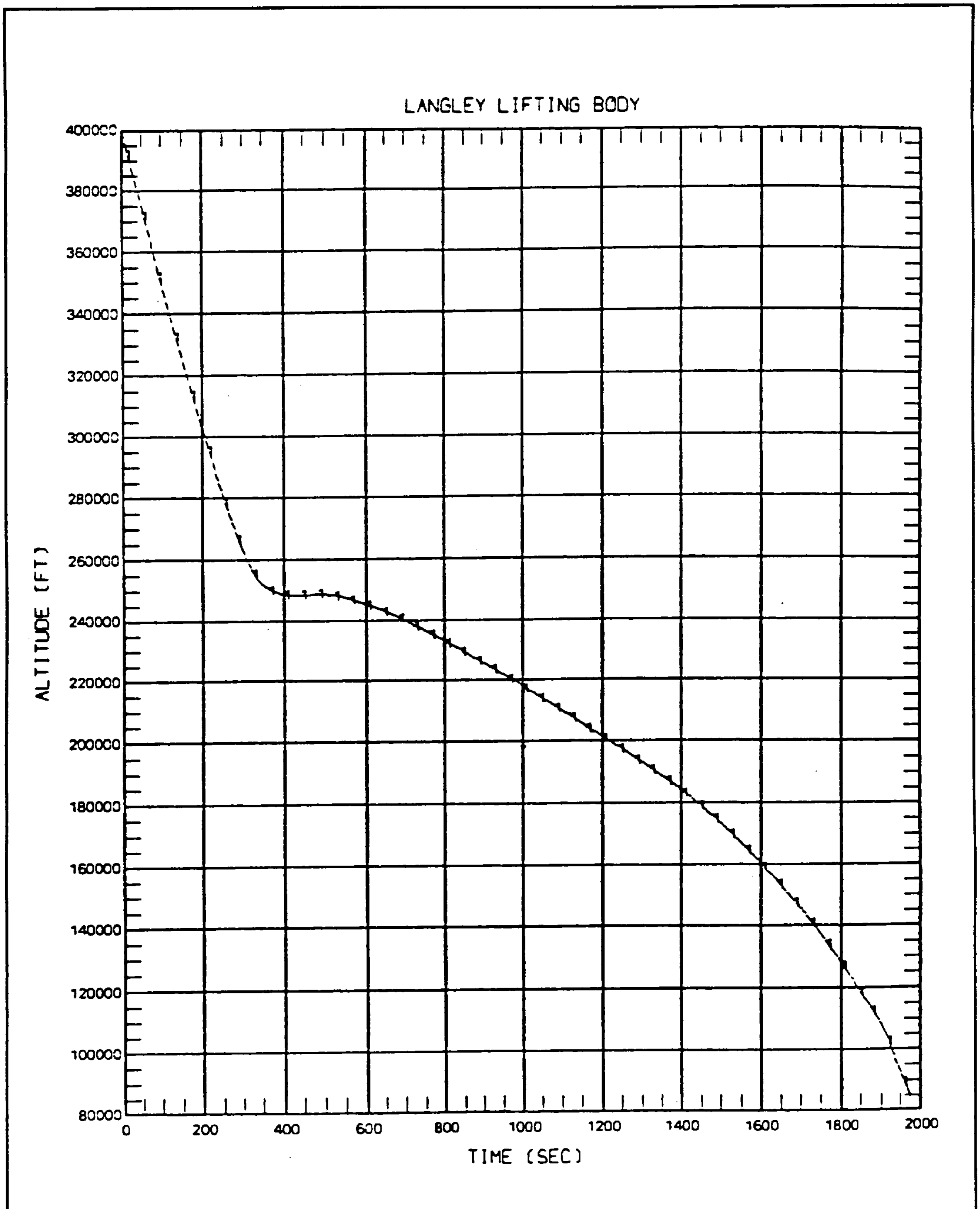


Figure 5-30. Reference Entry Trajectory for Thermal Analyses

flux value of 52 Btu/ft2-sec corresponds to a radiative equilibrium

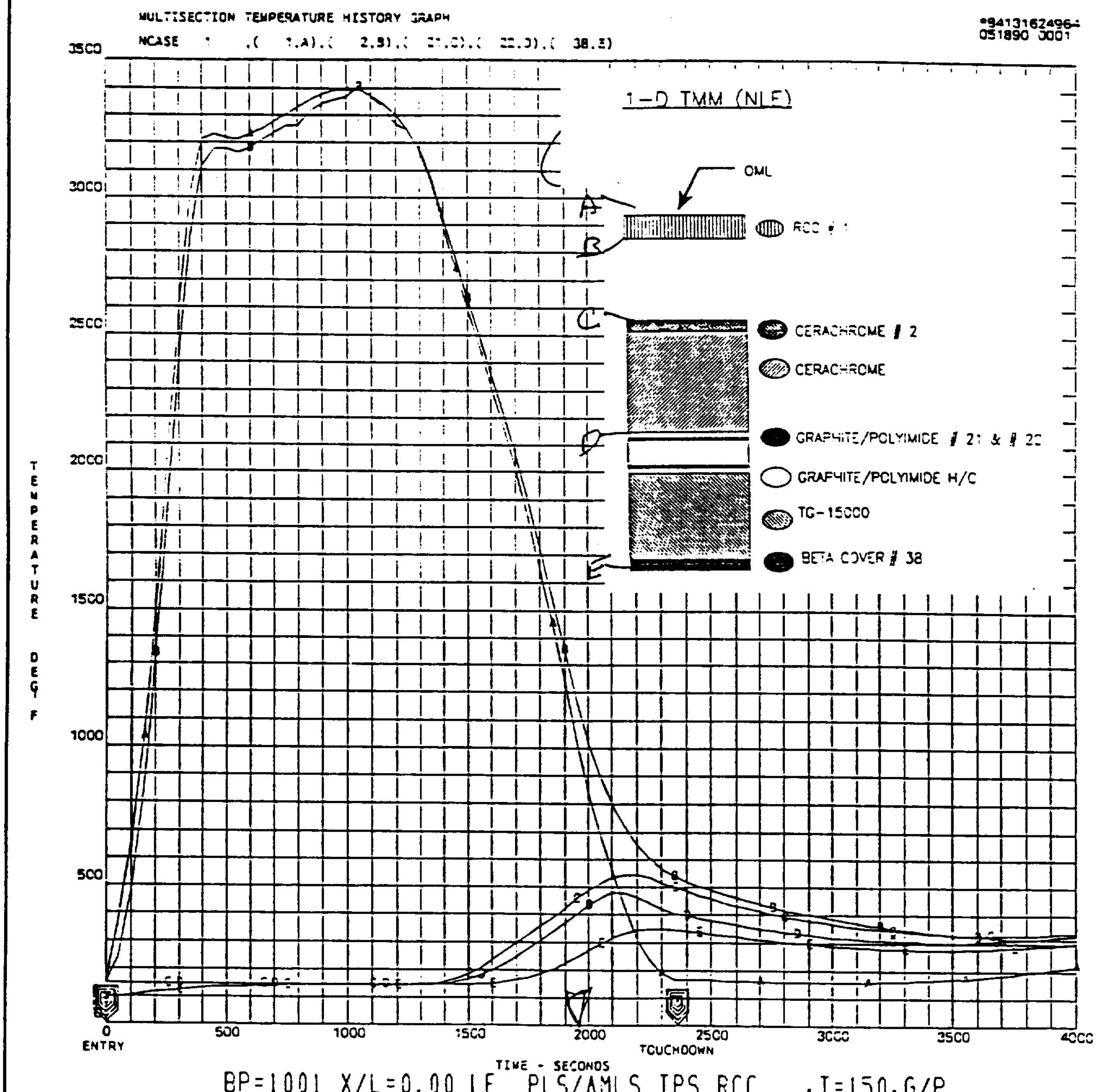


Figure 5-32. Nose Stagnation Point Heating Time History

wall temperature of 2900 degreesF which is the limit of high-temperature insulation (e.g. Shuttle tile). Above this value, higher temperature materials such as RCC or ACC must be considered. It is apparent that the vehicle will require some of these higher temperature materials around the nose area and on wing leading edges. All other areas can be covered with state-of-the-art insulation materials. All of the upper surface values are conservative. Minimum attached flow heating was used for the upper surface since leeward flow heating is difficult to predict analytically.

<u>TPS/TCS Descriptions.</u> The flight environment used to size the TPS materials is based on the nominal trajectory described above. Most of the materials specified for the AMLS/PLS vehicle have been certified for 100 missions by the Space Shuttle Program. The High Thermal Performance (HTP) tile material and graphite polyimide honeycomb structural material are relatively new to manned space vehicle design. These new materials will require a test program to obtain certifiable thermal performance data.

The TPS sizing methodology employed is derived from the Space Shuttle Program. Twenty four one-dimensional thermal math models were constructed at the locations shown in Figure 5-31 to simulate TPS and structure temperature response due to the reentry heating. The TMMs were analyzed using the Rockwell Multidimensional Heat Conduction Computer Program, XF0031.

The one-dimensional TMMs used for the PLS analysis ignored gap heating in the tile gaps because of the reduced tile gap dimensions using the direct-bond tile concept. The TMMs also ignored the delta t-bars (the effective heat capacity of nearby structure) because of the minimal heatsink capacity of the graphite polyimide heatshield structure material.

Inner moldline (IML) cooling, from Shuttle experience, was assumed to take place during the AMLS/PLS vehicle TAEM maneuver. IML cooling is convective heat transfer from the structure to the air. For the Shuttle, air vent opening during entry occurs at a velocity approximately 2400 ft/sec which is after peak reentry heating. For the AMLS/PLS TPS sizing analysis, this same velocity was assumed for the initiation of IML cooling.

For construction of the TMMs, the AMLS/PLS vehicle was divided into five representative areas. These areas are the lower surface, upper surface, side, wing and leading edges. The TMMs define the TPS and structural characteristics of each area.

The lower surface contains ten one-dimensional TMMs, of which five TMMs are located along the vehicle centerline coinciding with aeroheating analysis body point locations. The centerline TMM configurations are illustrated in Figure 5-34. The HTP-6 tile material was sized to a maximum bondline temperature of 550 degreesF and the TG-15000 internal insulation was sized to achieve a maximum aluminum primary structure temperature of 120 degreesF.

The remaining five lower surface TMMs are located at the same vehicle X-axis stations as the aeroheating body points but are located outboard of the crew cabin. The outboard TMM configurations are illustrated in Figure 5-35. The HTP-6 tile material was sized to a maximum bondline temperature of 550 degreesF and the TG-15000 internal insulation material was sized to produce a maximum enclosure and aluminum frame temperature of 350 degrees F.

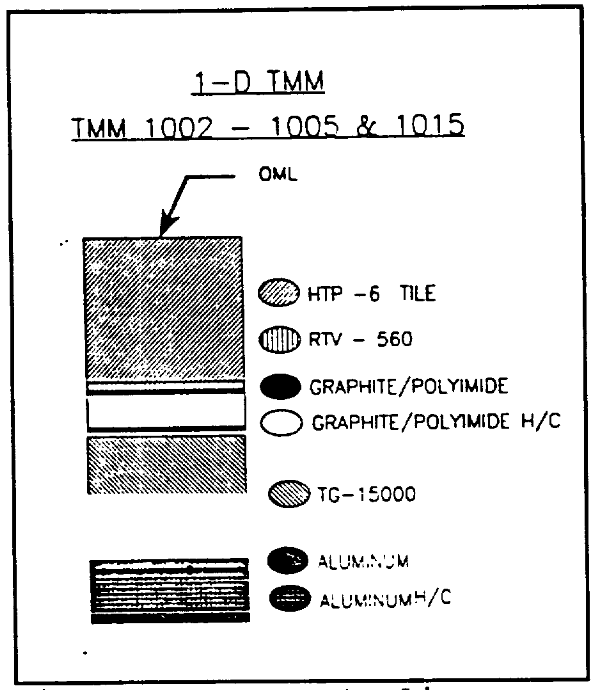


Figure 5-34. Centerline Thermal Math Models

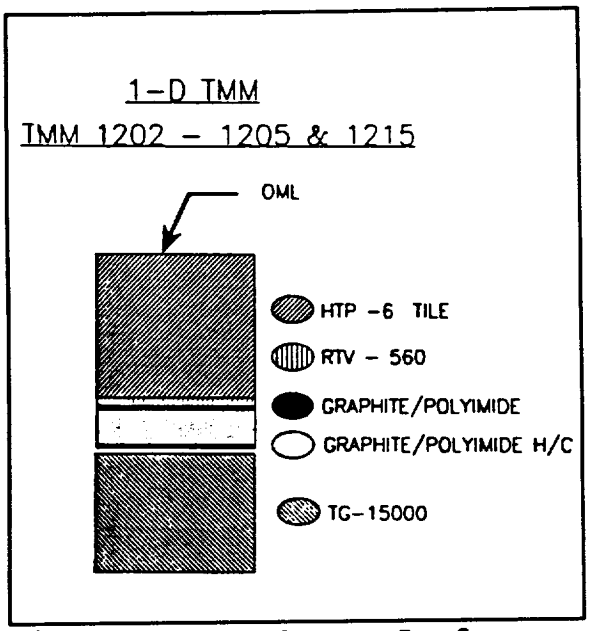


Figure 5-35. Lower Surface Outboard Thermal Math Models

The upper surface contains seven one-dimensional TMMs (Figure 5-36), of which four are located along the upper centerline coinciding with the aeroheating body point locations. The AFRSI insulation blanket material was sized to achieve a maximum bondline temperature of 550 degreesF and the TG-15000 insulations was sized to produce a maximum aluminum structural temperature of 120 degreesF.

The remaining three TMMs (Figure 5-37) are located at the corresponding vehicle X-axis stations of the aeroheating body points but outboard of the crew cabin, The AFRSI material was sized for a maximum bondline temperature of 550 degreesF and the TG-15000 internal insulation material was sized to achieve a maximum enclosure temperature of 350 degreesF.

The side area consists of three on-dimensional TMMs which coincide with the aeroheating body point locations. The side TMMs are illustrated in Figure 5-38. The HTP-6 tile material was sized to achieve a maximum bondline temperature of 550 degreesF and the TG-15000 insulation was sized to produce maximum enclosure temperatures of 350 degreesF.

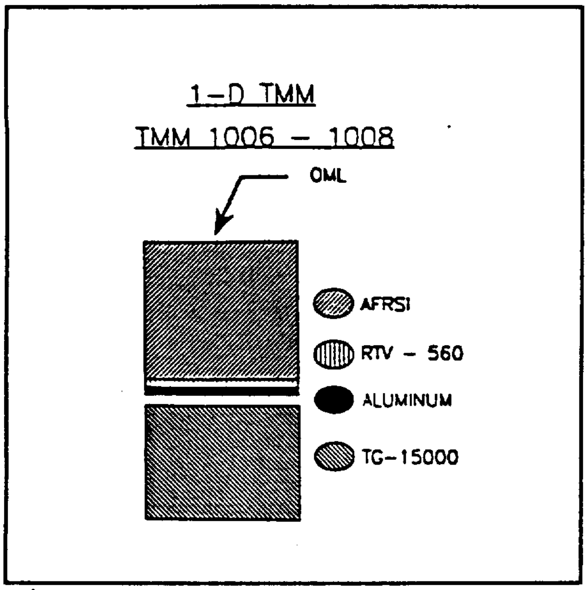


Figure 5-36. Upper Surface Thermal Math Models

The wing area contains a single on-dimensional TMM which is located at the pair of inner and outer aeroheating body points. This TMM is illustrated in Figure 5-39. The insulation materials were sized to achieve a bondline temperature of 550 degreesF.

The leading edge areas contain two one-dimensional TMMs located at the nose stagnation point and wing leading edge stagnation point. These areas are high entry heating locations and require an outer moldline material of RCC or ACC. In the nose area, the cerachrome insulation material was sized to a maximum structure temperature of 550 degreesF and the TG-15000 insulation was sized to produce a maximum interior surface temperature of 350 degreesF. For the wing leading edge area, the cerachrome insulation material was sized to produce a maximum spar structure temperature of 550 degreesF. The results of the TPS/TCS sizing analysis are presented in Table 5-2 which shows TPS thicknesses and maximum material temperatures. The temperature response histories for each Thermal Math Model are provided in Reference 5-1.

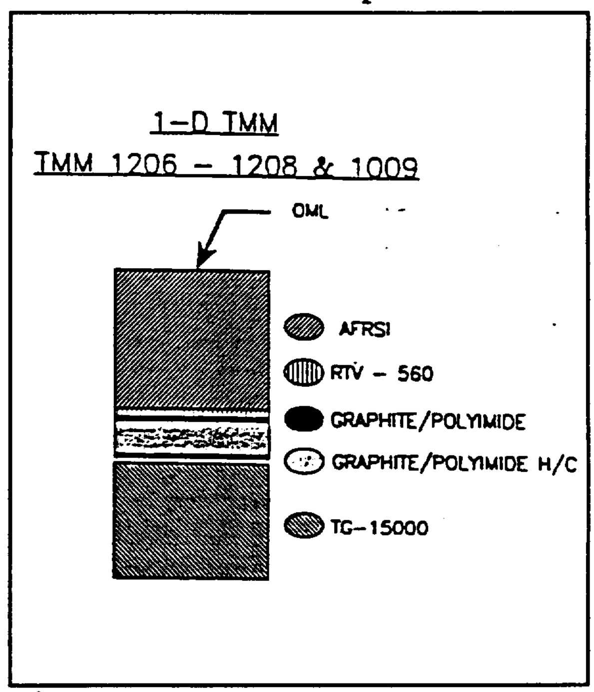


Figure 5-37. Thermal Math Models Outboard of the Crew Cabin

From the results of the analysis, two design issues were revealed. First, the 3400 degreeF temperature predicted for the nose stagnation point exceeds the maximum allowable temperature for RCC by some 200 degreesF. Either an ACC material must be certified for this temperature and application or the flight trajectory must be modified to reduce the stagnation point heating. A third option is to increase the nose radius of the vehicle to reduce the nose surface temperature. Full-temperature ceramics are a long range possibility but NASP applications find it necessary to use active cooling in their high temperature areas (2800 to 3100 degreeF).

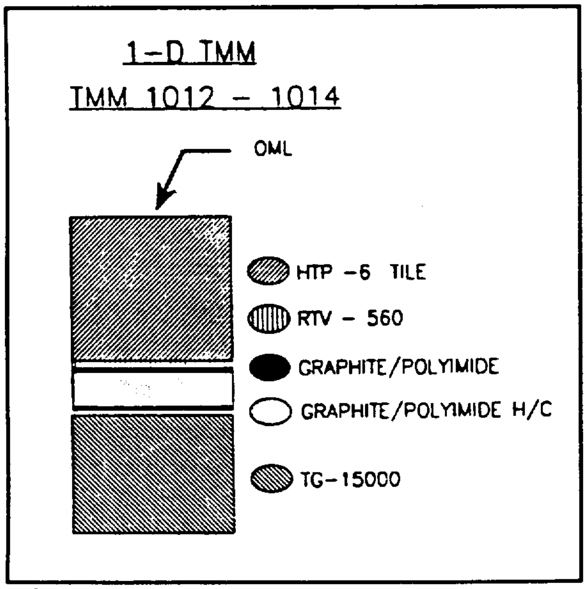


Figure 5-38. Thermal Math Models for the Side Area

The second design issue is really a manufacturing item. The calculated AFRSI thicknesses for PLS range from 0.88 inches to 0.08 inches. The AFRSI blankets for the Shuttle Program are currently being manufactured in nine different thicknesses ranging from 1.60 inches to only 0.41 inches. If the AFRSI material cannot be manufactured in the thinner dimensions, then some areas of the upper PLS surface will receive insulation with 400% design margin.

The results of the sizing analysis are based on the assumption of a nominal entry trajectory and a maximum entry interface temperature of 150 degreesF. These two parameters have a direct effect on the TPS thicknesses. A design entry trajectory will have to be developed which encompasses all possible cross range trajectories (landing sites) and abort scenarios

to size the TPS for a "worst case" mission. The design entry interface temperatures are based on the sun and Earth orientation to the vehicle, the TPS optical properties and the PLS on-orbit duration. The design entry interface temperatures will have to encompass all possible orbit operations to determine the highest temperatures the vehicle could experience at entry interface.

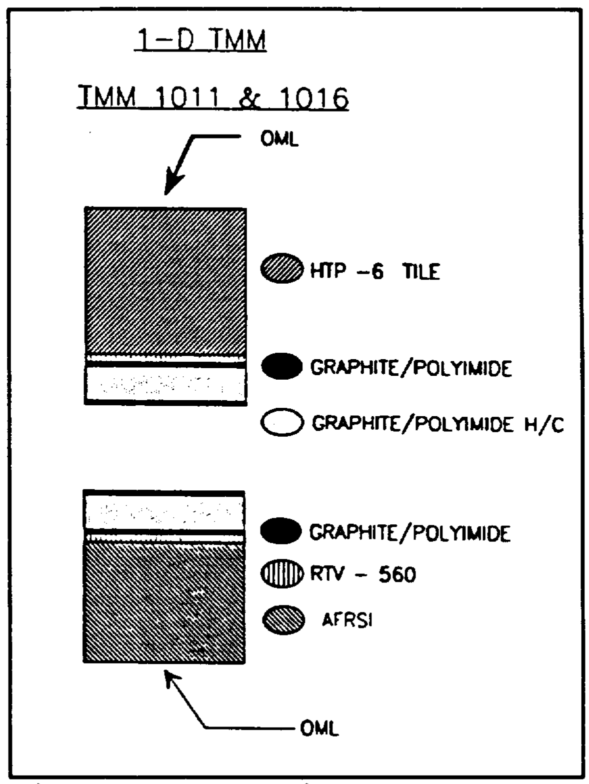


Figure 5-39. Wing Thermal Math Model

This doesn't suggest that the TPS sized here is marginal. On the contrary, the results summarized in Table 5-2, indicate a substantial design margin on most of the vehicle lower surface. The "reference" vehicle design allowed a six-inch dimension between the lower vehicle outer surface (OML) and the crew cabin lower structural moldline over the entire lower surface of the PLS for the total of TPS, heatshield and TCS thicknesses. As Table 5-2 shows, the sum of the tile thickness, the half-inch graphite polyimide heatshield honeycomb thickness and the internal insulation thickness at all body points is less (in some places, much less) than the 6-inch allowance. This means that either some growth in TPS/TCS dimension can be accommodated for more vigorous trajectories or that the crew cabin temperatures can be reduced from the levels assumed in this analysis.

Table 5-2. TPS Thicknesses and Material Temperatures

1-D TMM	PLS AREA	PLS X/L	TEMP. FIG #	EXTERNAL TPS THICK.	INTERNAL TPS THICK.	MAXIMUM MATERIAL TEMPERATURE							
						RCC	CERA	HTP-6	RTV	AL	G/P	AFRSI	TG15
1001	NLE	0	2.2	5.426	1.484	3397	3402	-----	-----	-----	548	-----	463
1002	LSCL	0.25	2.3	1.723	1.567	-----	-----	1719	548	120	547	-----	534
1202	LSOB	0.25	2.4	1.752	1.473	-----	-----	1719	548	-----	547	-----	535
1003	LSCL	0.5	2.5	1.501	1.526	-----	-----	1542	548	120	547	-----	535
1203	LSOB	0.5	2.6	1.537	1.509	-----	-----	1542	546	-----	546	-----	535
1004	LSCL	0.75	2.7	1.399	1.504	-----	-----	1509	548	120	547	-----	535
1204	LSOB	0.75	2.8	1.43	1.506	-----	-----	1509	548	-----	547	-----	538
1005	LSCL	1	2.9	1.337	1.487	-----	-----	1608	548	120	548	-----	535
1205	LSOB	1	2.10	1.386	1.48	-----	-----	1609	548	-----	547	-----	537
1006	USCL	0.1	2.11	0.876	1.108	-----	-----	-----	548	348	-----	1115	343
1206	USOB	0.1	2.12	0.298	2.482	-----	-----	-----	549	-----	547	1090	537
1007	USCL	0.25	2.13	0.588	0.725	-----	-----	-----	550	349	-----	915	344
1207	USOB	0.25	2.14	0.18	2.56	-----	-----	-----	549	-----	548	877	537
1008	USCL	0.5	2.15	0.428	1.1	-----	-----	-----	548	347	-----	783	342
1208	USOB	0.5	2.16	0.116	2.626	-----	-----	-----	549	-----	547	735	536
1009	USCL	1	2.17	0.078	2.708	-----	-----	-----	548	-----	546	623	539
1010	WLE		2.18	5.711	-----	2374	2345	-----	-----	-----	547	-----	-----
1011	WLS		2.19	2.437	-----	-----	-----	1613	550	-----	548	-----	-----
1012	SIDE	0.25	2.20	1.831	1.403	-----	-----	1863	548	-----	547	-----	535
1013	SIDE	0.5	2.21	1.542	1.528	-----	-----	1681	550	-----	549	-----	541
1014	SIDE	1	2.22	0.338	1.991	-----	-----	846	549	-----	548	-----	542
1015	LSCL	0.1	2.23	3.119	1.79	-----	-----	2504	548	-----	547	-----	533
1215	LSOB	0.1	2.24	3.266	4.912	-----	-----	2504	549	-----	549	-----	542
1016	WUS		2.19	0.052	-----	-----	-----	-----	551	-----	546	608	-----

To summarize the results of the TPS study, the AMLS/PLS vehicle design has no major TPS technical concerns. The TPS materials available today are adequate to protect the vehicle structure from the extreme temperatures of the proposed entry trajectory.

The AMLS/PLS vehicle design has a few minor TPS technical concerns to be addressed during the subsequent design phase. The AFRSI material thickness issue for the upper vehicle surface, the on-orbit thermal gradient control or assumption, the effect of the entry interface temperatures and the effect of the structural thermal gradients during entry are highlighted as important issues for follow-on assessment.

<u>Avionics Thermal Control Analysis.</u> A fundamental operational goal of the PLS program is to eliminate the use of cold plates for avionic component cooling and to adopt a "passive" (heatsink) cooling concept for all avionics (and other heat generating equipment to the extent possible). The historic trend of avionics components has been the general reduction in size, weight and

especially in power consumption. The PLS exploits this development by proposing a passive concept which is similar to that employed by the avionics systems of modern aircraft in unpressurized compartments. This study was undertaken to verify the feasibility of this approach and to understand the thermal performance of representative avionic system components in the PLS installation environment.

Another contribution to the viability of the passive cooling approach is the assumption that the PLS will be operated in modes which are compatible with the passive concept. The heat generation history of the PLS mission benefits from the low power consumption (and therefore, heat rejection) of the system during the powered-down mode which the vehicle assumes when it is docked at Space Station. Operating the PLS in this fashion potentially puts the vehicle in the simpler cooling method regime.

In addition to the low heat dissipation of modern avionics and the mission operation approach, a third factor supporting the passive concept is the higher heat tolerance characteristics of modern avionics. Recent studies have shown that there is very little steady state avionics temperature difference between active and passively cooled avionics configurations. Modern avionics temperature specifications for aircraft state requirements of 230 degreesF for 30 minutes which is above the environmental conditions of PLS as determined by this analysis.

A thermal math model was developed for the avionics boxes using Rockwell's General Thermal Analysis Program (GTAP). Various box stacking arrangements were considered and a combination was selected which is representative of a worst (hottest) case. This arrangement was analyzed for box temperature history and temperature distribution.

Avionics boxes are located both inside (principally the displays and controls) and outside the crew cabin. Figure 5-40 shows the general location of boxes external to the pressure vessel of the PLS vehicle in an "avionics bay". The electronic boxes are stacked in such a way so that the box with a higher power dissipation is next to one with a lower power dissipation. Any stacking order can be accommodated using connecting cables. Each box is attached to an aluminum frame which is integrated with the PLS primary structure. The list of avionics equipment and related duty cycles developed from the PLS power history analysis and used in this thermal assessment is shown in Table 5-3.

The following assumptions were made in developing the Thermal Math Model (TMM):

1. The mounting frame is at the same temperature as the vehicle structure.

2. The vehicle structure acts as a heat sink and can absorb all the thermal dissipation produced by the avionics.

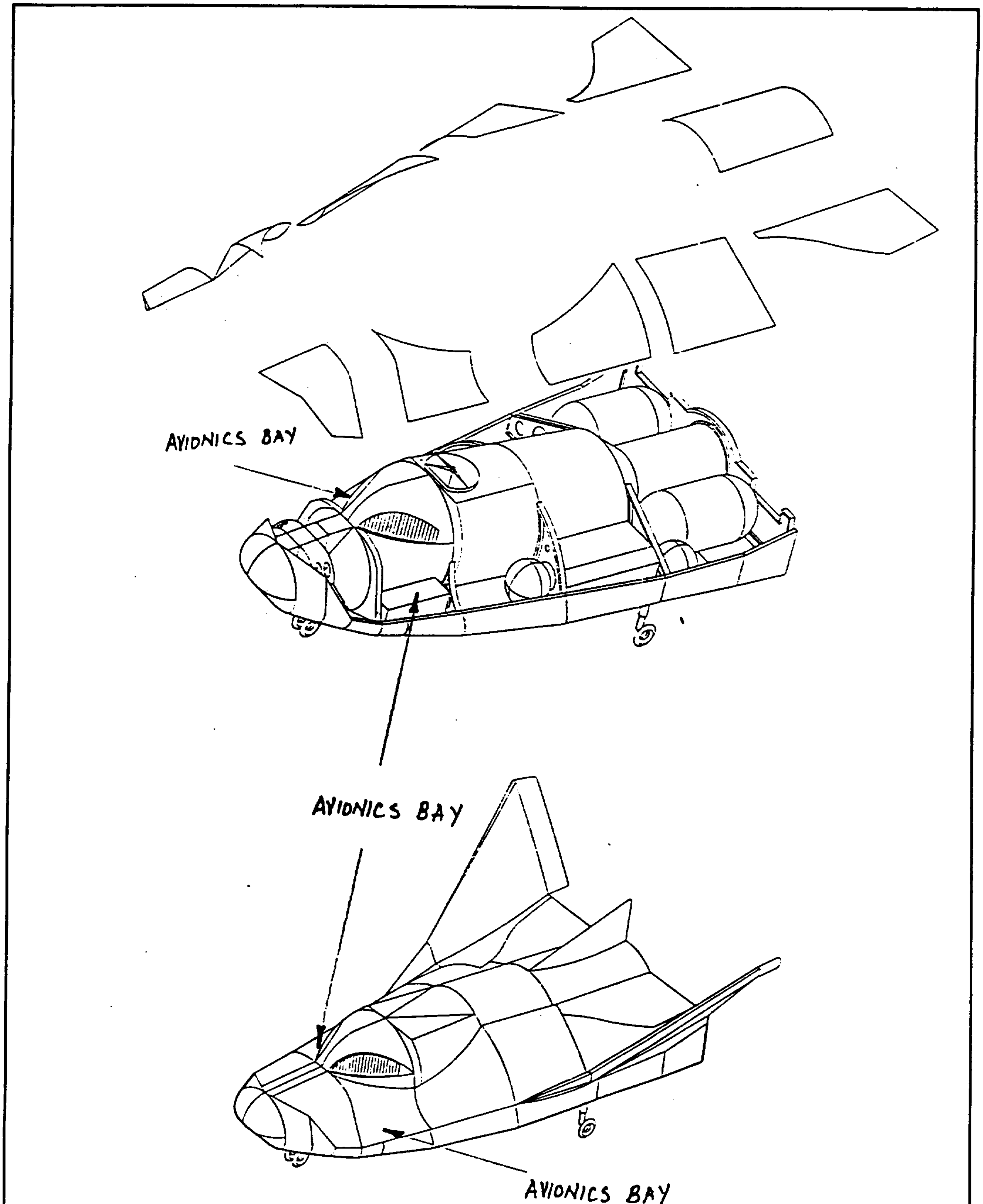


Figure 5-40. Location of Avionics Boxes

Table 5-3. Avionics Boxes and Related Duty Cycles

ASCENT	Unit	POWER		Duty Cycle (%)				Energy	
		Unit	Total	Pre-	Launch	OMS	Rend&Prox		
Vehicle Subsystem/Equipment	Qty	(Watts)	(Watts)	0.17	1.00	0.20	11.50	(kWh)	
Avionics (GN&C)									
IMU - LINS 750	2	33.00	66.00	100.00	100.00	100.00	100.00	0.85	Autonetics
GPS Receiver(5-Channel, 8lbs)	2	12.00	24.00	100.00	100.00	100.00	100.00	0.31	Collins
GPS Antenna	2	20.00	40.00	100.00	100.00	100.00	100.00	0.51	Collins
Altimiter	2	30.00	60.00	100.00	100.00	100.00	100.00	0.77	
Air Data Probe	1	72.00	72.00	0.00	0.00	0.00	0.00	0.00	
Air Data Assembly	1	35.00	35.00	0.00	0.00	0.00	0.00	0.00	
Horizon Sensor Elect.	2	11.00	22.00	100.00	100.00	100.00	100.00	0.28	
Horizon Sensor (Head)	2	25.00	50.00	100.00	100.00	100.00	100.00	0.64	
Startracker	1	30.00	30.00	0.00	0.00	0.00	0.00	0.00	
Microwave Landing System	1	10.00	10.00	0.00	0.00	0.00	0.00	0.00	
Rendezvous Radar	1	30.00	30.00	0.00	0.00	0.00	11.00	0.04	
Radar Signal Processor	1	30.00	30.00	0.00	0.00	0.00	11.00	0.04	
Avionics (Data Processing)									
Main Computers(A&C)	2	126.00	252.00	100.00	100.00	100.00	100.00	3.24	Drapper
Backup Computer (B)	1	111.00	111.00	100.00	100.00	100.00	100.00	1.43	Drapper
RCS Controllers	2	202.00	404.00	100.00	100.00	100.00	100.00	5.20	JIAWG
Battery Controllers	2	111.00	222.00	100.00	100.00	100.00	100.00	2.86	JIAWG
Main Engine Controller	1	202.00	202.00	100.00	100.00	100.00	0.00	0.28	JIAWG
Aerosurface Controller	4	132.00	528.00	100.00	100.00	0.00	0.00	0.62	JIAWG
Avionics (Health & Monitor)									
Backup Health & Monitoring	1	40.00	40.00	0.00	0.00	0.00	0.00	0.00	Hamilton Std
Sensors	700	0.10	70.00	100.00	100.00	100.00	100.00	0.90	Hamilton Std
Fire Suppression System	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	
Avionics (Comm. &Tracking)									
Audio Terminal	2	1.00	2.00	100.00	100.00	100.00	100.00	0.03	
GPS Antenna Switch	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	
Headset & Mike	2	5.00	10.00	100.00	100.00	100.00	100.00	0.13	
Panel Display	2	15.00	30.00	100.00	100.00	100.00	100.00	0.39	Collins
S-Band Antenna Switch	2	5.00	10.00	100.00	100.00	100.00	100.00	0.13	
S-Band Transponder	2	25.00	50.00	100.00	100.00	100.00	100.00	0.64	
SARSAT Transmitter	1	40.00	40.00	100.00	100.00	0.00	0.00	0.05	
UHF Transceiver	2	15.00	30.00	100.00	100.00	100.00	100.00	0.39	
Avionics (Disp. & Controls)									
Clock	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	
Display Controller	2	25.00	50.00	100.00	100.00	100.00	100.00	0.64	
Display and Selec. Control	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	
Elect. Time Base Gen.	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	
Head-up Display	2	40.00	80.00	100.00	100.00	100.00	100.00	1.03	
Lights, HI	2	40.00	80.00	100.00	0.00	0.00	100.00	0.93	
Lights, Low	10	5.00	50.00	100.00	100.00	100.00	100.00	0.64	
Switches (# As Required)		10.00	0.00	100.00	100.00	100.00	100.00	0.00	
Signal Conditioner	2	8.00	16.00	100.00	100.00	100.00	100.00	0.21	
Voice Recorder	1	5.00	5.00	100.00	100.00	100.00	100.00	0.06	23.54

3. The avionics boxes are attached to the standard aviation equipment racks with 6-32 fasteners.

Figure 5-41 illustrates the low heat production nature of the PLS mission operation. Because the heat generation is low and the temperature response of the avionics and the structure to which it is mounted is slow, the heatsink thermal technique is promising.

The avionics characteristics provided by Collins Military Avionics gave dimensions, connector requirements, operating conditions and power dissipation. Each avionics box is attached identically to the PLS avionics equipment racks. The resistance network for each box is also identical. In the computer model, all power consumption numbers are multiplied by a factor of 0.15 to convert them to thermal dissipation.

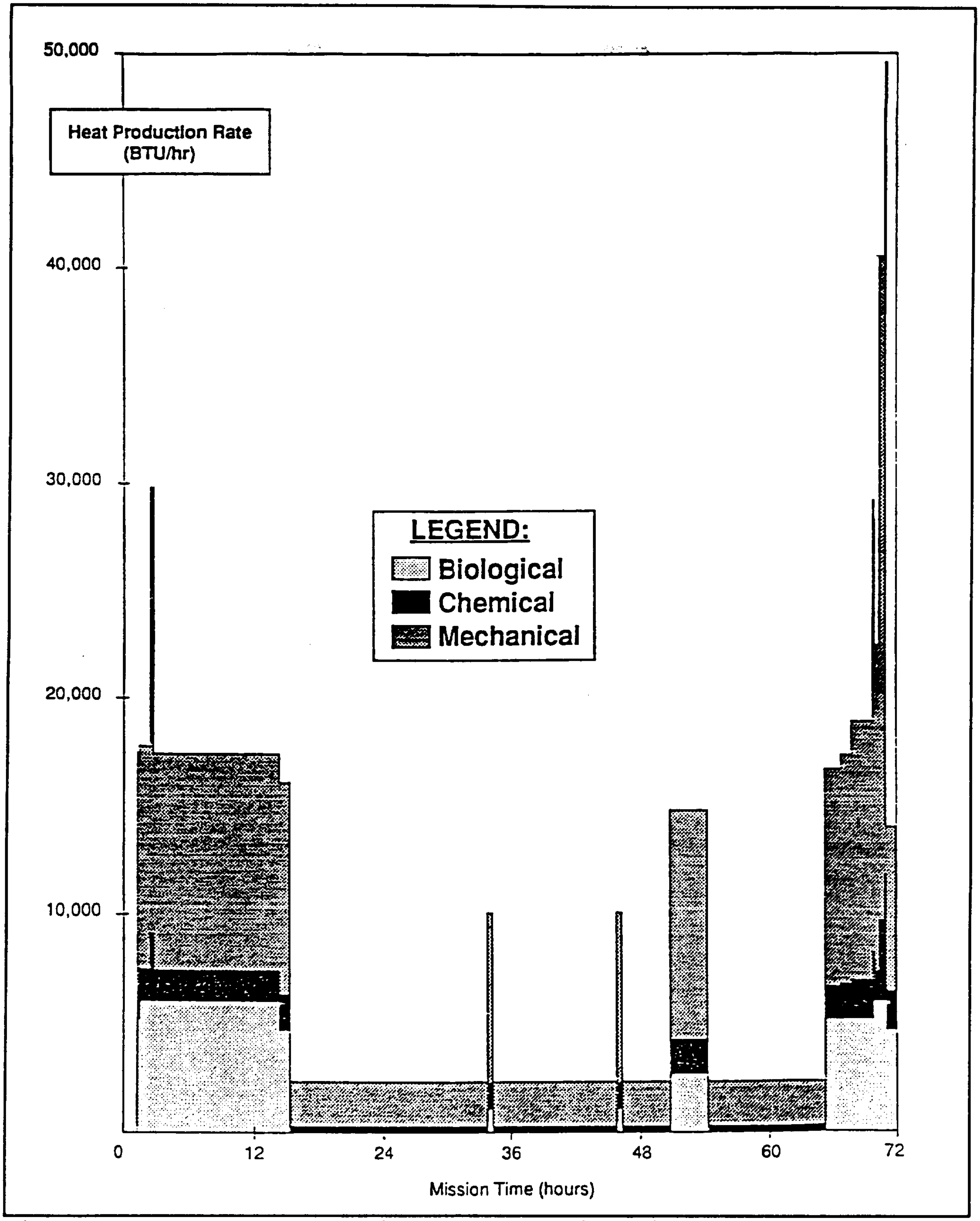


Figure 5-41. Heat Production of the Vehicle is Low

A transient thermal math model is developed by estimating the conducting mass of each avionics box. The electronic circuits, etc. may have copper, silver, aluminum and other conducting metals imbedded in plastic (polymer). Since the attach points are aluminum and most of the circuit support structure is also aluminum, an equivalent aluminum mass is assigned to each box. This is a conservative assumption, since the temperature response with time will be faster.

The conductance and heat dissipation were estimated, including the view factors between boxes and the surrounding enclosure. A thermal math model was then developed by inputting the conductances, surface areas, heat dissipation and view factors for the assigned nodes into the GTAP.

A steady state thermal model was run on an IBM 370 mainframe computer with various enclosure surface temperatures and also with additional conduction paths to the structure to reduce the box temperatures to the specified operating ranges.

Peak structural limitations established a temperature of 350 degreesF for the avionics bay as a TPS/TCS design upper limit. A temperature history of the avionics bay area was developed in the PLS aerothermal and TPS sizing study and is presented in Figure 5-42. The upper plot gives the temperature history of the outer moldline which peaks at about 1600 degrees F for this area of the vehicle surface. The three plots (traces B, C and D in the middle of the graph are for the graphite polyimide heatshield bondline area. Trace "E" represents the temperature history of the inside surface of the internal insulation (TG-15000) and shows that the peak of 330 degreesF is approached for only a few seconds before it begins to diminish. However, when the avionics thermal analysis was performed, the enclosure temperatures were defined as steady state temperatures of 250, 300 and 350 degrees F for the cases run. This approach adds a level of conservatism to the results.

The remaining parameter is the structure (sink) temperature. A range of 60 to 150 degrees F is used in the computer runs. Although the maximum design temperature of the crew cabin structure is 120 degrees F, the lower range is analyzed to broaden the study.

A series of computer runs showed that only in the case where the initial temperature of the cavity was 350 degreesF did the avionics briefly exceed their normal permitted steady state operating limits of 158 degreesF. With lower initial temperatures, the avionics are not expected to experience the marginal conditions.

5.1.8 Integrated Propulsion System

The propulsion system consists of an Orbit Maneuvering System (OMS) and a Reaction Control System (RCS). The propulsion system provides 1000 feet per second of delta-V capability with another 100 feet per second for Space Station proximity operations maneuvering.

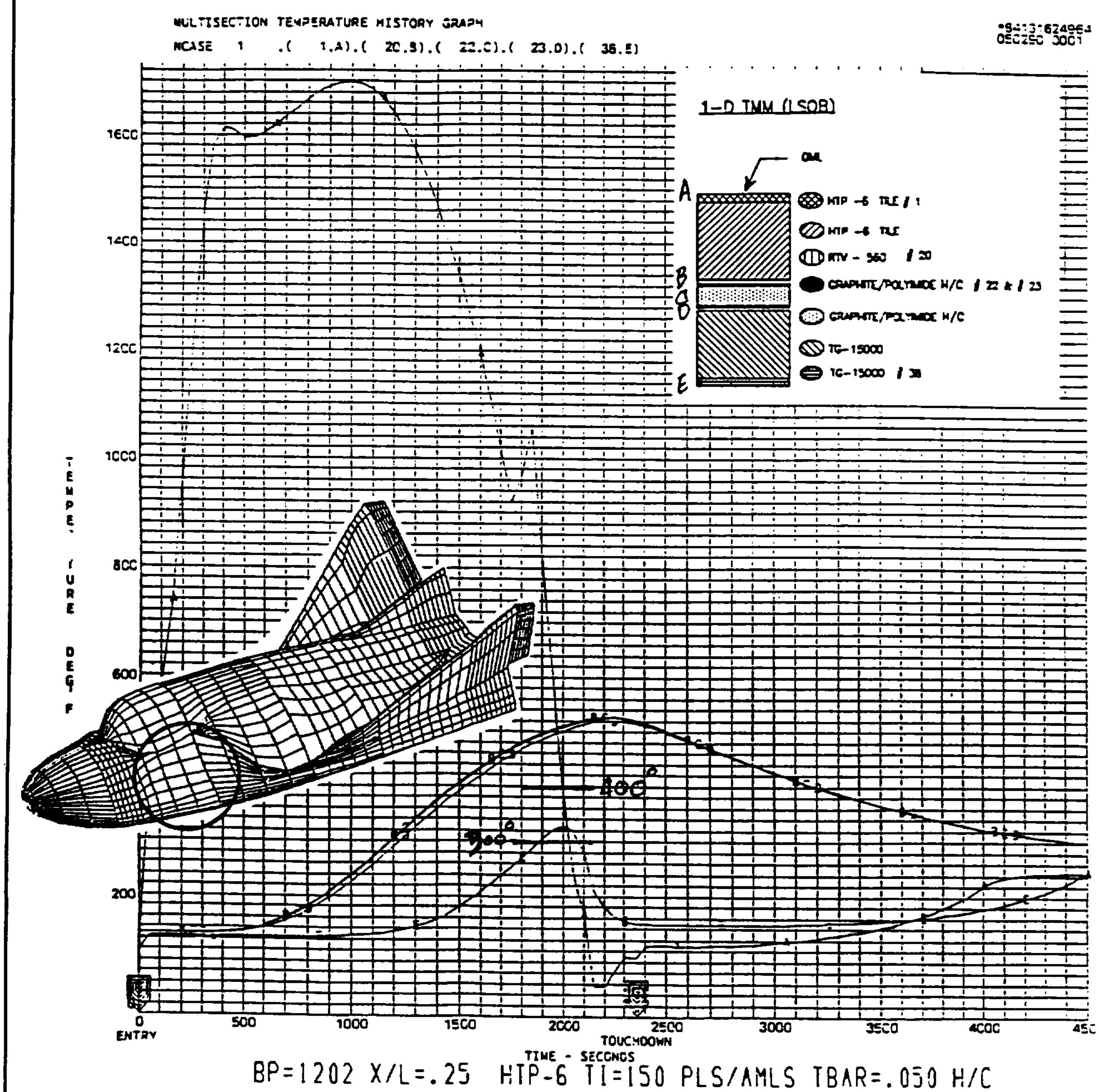


Figure 5-42. Temperature History for the Avionics Bay

The "reference" propulsion system concept for the PLS is a hydrazine monopropellant system designed with fail-op, fail-safe reliability. To support the ease of maintenance and rapid turnaround program objectives, the propulsion system design emphasizes low toxicity propellants and safe design approaches.

Alternate Propulsion Concepts. The propulsion system options to evaluate in the study are representative of a range of current propulsion technologies encompassing storable and cryogenic concepts. The concepts compared are:

1) the reference hydrazine system,

2) a bipropellant concept (MMH/N2O4),

3) three versions of a JP4/H2O4 system with different RCS modes,

4) a cryogenic concept (Methane/LOX).

Each propulsion system incorporates a full 6-degree of freedom RCS capability.

Hydrazine Monopropellant (Reference).

A monopropellant concept uses about half the hardware of a bipropellant system which increases system reliability, reduces ground operations and reduces system hardware and installation costs. However, the reduced performance of the hydrazine system results in a weight penalty. Additionally, the high toxicity of the propellant precludes "simple, airline-type" operations.

The system concept incorporates a cold gas (GN2), six degree of freedom system for use during Space Station proximity operations. The cold gas capability is integrated with the hot gas pressurization system to reduce tankage and hardware. (Space Station prox-ops is defined as being within 500 feet of the Space Station). The OMS and RCS tanks have been integrated to further reduce hardware and ground checkout.

Thrust levels are determined by scaling present Shuttle thrust levels to provide an equivalent thrust/weight ratio for the PLS. The results are an OMS thrust level of 1200 pounds (with four engines), an RCS thrust level of 80 pounds and a cold gas vernier thrust level of 2 pounds. Propellant weights are determined using the rocket equation and the following performance assumptions: an OMS Engine Isp of 230 seconds (Rocket Research MR-87a, 300lbf engine), an RCS Engine Isp of 215 seconds (Rocket Research MR-104, 80 lbf engine) and a GN2 engine Isp of 72 seconds.

Gaseous nitrogen is used as the cold gas for this system concept due to its low cost and ease of servicing compared to helium. It is assumed that the starting mass of the GN2 is the initial wet weight less 25% of the OMS and RCS load since, before cold gas initiation, it is assumed that the vehicle has achieved orbit and maneuvered to the Space Station. All propellant weights listed in the hardware list include a 15% margin for off-nominal performance, GN&C errors, etc. Tank sizes are determined using the density of hydrazine at 68 degrees F and adding 25% for ullage. This provides a blowdown capability in the event of a pressurization system failure. The GN2 tanks for propulsion and pressurization are integrated to reduce the number of regulators, tanks and isolation valves in the system. The GN2 tanks are sized based of a maximum operating pressure of 4000 psi per current Shuttle design practice.

The monopropellant, fail-op/fail safe concept (Figure 5-43) achieves two fault tolerance and continuous abort capability through dual-redundant feed paths for both the pressurant and the propellant, redundant thrusters and a limited blowdown capability.

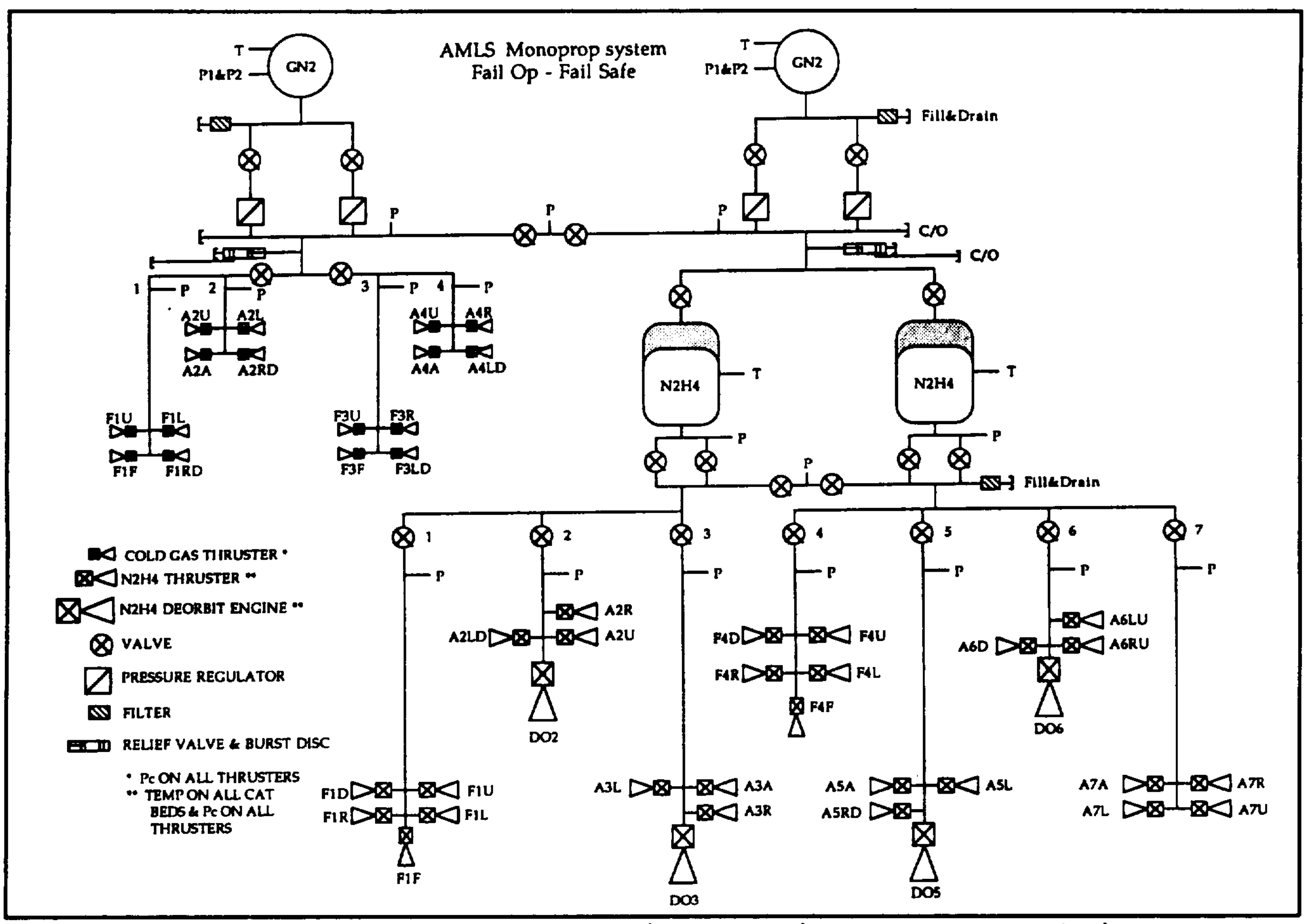


Figure 5-43. Monopropellant Fail-op/Fail-safe Propulsion System Concept

The hardware for this concepts is as follows:

Component	Weight	Notes
N2 Tanks (2)	200 pounds	5.9 cubic foot volume
N2H4 Tanks (2)	500 pounds	73.5 cubic foot volume
Lines	49 pounds	5% of Total Weight
Regulators (4)	8 pounds	Existing Components
GN2 Iso (10)	22 pounds	Existing Components
N2H4 Iso (13)	46 pounds	Existing Components
Relief (2)	4 pounds	Existing Components
Fill/Drain (7)	7 pounds	Existing Components
RCS Prim. (26)	107 pounds	Magellan, RR MR-104
RCS Vern. (16)	2 pounds	Existing Design
OMS (4)	80 pounds	RR Mr-87a
GN2	240 pounds	
N2H4	3842 pounds	
Dry Weight	1023 pounds	
Total Weight	5105 pounds	

There are a number of technological issues associated with the hydrazine monopropellant concept. They are:

1. The bladders for the non-spherical tanks may be more prone to twist and leak than for spherical tanks. Metal bladders may not be reusable.

2. Each mission may require 1.2 million lb-sec total impulse from the OMS engines. Present thrusters have demonstrated only 700,000 lb-sec total impulse. This is especially a problem with high thrust engines since the large propellant flow causes degradation of the catalyst beds.

3. The N2H4 components will require a man-rating program.

MMH/N2O4 Bipropellant Alternate.

A bipropellant system requires almost twice the hardware of the reference monopropellant system which decreases the system reliability and increases ground operations as well as hardware and installation costs. However, the increased performance of the MMH/N2O4 system results in significant weight savings (about 1300 fewer pounds at launch). Similar to the monopropellant concept, the hypergolic's high toxicity will make simple, airline-type ground operations difficult.

The concept also incorporates a cold gas (GN2), six degree of freedom system for use during Space Station proximity operations to reduce tankage and hardware. The design groundrules are the same as for the reference hydrazine system. The assumed thrust levels are 1200 lbf (total) for the OMS, 80 lbf for each RCS engine and 2 lbf for each cold gas vernier engine. Propellant weights are determined using the rocket equation and the following performance assumptions: an OMS engine Isp of 294 seconds (based on the performance of the Marquardt Mariner/Viking 300 lbf engine); an RCS engine Isp of 304 seconds (based on the performance of the Marquardt Apollo SM 100 lbf engine) and a GN2 cold gas engine Isp of 72 seconds. Gaseous nitrogen is used as the cold gas for the prox/ops due to its low cost and ease of servicing compared to helium. It is assumed that the starting mass for the GN2 is the initial wet weight less 25% of the OMS and RCS load since, before cold gas system initiation, it is assumed that the vehicle has achieved orbit and maneuvered to the Space Station.

All propellant weights in the hardware list include a 15% margin for off-nominal performance, GN&C errors, etc. Tank sizes are determined using the density of MMH/N2O4 at 68 degrees F with 25% volume added for ullage. This provides a blowdown capability in the event of a pressurization system failure. The GN2 tanks for propulsion and pressurization have been integrated to reduce the number of regulators, tanks and isolation valves in the system. The GN2 tanks are sized based on a maximum operating pressure of 4000 psi (current Shuttle RCS tank design).

The biprop system features the elimination of all check valves. Shuttle experience has shown that the check valves do not always accomplish their goal of preventing vapors from migrating through the OMS system. In addition, ground operations efforts associated with the checkout of check valves are very expensive. For the PLS, check valves have been replaced with normal isolation valves. This allows a larger procurement of isolation valves, eliminates the small procurement of check valves and reduces the number of different checkout procedures imposed on the ground crew. The primary mode of operation for this system would specify closed tank isolation valves during on-orbit operations with an occasional opening for repressurization and OMS burns. This minimizes the pressurization system exposure to vapors.

The philosophy used in the biprop system is similar to the that of the monopropellant concept with one exception. In the monoprop system, tankage pressurant isolation was accomplished with a single isolation valve. A fail-open did not affect the system performance and the first fail-closed still left half of the propellant under pressure regulation and the other half with blowdown capability. If both tank isolation valves failed-closed there was blowdown on both tanks. In the biprop system, this is not acceptable since the first fail-closed would leave the whole system in a blowdown condition. For this reason, both pressurization systems in the biprop have the required fault tolerance.

The fail-op/fail-safe system shown in Figure 5-44, achieves two fault tolerance and continuous abort capability through dual redundant thrusters. During nominal operation, the two sides of the system are kept separated for faster fault isolation and to allow easier propellant quantity determination. In addition, the GN2 used for the cold gas thrusters is isolated from the pressurant GN2 to eliminate vapors from entering the cold gas system. If required, the opening of the isolation valve allows cold gas GN2 to pressurize the biprop system. The hardware for this concept is as follows:

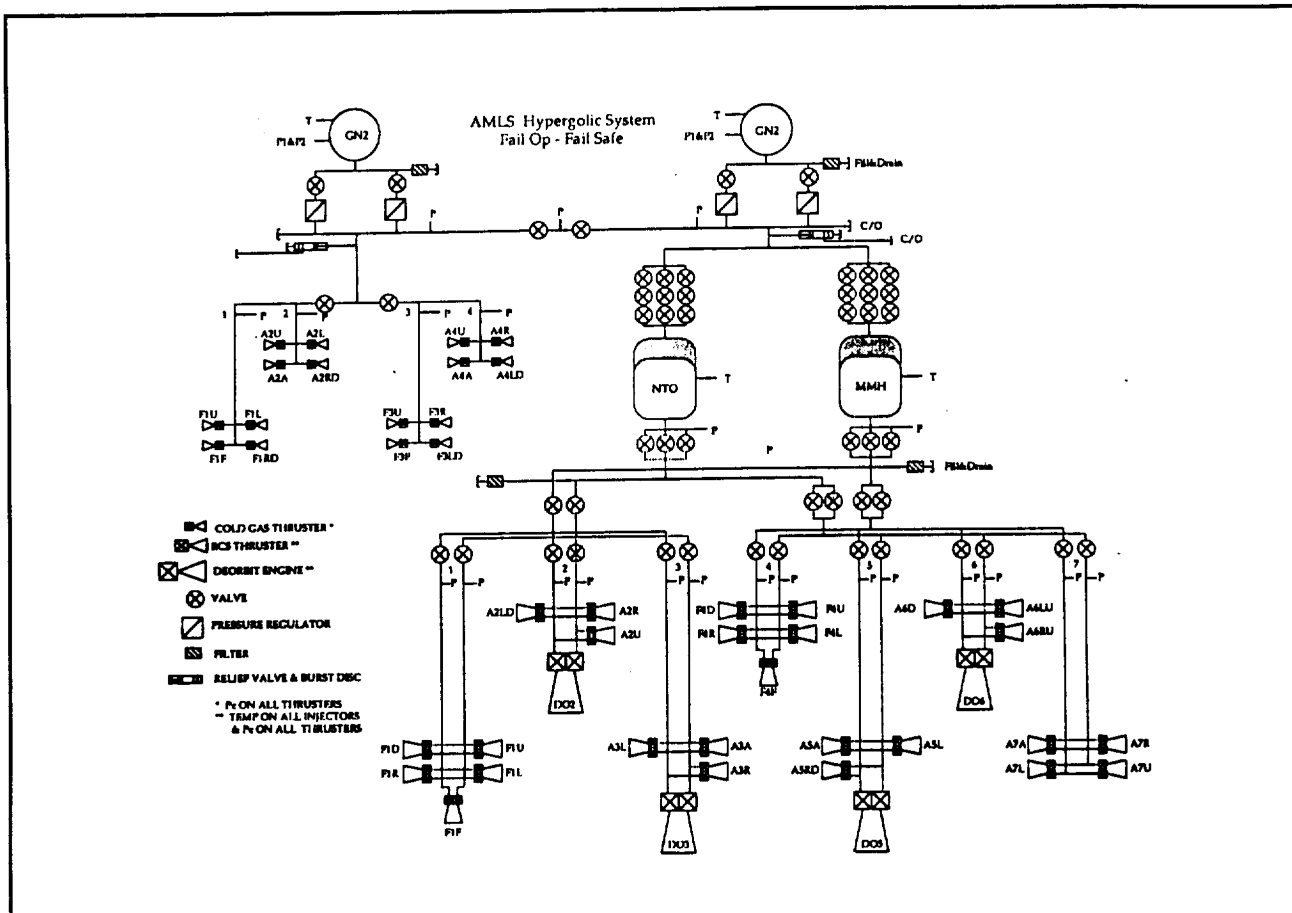


Figure 5-44. Bipropellant Fail-op/Fail-safe Propulsion System

<u>Component</u>	<u>Weight</u>	<u>Notes</u>
N2 Tanks (2)	200 pounds	4.9 cubic foot volume
N2O4 Tanks (2)	96 pounds	12.7 cubic foot volume
MMH Tanks (2)	96 pounds	12.7 cubic foot volume
Lines	43 pounds	5% of Total Weight
Regulators (4)	16 pounds	Existing Components
GN2 Iso (26)	54 pounds	Existing Components
N2O4 Iso (13)	52 pounds	Existing Components
MMH Iso (13)	52 pounds	Existing Components
Relief (2)	4 pounds	Existing Components
Fill/Drain (8)	8 pounds	Existing Components
RCS Prim. (26)	216 pounds	Apollo SM, 100 lbf Design
RCS Vern. (16)	1 pound	Existing Design

OMS (4)	74 pounds	Mariner/Viking, 300 lbf
GN2	186 pounds	
N2O4	1600 pounds	
MMH	1066 pounds	
Dry Weight	912 pounds	
Total Weight	3764 pounds	

Technological issues with the bipropellant concept are:

1. The bladders for the non-spherical tanks may be more prone to twist and leak than for spherical tanks. Metal bladders may not be reusable.

2. The identified thrusters have not been reused before and may not have the total impulse required for repeated missions.

JP4/H2O2 Propulsion Concepts.

Hydrogen peroxide propulsion systems were investigated because they offer low toxicity and reasonable performance and may also yield life cycle cost savings compared to conventional storables. Since the products of the catalytic breakdown of H2O2 are water and oxygen, hydrogen peroxide may also be used in a monopropellant mode for Space Station proximity operations. This allows the integration of both propulsion mode tanks (OMS and RCS) into a single propellant tank with one pressurization system. The peroxide system has been used safely and reliably on high performance manned vehicles, most notably, the X-15 (which used peroxide in the reaction control system) had 195 flights 25 years ago.

Thrust levels are determined by scaling present Shuttle thrust levels down to provide an equivalent thrust/weight ratio for the PLS. The assumed thrust levels are 1200 lbf (four engines) for the OMS and 80 lbf for each RCS engine and 2 lbf for each cold gas vernier engine. Propellant weights are determined using the rocket equation and the following performance assumptions based on analysis: an OMS engine Isp of 277 seconds and an RCS engine Isp of 250 seconds for biprop operation and 150 seconds for monoprop operation. It is assumed that the starting mass for the prox-ops system is the initial wet weight less 25% of the OMS and RCS load since, before cold gas system initiation, the vehicle must have achieved orbit and maneuvered to the Space Station. All propellant weights in the hardware list include a 15% margin for off-nominal performance, GN&C errors, etc.

Tank sizes are determined using the density of JP4/H2O2 at 68 degrees F with 25% volume added for ullage. This provides a blowdown capability in the event of a pressurization system failure. The GN2 tanks are sized based on a maximum operating pressure of 4000 psi (current Shuttle RCS tank design).

Due to the large mixture ratio of 7 for this propellant combination and the resulting difference in propellant volumes, each side of the PLS vehicle is provided with a fuel tank and an oxidizer tank. For this reason, three-by-three tank isolation valves are not needed. Two failures will, at most, isolate one tank, leaving one tank pressurized for the deorbit burn.

Three versions of the hydrogen peroxide concepts are evaluated to define the operational or cost benefits of different RCS designs: 1) dual mode thrusters, 2) separate mono- and biprop RCS thrusters and 3) mono- only thrusters).

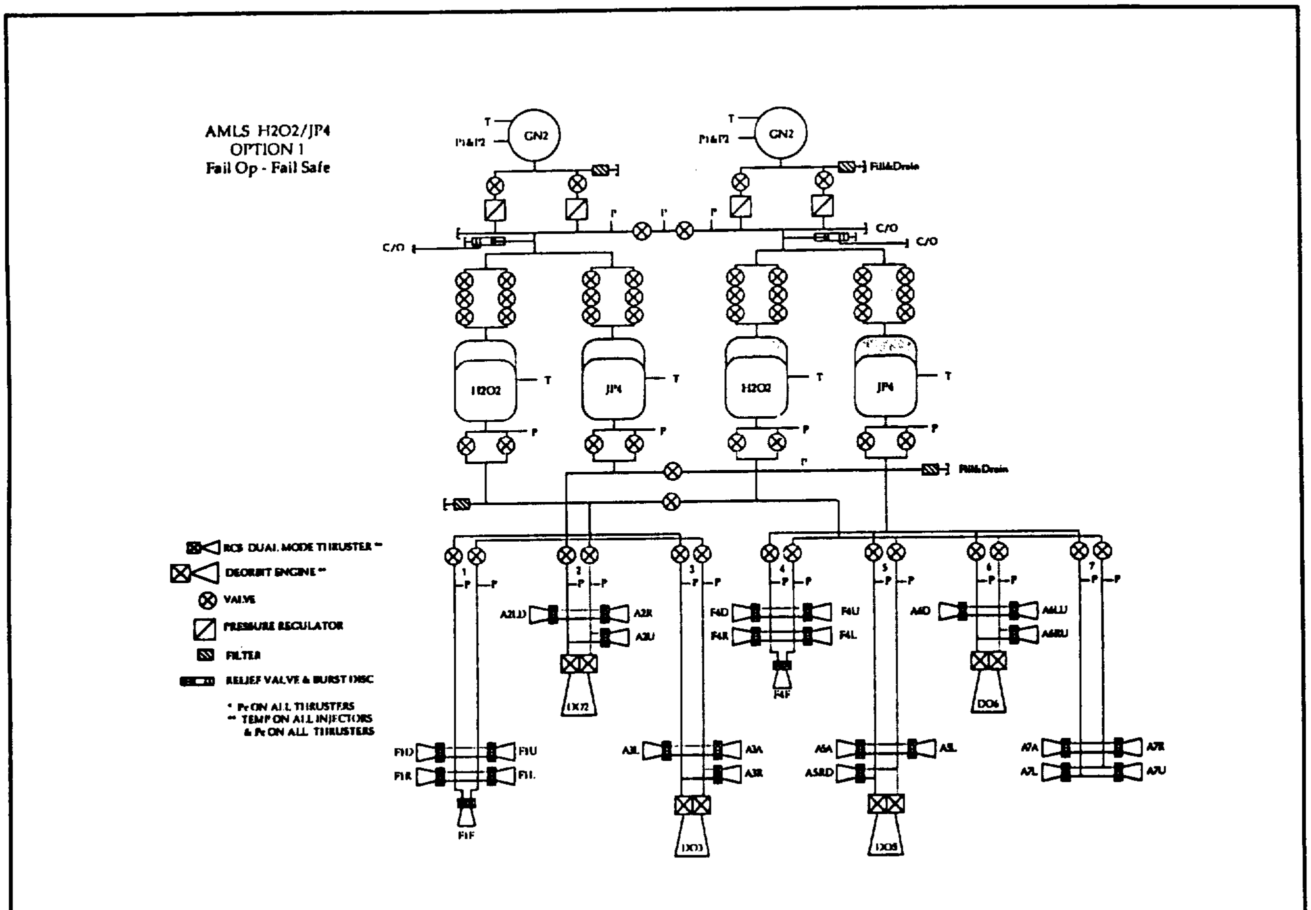


Figure 5-45. Hydrogen Peroxide Fail-op/Fail-safe Propulsion System

The JP4/H2O2 Option 1 (Figure 5-45) concept achieves two fault tolerance and continuous abort capability through dual redundant feed paths for both pressurant and propellant and through redundant thrusters. During normal operation, the two sides of the system are separated for faster fault isolation and for easier propellant quantity determination. This concept features an RCS thruster design which operates in a bipropellant mode for on-orbit operations and in a monoprop mode for proximity operations. The advantages of this approach are low weight and a reduction in the number of thrusters needed. The disadvantage is higher development costs since the engine must be developed and tested in both modes.

The hardware for this option is as follows:

Component	Weight	Notes
N2 Tanks (2)	87 pounds	2 cubic foot volume
JP4 Tanks (2)	88 pounds	4 cubic foot volume
H2O2 Tanks (2)	320 pounds	19.9 cubic foot volume
Lines	52 pounds	8% of Total Weight
Regulators (4)	16 pounds	Existing Components
GN2 Iso (30)	62 pounds	Existing Components
Prop Iso (24)	96 pounds	Existing Components
Relief (2)	4 pounds	Existing Components
Fill/Drain (8)	8 pounds	Existing Components
RCS (26)	216 pounds	New Design
OMS (4)	74 pounds	New Design
GN2	72 pounds	
JP4	321 pounds	
H2O2	2770 pounds	
Dry Weight	1023 pounds	
Total Weight	4186 pounds	

The JP4/H2O2 Option 2 concept (Figure 5-46) replaces the dual-mode RCS thrusters of Option 1 with separate thrusters for the biprop and monoprop RCS functions to avoid the thruster development issue. The hardware for this option is as follows:

Component	Weight	Notes
N2 Tanks (2)	87 pounds	2 cubic foot volume
JP4 Tanks (2)	88 pounds	4 cubic foot volume
H2O2 Tanks (2)	320 pounds	19.9 cubic foot volume
Lines	55 pounds	8% of Total Weight
Regulators (4)	16 pounds	Existing Components
GN2 Iso (30)	62 pounds	Existing Components
Prop Iso (24)	96 pounds	Existing Components
Relief (2)	4 pounds	Existing Components
Fill/Drain (8)	8 pounds	Existing Components
RCS Prim. (26)	216 pounds	New Design
RSC Vern. (16)	32 pounds	New Design
OMS (4)	74 pounds	New Design
GN2	72 pounds	
JP4	321 pounds	
H2O2	2770 pounds	
Dry Weight	1057 pounds	
Total Weight	4220 pounds	

The JP4/H2O2 Option 3 concept (Figure 5-47) uses RSC thrusters which operate in a monoprop mode at all times. This creates a simpler system since fewer valves are required to

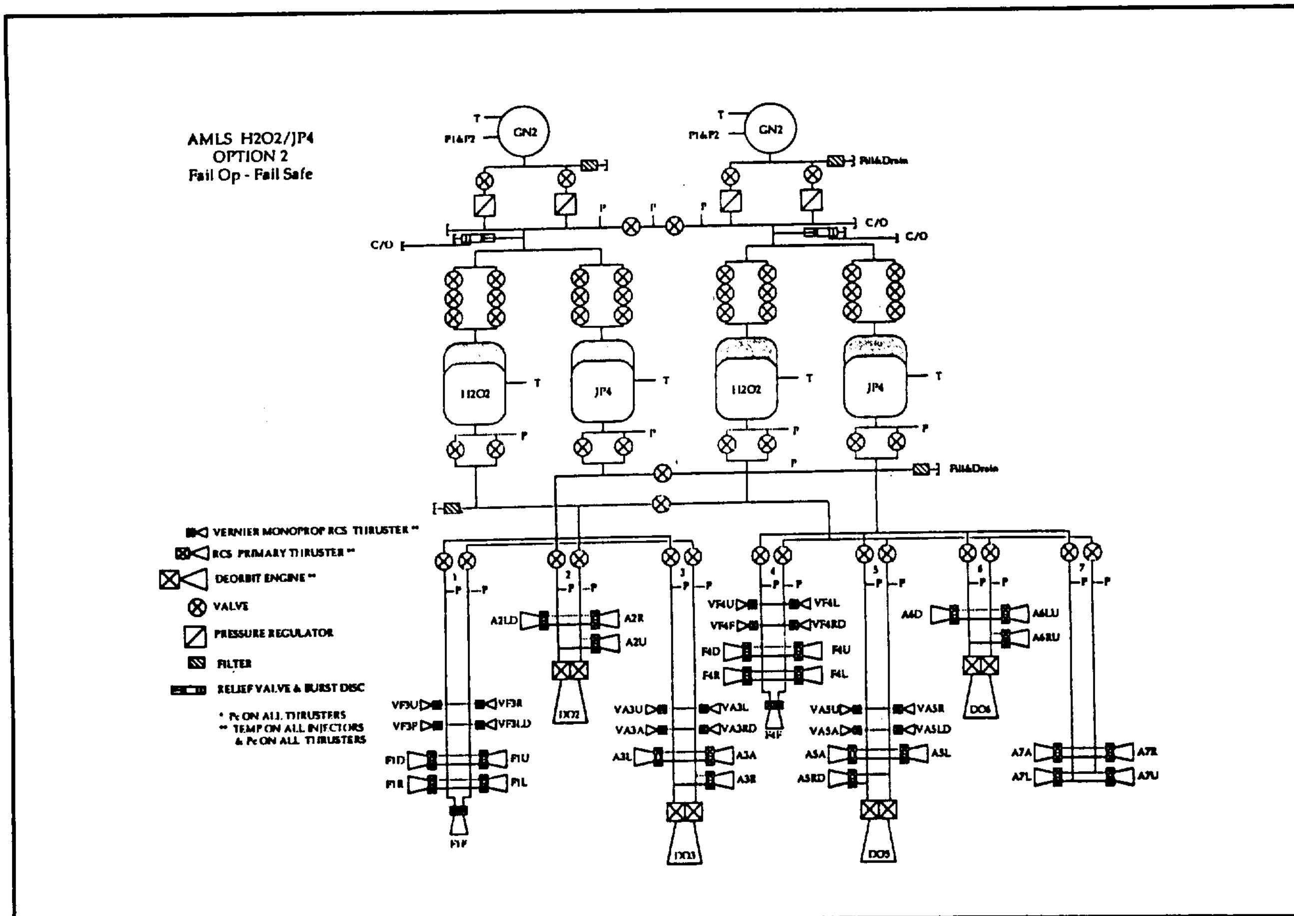


Figure 5-46. Hydrogen Peroxide Propulsion System with Separate Thrusters

feed the RCS thrusters. The decreased performance of the monopropellant thrusters results in a weight penalty of about 200 pounds in additional propellant.

The hardware for this option is as follows:

Component	Weight	Notes
N2 Tanks (2)	90 pounds	2 cubic foot volume
JP4 Tanks (2)	88 pounds	4 cubic foot volume
H2O2 Tanks (2)	320 pounds	19.9 cubic foot volume
Lines	54 pounds	8% of Total Weight
Regulators (4)	16 pounds	Existing Components
GN2 Iso (30)	62 pounds	Existing Components
Prop Iso (21)	84 pounds	Existing Components
Relief (2)	4 pounds	Existing Components
Fill/Drain (8)	8 pounds	Existing Components
RCS Prim. (26)	216 pounds	New Design
RCS Vern. (16)	32 pounds	New Design
OMS (4)	74 pounds	New Design
GN2	72 pounds	
JP4	340 pounds	
H2O2	2920 pounds	

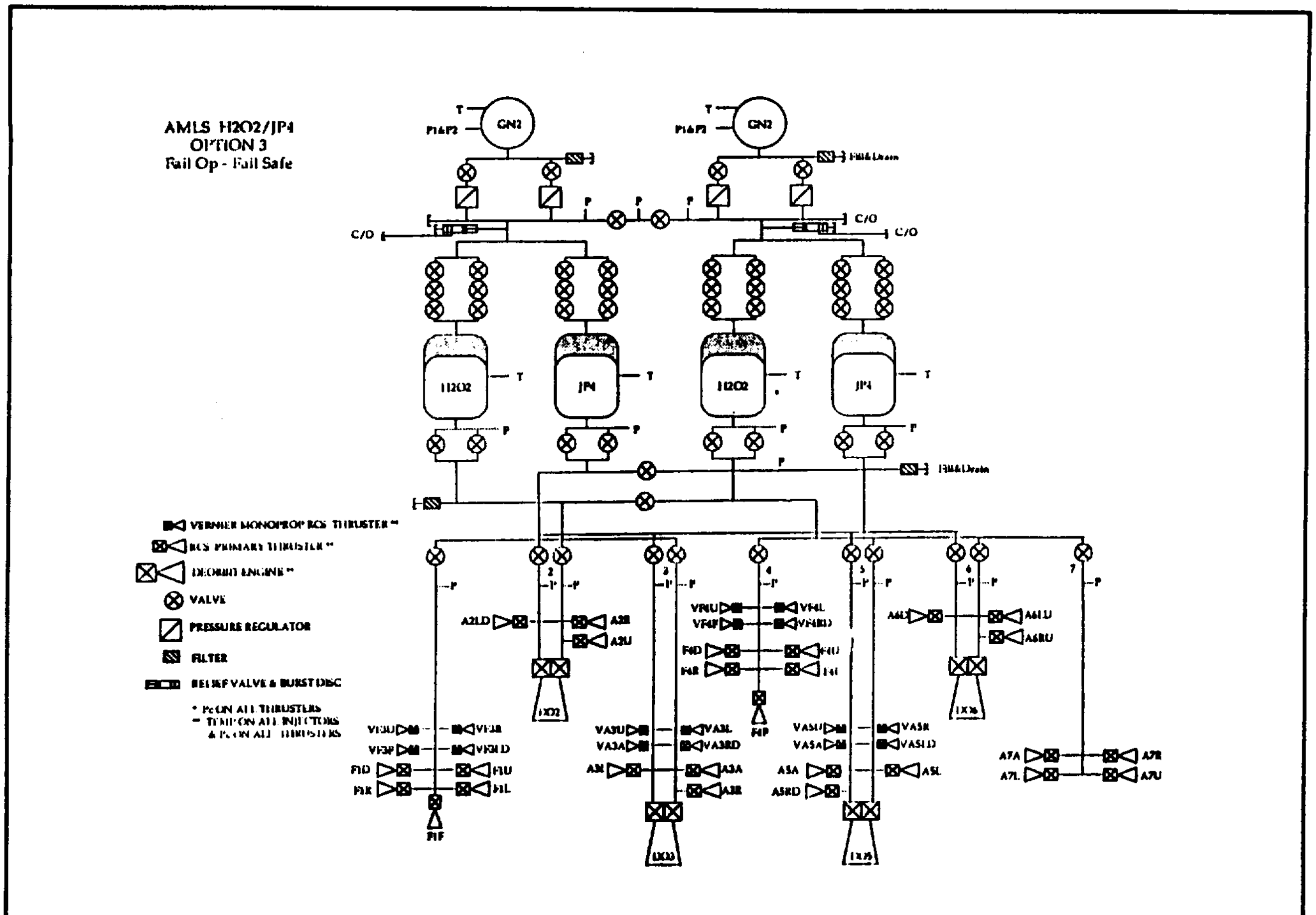


Figure 5-47. Hydrogen Peroxide Propulsion with Monoprop RCS Thrusters

Dry Weight	1048 pounds
Total Weight	4380 pounds

Technological issues associated with the hydrogen peroxide concepts are:

1. The bladders for the non-spherical tanks may be more prone to twist and leak than for spherical tanks. Metal bladders may not be reusable.

2. The identified thrusters have not been reused before and may not have the total impulse capability required for repeated missions.

3. Hydrogen peroxide has been known to detonate when stored in a closed container. In a clean system using proper materials and temperature control this should not be a problem.

Cryogenic Propulsion Concepts (Methane/LOX).

A cryogenic propellant combination is included to assess its suitability for the PLS mission and to identify any advantages over the storable propellants previously investigated.

Since hydrogen is roughly 25% more efficient than its nearest cryogenic competitor, it is investigated first. Using the rocket equation and assuming an Isp of 420 seconds, the hydrogen and oxygen needed to perform the PLS OMS function is calculated to be 1675 pounds. This represents a propellant weight savings of about 1200 pounds over the lightest storable concept. Unfortunately, the low density of hydrogen dictates that 64 cubic feet is required to store the hydrogen needed for just the OMS burns. This is nearly twice the volume needed by the entire propellant systems of the other propellant combinations. The storage of hydrogen in solid hydrides was deemed to advanced for the PLS program groundrules. For these reasons, a hydrogen-based system is not investigated further.

A better cryogenic option is Methane/LOX due to its relatively high specific impulse of 320 seconds and a methane density of six times that of hydrogen. The lower carbon content of methane reduces the coking concerns associated with other hydrocarbon fuels such as propane. The OMS propellant weight is calculated to be 2525 pounds which is equivalent to the biprop, MMH/N2O4 concept. System volume requirements were roughly the same as for the reference hydrazine system.

In trying to develop a system schematic for the Methane/LOX system, a number of technical issues are raised. While the OMS system was relatively straight forward, a cryogenic RCS is complex. An all-cryogenic RCS thruster capable of multiple, short pulses has never been developed. The greatest problem in developing this thruster is heat soak-back to the propellant feed lines and the subsequent creation of high pressure gas in the lines. Since this development is significant, liquid thrusters are not considered feasible within the PLS program groundrules. Gaseous methane/oxygen thrusters have been developed and test fired although none have been flown. Unfortunately, there is not enough volume in the PLS vehicle to store the entire quantity needed for OMS burns and all of the RCS maneuvers. The solution to this problem is a gas generator, accumulator and heat exchanger to produce the gaseous propellant. However, this solution itself presents a number of technical issues and adds complexity and hardware to the system.

The conclusion reached in the investigation of cryogenic propulsion options is that none of the cryogenic concepts will meet the mission objectives of the PLS within volume, operations of technology constraints, although such systems offer significant weight savings over the storable alternates.

<u>Propulsion Concept Trade Conclusions.</u> The preferred propulsion system concept for PLS is the JP4/hydrogen peroxide alternate which uses the RCS in a monoprop mode (Option 3, Figure

5-47). The low toxicity of this propellant combination makes it a preferred system for PLS since it has the potential to reduce operations costs and concerns. The concept also has reasonably good performance capability with system weight and complexity falling between the MMH/N2O4 and reference hydrazine concepts.

5.1.9 Prime Power and Electrical Distribution System

The prime power and electrical distribution system consists of the battery, distribution, and power control systems.

The power source is primarily 28 Vdc battery power capable of sustaining approximately 3 kw total peak load with adequate redundancy for automatic load control and provision for automatic priority and load shedding control. Embedded circuitry is provided for continuous BITE testing supports fault detection/fault isolation/self healing. A separate, redundant 28 Vdc power source provides power for the pyrotechnic devices. Power conversion from 28 Vdc to 115 Vac, 400 cycle can either be centrally provided by a dedicated inverter or by built-in circuitry at the using element (preferred).

The preferred prime power concept consists of a three-bus, 28Vdc, all-battery system. The tri-bus has a left-hand bus (Bus 1), a center bus (Bus 2) and a right-hand bus (Bus 3) with provisions for auto-crossover and bus-split between Bus 1 and Bus 2. Bus 1 is the primary power source for the using elements in the external, left-hand subsystem compartments, Bus 2 supports the using elements inside the crew cabin and Bus 3 is the primary source for the using elements in the right-hand external compartments.

The following are required LRUs:

1. An internally redundant bus control unit (BCU) to provide automatic bus load control, priority control and load shedding.

2. An internally redundant converter to provide 270 Vdc to using elements.

3. An internally redundant static inverter to provide 115 Vac, 400 cycle if required.

The cockpit control remote control circuit breakers (RCCBs) are located in the vicinity of the batteries.

Power History Development. A detailed power-use history of the PLS vehicle in performing its nominal DRM-1 mission is needed to size the power generation system and to determine the required heat rejection capacity of the environmental control system. The following analysis discusses the assumptions and processes that were used to develop the power requirements for the 72-hour Space Station crew rotation mission (DRM-1).

Mission operation assumptions were made to aid the development of the PLS power consumption schedule and to determine LRU and subsystem component duty cycles:

1. The PLS 72-Hour Mission Timeline will be followed.

2. The PLS power history will include an allowance of ten minutes of on-board power (independent of the overall power system design margin) prior to launch to support a launch hold after GSE disconnect.

3. When the controllers are not required to operate aerodynamic control surfaces during the mission, they will be turned off. Controllers will be turned on 1.5 hours before use to allow adequate time for warmup and fault detection/resolution. Additionally, controllers will be activated during the final PLS checkout activity when docked to Space Station.

4. The PLS ECLS system will run at 100% when crew members are onboard to provide necessary CO2, humidity and heat removal. During periods of quiescence (when docked at Space Station Freedom), the system is not required to operate. The temperature control system will not need to run at 100% to remove the additional heat produced during the PLS checkout periods due to the long thermal transients of the vehicle. This additional heat, which will raise the cabin temperature only a few degrees, can be dissipated during the subsequent quiescent time.

5. During the final PLS checkout all systems will be activated to provide assurance that all systems are functioning properly. This allows the fault detection and resolution process to function prior to separation from the Space Station.

6. The PLS guidelines specify that the PLS shall not impact SSF operations; however, it is unrealistic to expect the ten arriving PLS personnel to confine themselves to the PLS volume after docking to the Space Station is accomplished. It is assumed that for the sleep periods within the 72-hour mission, the new crew would choose to occupy the larger SSF volume rather than return to the PLS. Therefore, the crew is assumed to live (eat, breathe, generate heat, etc.) entirely onboard SSF while the PLS is docked.

7. Propellant and ECLSS valves require power only when they are cycled. Redundant valves are used to open lines after valve failures occur.

8. The JIAWG modules in the PLS computers are used to compute the power consumption figures for data processing and health monitoring. (Alternate computer

configurations exist and may be substitutes for this assumed system).

9. A maximum amount of passive cooling and allowance for temperature variations is assumed to be incorporated in the PLS design. This eliminates the large power loads associated with maintaining tight control of component temperatures especially during quiescent periods.

10. The power history does not reflect the self-induced power requirements of the power generating system itself. Since the purpose of the prime power trade study is to make that system selection, the power history impacts of that selection appear in the prime power trade study report (See Reference 5-1).

11. Conservative estimates were made for power consumption and duty cycles. Therefore, power and energy estimates tend to be too high rather than too low. This approach reduces the growth margin as subsystems become more precisely defined. The estimated growth margin is approximately 15%, which should accommodate additional thermal control (cabin fans, heaters and heat rejection power) and actual hardware power requirements.

The list of PLS power users and duty cycles was developed using the "reference" PLS subsystem description. Where applicable, "off-the-shelf" LRU technical data were used to increase the accuracy of estimates for power consumption and thermal environment requirements. Technical data were obtained from the following list of suppliers for the indicated LRUs:

Supplier	LRU
Boeing	OMS Thrusters
Draper Laboratories	Main and Backup Computers
Garrett-AiResearch	Actuators
Hamilton Standard	ECLSS, Sensors
JIAWG	Avionics Interfaces
Loral Braking Systems	Electric Brakes
Marquardt	RCS Thrusters
Moog	GN2 Cold Gas Thrusters
Rockwell (Autonetics)	GPS Receiver
Rockwell (Collins)	Communications, Instrumentation

Power and energy estimates are generated for each mission event on the PLS timeline (Figure 5-49) using the subsystem power consumption and duty cycle data. The estimates are grouped by voltage requirement, 28 Vdc or 270 Vdc, and are shown in Figure 5-50. Loads (power and energy) at each voltage and the total of both are provided for each mission event. Energy load totals are also provided for each mission phase and for the entire mission.

PHASE	Event(s)	Time(hrs)
ASCENT	Pre-launch	0.17
	Launch	1.0
	OMS	0.2
TOTAL		1.37
ORBIT	Rendezvous & Prox Ops	11.5
	PLS Checkout	1.0
	Downtime	18.5
	PLS Checkout	0.5
	Downtime	11.5
	PLS Checkout	0.5
	Downtime	4.5
	PLS Stow & Checkout	3.5
	Downtime	11.0
	Undock Prep.	1.5
	Separation	1.0
TOTAL		65.0
DESCENT	Deorbit Prep.	2.0
	Deorbit Burn	0.1
	Exo Entry	0.5
	Atm Entry	0.5
	Landing	0.01
	Recovery	1.0
TOTAL		4.11
MISSION GRAND TOTAL		70.48

Figure 5-49. Estimated Power and Energy Requirements

The resulting PLS power history timeline for DRM-1 is shown in Figure 5-51. The data reveal sharp rises in power consumption during orbital maneuvers, atmospheric reentry and especially during the landing phase where worst case assumptions are made. These high peak electrical loads may influence the results of the prime power trade study due to the large deviation from nominal operating loads. A secondary high power rate system may be necessary to augment the main power supply.

The very low power consumption during the downtime at SSF is achieved by having a backup health monitoring system to check the status of the critical elements (propellant tanks, batteries, etc.) and an independent temperature regulator to sense LRU temperatures and to turn on heaters when needed. Thus, the main avionics and thermal control systems may be off for most of the time.

ASCENT	Time (hr)	Power (kW)			Energy (kW-hr)		
		28 Vdc	270 Vdc	Total	28 Vdc	270 Vdc	Total
Pre-Launch	0.17	3.1	0.8	3.9	0.5	0.1	0.6
Launch	1.00	3.0	0.8	3.8	3.0	0.8	3.8
OMS	0.20	2.7	2.8	5.5	0.5	0.6	1.1
TOTALS	1.37				4.0	1.5	5.5
ORBIT	Time (hr)	Power (kW)			Energy (kW-hr)		
		28 Vdc	270 Vdc	Total	28 Vdc	270 Vdc	Total
Rendezvous	11.50	2.2	0.8	3.0	25.3	9.2	34.5
PLS Checkout	1.00	2.1	0.8	2.9	2.1	0.8	2.9
Downtime	18.50	0.6	0.0	0.6	11.1	0.0	11.1
PLS Checkout	0.50	1.4	0.8	2.2	0.7	0.4	1.1
Downtime	11.50	0.6	0.0	0.6	6.9	0.0	6.9
PLS Checkout	0.50	1.4	0.8	2.2	0.7	0.4	1.1
Downtime	4.50	0.6	0.0	0.6	2.7	0.0	2.7
PLS Stow & Checkout	3.50	3.1	0.8	3.9	10.9	2.8	13.7
Downtime	11.00	0.6	0.0	0.6	6.6	0.0	6.6
Undock Prep	1.50	2.2	0.8	3.0	3.3	1.2	4.5
Separation	1.00	2.4	0.8	3.2	2.4	0.8	3.2
TOTALS	65.0				72.7	15.6	88.3
DESCENT	Time (hr)	Power (kW)			Energy (kW-hr)		
		28 Vdc	270 Vdc	Total	28 Vdc	270 Vdc	Total
Deorbit Prep	2.00	3.0	0.8	3.8	6.0	1.6	7.6
Deorbit Burn	0.10	3.2	2.8	6.0	0.3	0.3	0.6
Exo Entry	0.50	3.4	1.3	4.7	1.7	0.7	2.4
Atm Entry	0.50	2.2	5.4	7.6	1.1	2.7	3.8
Landing	0.01	2.2	8.5	10.7	0.0	0.1	0.1
Recovery	1.00	1.5	0.8	2.3	1.5	0.8	2.3
TOTALS	4.11				10.6	6.2	16.8
Totals For Mission	70.48				87.3	23.3	111.6

Figure 5-50. Power Estimates Grouped by Voltage Requirement

It is recommended that the Space Station Freedom operational guidelines should reflect the need to provide support for the two crews (old and new) during the personnel exchange. It is much easier for SSF to absorb the additional ECLSS load from the new crew than it is for the PLS to support them for the entire 72-hour mission duration. The PLS power requirements and mission costs increase dramatically if there is a requirement to fully support the new crew during docked operations for handoff.

<u>Alternate Prime Power Concepts.</u> The "reference" PLS prime power system is an all-DC/battery system to support the program objective of efficient operations. This concept, which was made possible by the low power consuming nature of the PLS DRM-1 mission, reduces the procurement and maintenance costs associated with more complicated power generation systems like fuel cells. The "reference" prime power concept for PLS is a non-rechargeable set of lithium thionyl chloride batteries. This concept has a high energy density but is also very reactive and poses a potential

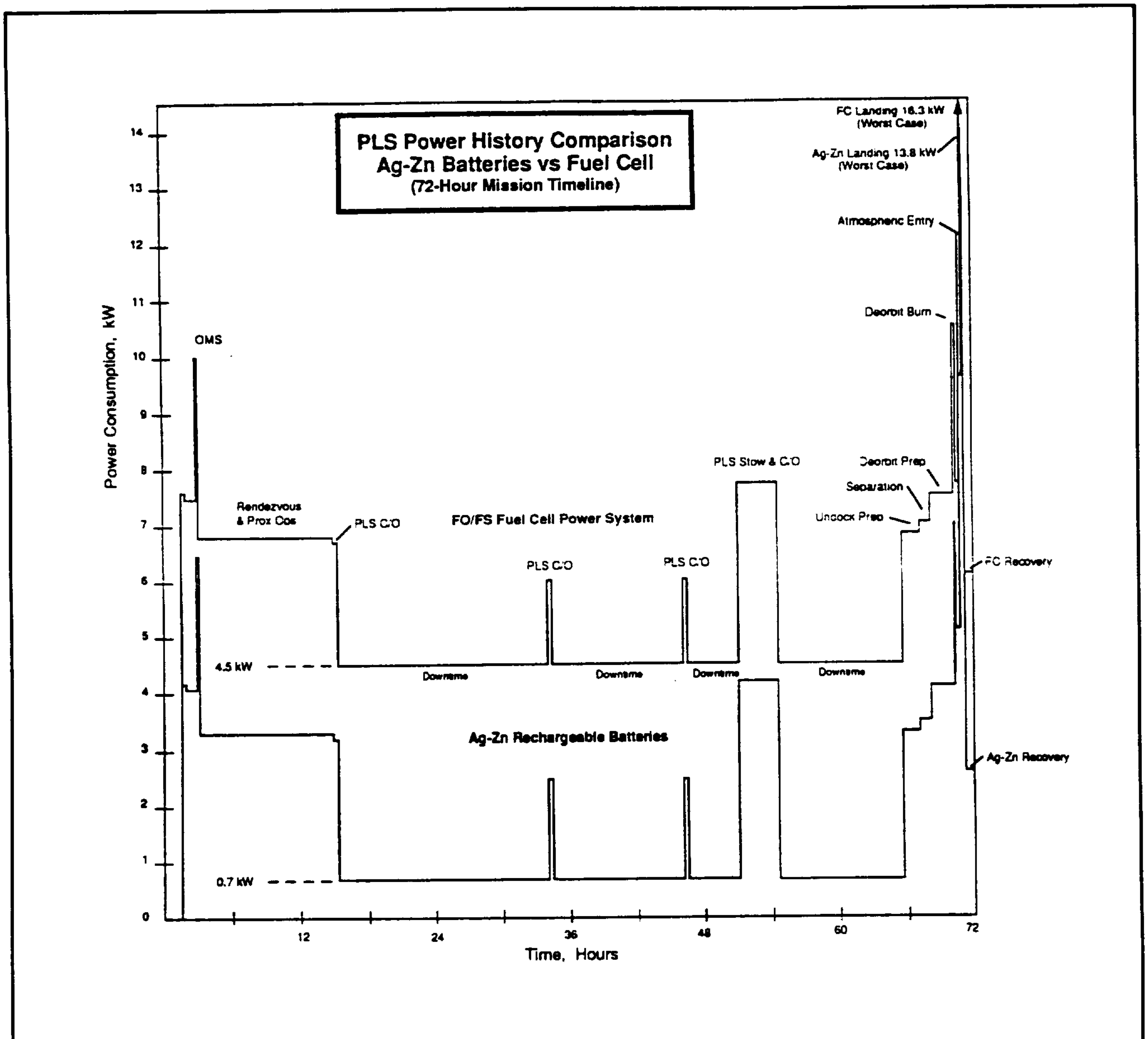


Figure 5-51. Estimated PLS Power Consumption History

hazard to ground personnel. Since it is not rechargeable, it represents a recurring flight cost. This trade study was performed to determine what prime power options are available to support the PLS program goals of low operations cost and efficient operations and to quantify the relative desirability of these options.

All of the optional power systems traded were required to meet the basic PLS mission power expenditure history shown in Figure 5-51. While the average rate of power consumption is relatively low, the high energy peak requirements at the end of the flight when all of the auxiliary systems are operating, must also be met.

For the chemical batteries, a number of options are available for consideration. Figure 5-52 is a trade tree which shows some of the options that could be investigated. The group is divided into

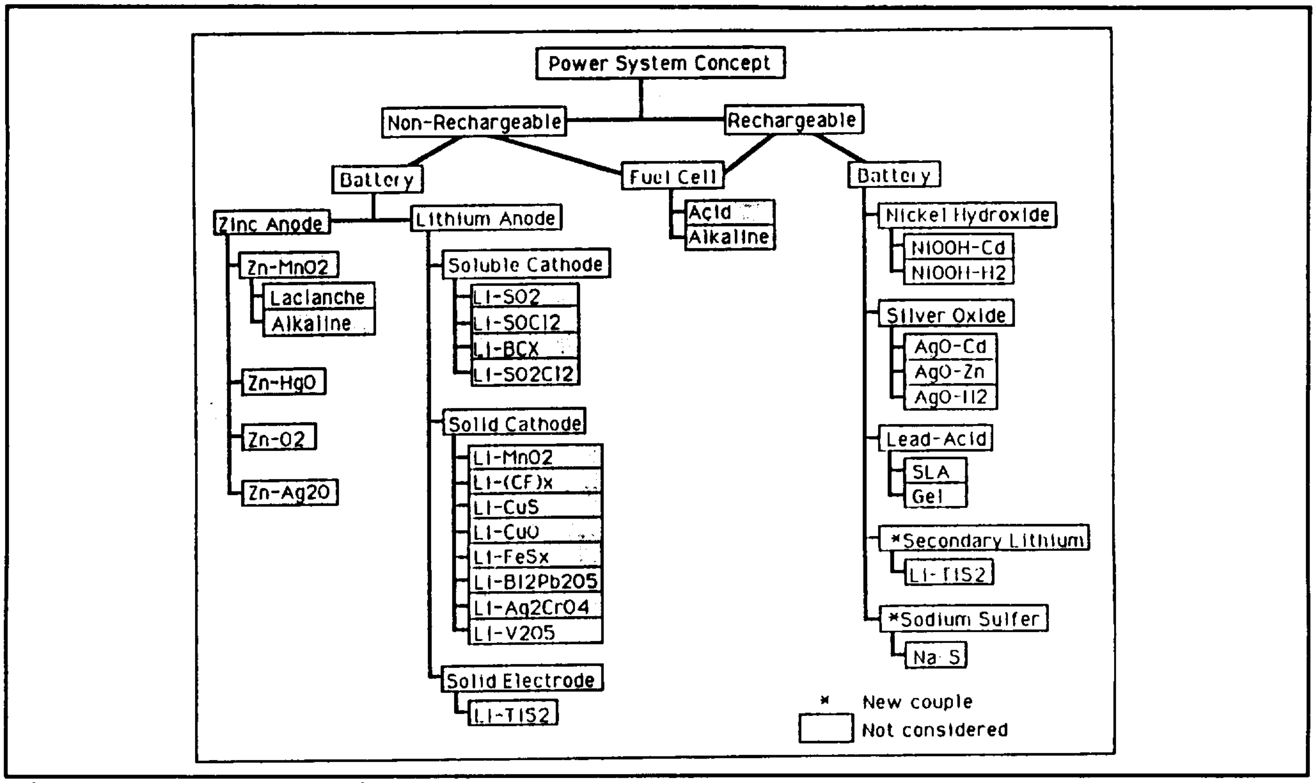


Figure 5-52. Chemical Battery Trade Tree

rechargeable and non-rechargeable categories and includes a high performance fuel cell. The unshaded blocks in the figure identify the concepts, representative of the different chemistries available, which were included in this trade study evaluation. Data on the various concepts were obtained from recent STS Orbiter upgrade studies and from industry contacts, principally from Eagle-Picher and the Jet Propulsion Laboratory.

Traditionally power density has dominated the evaluation of competing power systems for spacecraft applications because of the importance of minimum weight. For PLS, other characteristics are increasingly important such as initial cost and servicing requirements. The following parameters were used to trade the various prime power system candidates: energy density, power density, weight, volume, safety, technological risk, temperature range, refurbishment process, survivability, storage life, procurement time and subsystem cost. For each of these parameters, a scale from one to ten was defined based on quantifiable attributes. This rating definition is presented in Table 5-4.

Using this table, each of the candidate power systems was scored on each of the parameters. The results of this scoring process are presented in Table 5-5. Finally, a weighting system was devised to assign relative importance to each of the prime power concept parameters. As shown in Table 5-6, this weighting system places the highest value on safety to simplify vehicle ground operations. Weight (energy density), ruggedness and cost are almost as important.

Table 5-4. Ratings for Power System Candidates

PARAMETER	NAME	MEANING OF SCALE VALUES 10	9	8	7	6	5	4	3	2	1
Subsystem Energy Density	EDENS	220 W-hr/lb (or more)	180 W-hr/lb (up to)	140 W-hr/lb (up to)	100 W-hr/lb (up to)	80 W-hr/lb (up to)	60 W-hr/lb (up to)	40 W-hr/lb (up to)	30 W-hr/lb (up to)	20 W-hr/lb (up to)	10 W-hr/lb (up to)
Subsystem Power Density	PDENS	500 W/lb (or more	400 W/lb (up to)	300 W/lb (up to)	200 W/lb (up to)	100 W/lb (up to)	80 W/lb (up to)	60 W/lb (up to)	40 W/lb (up to)	20 W/lb (up to)	10 W/lb (up to)
Subsystem Weight	WT	1,000 lb (up to)	1,500 lb (up to)	2,000 lb (up to)	2,500 lb (up to)	3,000 lb (up to)	3,500 lb (up to)	4,000 lb (up to)	4,500 lb (up to)	5,000 lb (up to)	5,500 lb (or more)
Subsystem Volume Requirement	VOL	6 cu-ft (up to)	8 cu-ft (up to)	10 cu-ft (up to)	12 cu-ft (up to)	14 cu-ft (up to)	16 cu-ft (up to)	18 cu-ft (up to)	20 cu-ft (up to)	22 cu-ft (up to)	24 cu-ft (or more)
Hazard to Personnel	HZRD	little or no problem	•	•	•	•	problem definitely manageable	•	•	•	potentially manageable problem
Technology Risk Factor	TECR	flown	Already flown in similar applic..	•	Qualified for space flight	•	Lab-tested devel.. system	•	Developmt nearing test phases	•	Technology Developmt
Subsystem Temperature Range Capability	THML	little or no problem	•	•	•	•	manageable problem with minor changes	•	•	•	manageable problem with major changes
Ease of Refurbishing Subsystem	REFB	no refurb.	•	•	•	•	minor refurb.	•	•	•	40 hrs or more refurb.
Subsystem Random Vibration Survivability	RUGD	Survive as supplied	•	•	•	•	Survive as repkgd (minor developmt)	•	•	•	Survive as repkgd (major developmt)
Subsystem Active Storage Life	SHLF	20 yrs (or more)	10 yrs (up to)	2 yrs (up to)	1 yr (up to)	1/2 yr (up to)	1 mo. (up to)	days (up to)	1 day (up to)	1 hr (up to)	1 min (up to)
Delay in Procurement	AVAIL	Off the shelf & qualified	1.5 yr including qualify	•	•	•	2.5 yr incl qualify (minor develpmt)	•	•	•	4.5 yr incl qualify (major develpmt)
Subsystem Cost	$	$50k (up to)	$100k (up to)	$200k (up to)	$400k (up to)	$600k (up to)	$800k (up to)	$1M (up to)	$2M (up to)	$3M (up to)	$5M (or more)

The raw numbers calculated from the scoring and weighting system described above showed that the Silver-Zinc battery option is preferred for use on PLS. The Ag-Zn concept is desirable from its low cost and ease of handling. The Ni-Cd battery concept is a close second but it has peculiar current supply characteristics that are less desirable. The Fuel Cell option scored very poorly because of its high cost and operations requirements.

Table 5-7 presents the trade study results in terms of the percentage each concept ranked in first through tenth place. The conclusion is the same; that given the relatively low power consumption requirements of DRM-1, and the desire for efficient operations, the preferred prime power concept is the rechargeable Ag-Zn battery system.

Preferred Prime Power Concept. Data on existing Ag-Zn cells were obtained from Eagle-Picher. An applicable battery pack design was developed for the PLS power requirements.

Table 5-6. Weighting System for Power System Trades

PARAMETER	NAME	RANKED IMPORTANCE (1 to 10)	PARAMETER DEFINITION - HIGH IMPORTANCE CONNOTES AS FOLLOWS:
Subsystem Energy Density	EDENS	8	Subsystem Delivers Power Over a Long Time for the Weight
Subsystem Power Density	PDENS	5	High Subsystem Power Levels for the Subsystem Weight
Subsystem Weight	WT	9	Low Subsystem Weight
Volume Requirement	VOL	6	Low Subsystem Volume Requirement
Hazard to Personnel	HZRD	10	Low Hazard to Personnel
Technology Risk Factor	TECR	2	Low Technology Risk Factor
Temperature Range Capability	THML	6	Wide Temperature Range Capability
Ease of Refurbishing	REFB	4	Little or No Refurbishing
Random Vibration Survivability	RUGD	7	Random Vibration Survivability With No Repackaging
Active Storage Life	SHLF	1	Long Active Storage Life
Delay in Procurement	AVAIL	3	Little Delay In Procurement
Subsystem Cost	$	7	Low Subsystem Cost

Both single and dual fault tolerant power system options were defined for PLS for different numbers (and sizes) of battery packs. The concept chosen for the preferred PLS power system is one with FO/FO/FS redundancy and eight battery packs which weighs 2720 pounds. Six of the battery packs are required to meet the DRM-1 power requirements with the remaining two providing the dual fault tolerant capability.

Figure 5-53 shows the location of the eight battery packs in the PLS vehicle (cross-hatch). Note that the four packs at midships balance a nearly identical mass of parachutes on the opposite side of the vehicle.

Table 5-7. Power System Trade Study Results

Trade Study--Unweighted and Weighted Parameter Influences.

Results	SDI Fuel Cell	Li-SOCl2	Ag-Zn	Ni-H2	Ag-H2	Ag-Cd	Ni-Cd	Pb-H2SO4	*Li-TiS2	*Na-S
Unweighted	9th	5th	2nd	3rd	6th	4th	1st	7th	8th	10th
Weighted	9th	5th	1st	4th	6th	3rd	2nd	8th	7th	10th

Sensitivity Study--Emphasis on Parameters Singlely and in Pairs.

Place (%)	SDI Fuel Cell	Li-SOCl2	Ag-Zn	Ni-H2	Ag-H2	Ag-Cd	Ni-Cd	Pb-H2SO4	*Li-TiS2	*Na-S
1st	1.25	21.25	36.25	0.00	0.00	6.25	27.50	7.50	0.00	0.00
2nd	5.00	10.00	22.50	6.25	1.25	16.25	33.75	3.75	1.25	0.00
3rd	3.75	1.25	15.00	27.50	3.75	33.75	7.50	3.75	3.75	0.00
4th	0.00	12.50	2.50	26.25	18.75	21.25	5.00	0.00	13.75	0.00
5th	0.00	16.25	5.00	15.00	18.75	15.00	7.50	12.50	8.75	1.25
6th	3.75	5.00	8.75	11.25	28.75	3.75	5.00	11.25	21.25	1.25
7th	13.75	8.75	8.75	10.00	17.50	0.00	6.25	10.00	23.75	1.25
8th	17.50	13.75	1.25	2.50	3.75	3.75	3.75	27.50	17.50	8.75
9th	36.25	11.25	0.00	0.00	1.25	0.00	2.50	6.25	10.00	32.50
10th	18.75	0.00	0.00	1.25	6.25	0.00	1.25	17.50	0.00	55.00

Place (%)	SDI Fuel Cell	Li-SOCl2	Ag-Zn	Ni-H2	Ag-H2	Ag-Cd	Ni-Cd	Pb-H2SO4	*Li-TiS2	*Na-S
1st-10th	-17.50	21.25	36.25	-1.25	-6.25	6.25	26.25	-10.00	0.00	-55.00
2nd-9th	-31.25	-1.25	22.50	6.25	0.00	16.25	31.25	-2.50	-8.75	-32.50
3rd-8th	-13.75	-12.50	[illegible]	25.00	0.00	30.00	3.75	-23.75	-13.75	-8.75
Total	-62.50	7.50	72.50	30.00	-6.25	52.50	61.25	-36.25	-22.50	-96.25

Trade results based on the previously discussed criteria show that the PLS should use silver-zinc (Ag-Zn) rechargeable batteries for the power supply.

* New couple

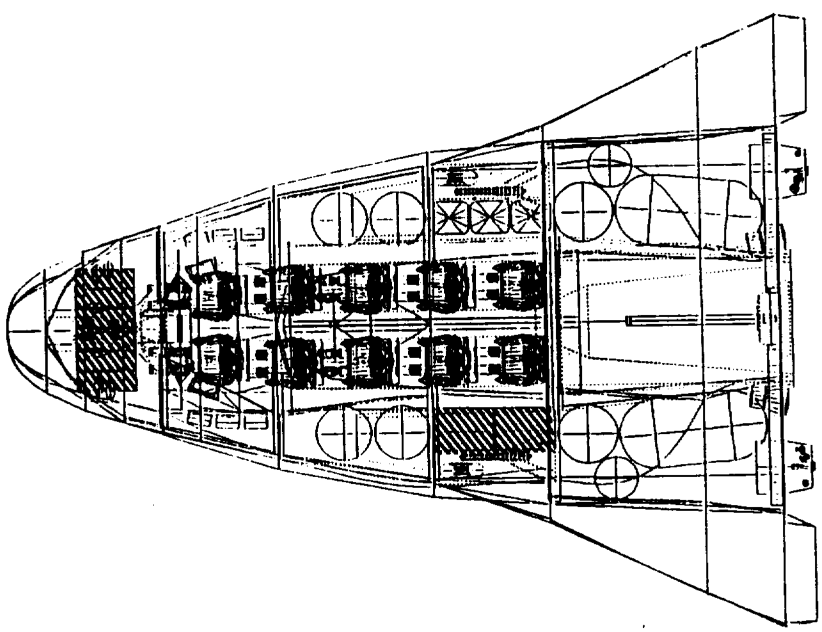


Figure 5-53. Battery Pack Location in the Glider

5.1.10 Avionics

The PLS avionics's five primary functions interface with each other to a total integrated and autonomous system that provides navigational autocontrol through all phases of flight. This capability includes complete uplink/downlink communication and telemetry, subsystem intercommunication, central health monitoring and maintenance analysis support. The avionics subsystem controls the operation of the flight control surfaces and nose/main landing gear in the auto-mode.

Alternate Avionics Architectures. The "reference" avionics architecture for PLS is based on the NASA LaRC commissioned efforts by Draper Laboratories to develop a fault tolerant system for manned Earth-to-orbit vehicles. This trade study activity defined alternative architectures and hardware selections to evaluate the resulting system characteristics. This section summarizes the fundamental differences between the alternate concepts in terms of fault tolerance and relative performance. Appendices in the trade study document (Reference 5-1) present the detailed diagrams of some of the alternate architectures either defined or evaluated in the trade study.

Several options for the PLS avionics architectures were considered. The study incorporated near-term advanced hardware with the system-level, integrated avionics approach initiated by the Pave Pillar program, to define architectures which support the low failure rates, high level of fault detection and isolation and health monitoring desired by the PLS program. Table 5-8 summarizes the characteristics of the primary concepts evaluated in the study. The appendices include definitions of additional architectures which were considered but which were not formally traded because of functionality differences.

The preceding architecture analyses are summarized in Table 5-9. Alternate 4B is selected as the preferred architecture for PLS because it provides fail-op/fail-op/fail-safe redundancy at a modest increase in weight and power consumption. The high performance ASCM computer modules are designed for fault tolerance and space applications and support the level of self test required by the PLS program.

Preferred Avionics Architecture. Figure 5-54 shows the block diagram of the preferred avionics architecture of PLS.

Guidance, Navigation And Control.

The guidance, navigation and control subsystem must generate and update a state vector with known local vertical alignment by autonomous methods during ascent, rendezvous with the Space Station, deorbit and throughout the entry and landing phases. The GN&C system provides stable control configuration and guidance solutions in all mission phases. The PLS concept uses a combination of GPS and an inertial measurement device of medium performance. A horizon scanner

Table 5-8. Summary Characteristics of Candidate Avionics Architectures

CONCEPT	SYSTEM	COMMENTS
REF	FAIL/OP FAIL/SAFE DRAPER W/NEWLY DEVELOPED AVIONICS	• SYSTEM BASED ON DRAPER BIT SYNC TRIPLE VOTING CONCEPT • SYSTEM NOT DEVELOPED OR PACKAGED - VERY HIGH DEVELOPMENT COSTS • SYSTEM WOULD BE UNIQUE THEREFORE VERY HIGH PRODUCTION COSTS • NO DEFINITION OF REMOTES GIVEN
1A	FAIL/OP FAIL/SAFE DRAPER W/JIAWG MODULES	• USE UNIQUE DRAPER TMR CPU's & JIAWG MODULES FOR EVERYTHING ELSE • REMOTES CONSOLIDATED TO REDUCE POWER & WEIGHT • 1553 BUS MAY HAVE THROUGHTPUT PROBLEMS • SOME HARDWARE & SOFTWARE NEW - MODERATE TO HIGH DEVELOP. COST • DRAPER CPU's UNIQUE - MODERATE PRODUCTION COST
2A	FAIL/OP FAIL/SAFE NASP DUAL CPU	• STANDARD JIAWG CARDS ARE SINGLE CPU W/SOFTWARE VOTE - UNISYS PROPOSES NEW DUAL CPU CARD • NEW CARD & NEW SOFTWARE - MODERATE TO HIGH DEVELOPMENT COST • UNIQUE CPU & SINGLE SOURCE - MODERATE PRODUCTION COST
2B	FAIL OP/FAIL OP/FAILSAFE - NASP DUAL CPU	• SAME COMMENTS AS 2A • FO/FO/FS PROBABLY REQUIRED FOR MANNED SPACECRAFT
3A	FAIL OP/FAIL SAFE - UPDATED SHUTTLE AVIONICS	• SOFTWARE VOTING W/NEW SOFTWARE - MOD TO HIGH DEVELOP. COST • UNIQUE EXPENSIVE HARDWARE - HIGH PRODUCTION COSTS • OLDER GENERATION HARDWARE W/HIGHER WEIGHT & POWER
4A	FAIL OP/FAIL SAFE - ADVANCED SPACEBORNE COMPUTER MODULE	• RADIATION HARD, SPACE QUALIFIED, DENSE PKG, LOWER POWER COMP. • FAULT TOLERANT DESIGN W/DUAL COMPUTERS & FAULT TOLERANT OP SYS • AF DEVELOPING CORE CARDS & OP SYS - LOW TO MED DEVELOP COST • MULTI PROGRAM CORE CARD USAGE - LOW TO MED PRODUCTION COST DEVELOP. COST DEPENDING ON PARTS RELIABILITY QUAL LEVEL
4B	FAIL OP/FAIL OP/FAILSAFE - ADV. SPACEBORNE COMPUTER MODULE	• FO/FO/FS PROBABLY REQUIRED • ACTUAL POWER MAY BE LOWER W/CLOCK OFF TO SPARES

is included to establish and maintain local vertical data. Redundant air data sensor assemblies account for relative velocity under variations due to winds. Aerodynamic control, steering, and braking while on the runway created the need for redundant interfacing electronics for each of the mechanical control elements.

The IMU uses state-of-the-art laser or fiber optic gyros, which use little power and have extremely low drift. The gyros are aligned prior to liftoff at rates commensurate with their low drift. The GPS receiver will update the position gyros at programmed intervals, or can be used as the sole position sensor if required.

In the event of an aborted launch and the possibility of a runway recovery, the PLS pilot has basic gyro and conventional pitot static system (a ram air turbine system could be used to generate contingency power) in addition to a radar altimeter and differential GPS navigation sets. The altitude and vehicle speed at abort initiates the landing maneuver from an onboard stored data.

Table 5-9. Avionics Architecture Analyses

CONCEPT	SYSTEM	POWER	WEIGHT	COMMENTS
REF	FAIL OP/FAIL SAFE - DRAPER W/NEWLY DEVELOP AVIONICS	NOT AVAILABLE	NOT AVAILABLE	COMPUTERS 324W & 90 LBS; REMOTES UNDEFINED
1A	FAIL OP/FAIL SAFE - DRAPER W/JIAWG MODULES	984W	129.1 LBS	CONSOLIDATED REMOTES
2A	FAIL OP/FAIL SAFE - NASP W/DUAL CPU (JIAWG) MODULES	910W	109.9 LBS	UNISYS CONCEPT FOR CPU
2B	FAIL OP/FAIL OP/FAIL SAFE - NASP W/DUAL CPU JIAWG MODULES	1551W	174.9LBS	UNISYS CONCEPT FOR CPU
3A	FAIL OP/FAIL SAFE - UPDATE SHUTTLE AVIONICS	2000W	379.5 LBS	ONLY 5 REMOTES IN ESTIMATE
4A	FAIL OP/FAIL SAFE - ADVANCED SPACEBORNE COMPUTER MODULE	624W	109.1 LBS	ASCM CONCEPT
4B	FAIL OP/FAIL OP/FAIL SAFE - ADVANCED SPACEBORNE COMPUTER MODULE	1014W	174.9 LBS	ASCM CONCEPT

Communications And Tracking.

The communications requirements include telemetry and voice uplink from Earth, telemetry, voice and video downlink and GPS reception. Also included are requirements for EVA and Space Station communications and air traffic control (ATC) communications and navigation for recovery at a non-NASA facility. For minimum logistics and low cost, components such as pulse code modulated (PCM) encode/decode, audio and video drivers, S-band transponders, UHF/VHF communications and navigation hardware is chosen from in-production items for SSF, Shuttle, or other man-rated programs. The small number of PLS vehicles precludes a large investment in custom avionics. A shipset of equipment to perform all of the required tasks can readily be assembled from available components.

Data Processing.

The data processing subsystem provides, with a high degree of autonomy, the computational control and function to support all flight conditions. Embedded sequences support the requirement for automated test and checkout.

The computer complex for PLS is composed of a modified version of a fault tolerant concept developed by Draper Laboratories. The system uses a triple bus with crossover to

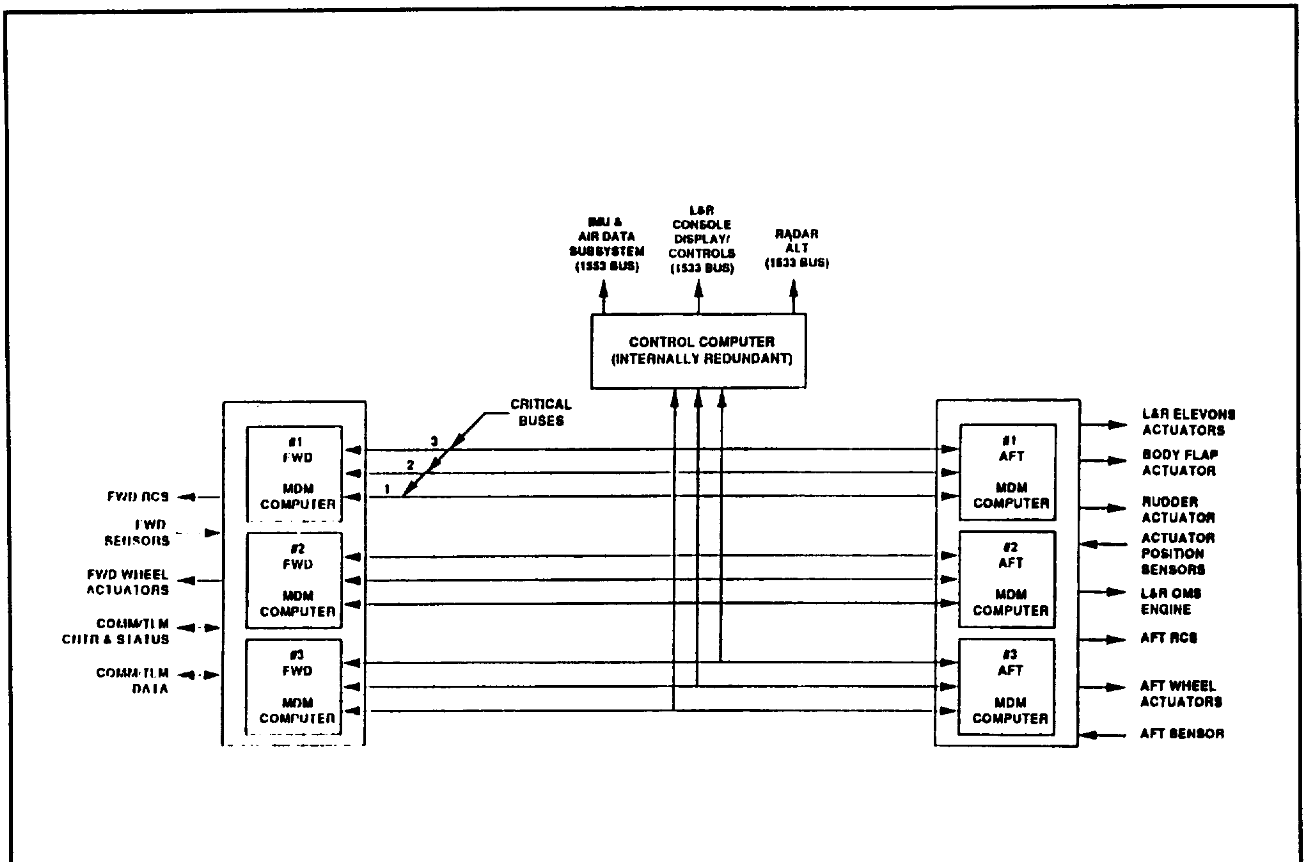


Figure 5-54. Block Diagram of Preferred Avionics Architecture

three processors which allows dual failures with the surviving components continuing the program. The Advanced Spaceborne Computer Module is used at each processing node. Each ASCM has dual self-checking submicron technology (VHSIC II) very high speed elements capable of operating at 3-4 megaflops (an order of magnitude faster than the Shuttle AP101 computers). This kind of computational speed in a small, simple vehicle like the PLS, allows for considerable overhead in activities such as all of the health and status monitoring, with work-arounds such as repartitioning around defective memory addresses, calling archival storage and restructuring data pathways. The use of VHSIC modules developed under the PAVE PILLAR and PAVE PACE programs minimizes development costs and capitalizes on a very large military development program.

Health Monitoring System Definition.

The objective of the PLS vehicle health monitoring system is to increase the probability of mission success for the vehicle, reduce the turnaround time and minimize the development and operational costs. Elements of the health monitoring system include: sensors, signal processors, algorithms, control logic, ground checkout equipment, expert systems application and maintenance system databases. To support the PLS program goals, a testability and integrated diagnostics effort must be implemented as part of the system engineering process from the beginning of the design process.

The fault detection/fault isolation process through BITE at the functional level provides functional operational status to the central health monitoring control processor.

Historically, the specification and design of BITE and monitoring systems has been confined to individual LRUs or isolated subsystems. Where safety or operational necessity are major considerations, subsystem/BITE monitoring is given high priority. Commercial aircraft systems (e.g., B-767/757) have dedicated 56% of memory for maintenance and redundancy management software compared to the 27% devoted to control of the aircraft. Such high-end system implementations must be contrasted with the inadequate BITE provisions in many existing LRUs. The typical bottom level approach is based on LRU BITE fault isolation specifications. Sometimes these criteria are not clearly identified with system level performance, system interoperability or system environment. For that reason, it has been notoriously difficult to verify actual performance against specifications. Typical current BIT/BITE implementations identify maintenance events that result in cannot duplicate (CND) rates between 45% to 54% and Retest OK (D) between 42% and 52%. (Emphasis in the system design of an integrated vehicle test system and supporting expert system implementation on the Unit aircraft, reduced these rates to 27% and 29% respectively).

Crew, technician, and management confidence in the health monitoring system is crucial. It is necessary to develop a system-level non-specific, integrated, BITE/monitoring approach starting with an analysis of the requirements, an evaluation of current implementations and the development of a descriptive model. The BITE system must be user friendly with CRT displays preferred. The BITE user interface and system must be more reliable than the vehicle operational system. Test procedure manuals on the vehicle and a test system linked to the test activities assure simplified operations and assure full configuration control.

The current trend in technology and requirements for system status is to provide health monitoring coupled to dynamic reconfigurability. The trend in fault detection is to establish a system architecture that can partition and structure lower elements and use embedded support within the lower elements. Fault isolation is being driven to "on-chip" test functionality due to advances in density and performance. The embedded and distributed test functions are being tied together with standard maintenance bus systems. Corrective action trends are putting increased emphasis on mission impact, with fault tolerant design allowing deferred maintenance and increased use of portable (on-board) maintenance aids.

The next generation of vehicle health monitoring systems will assure crew safety and operational efficiency at a minimal cost. The system will be used to prepare the vehicle for flight, monitor the condition of the vehicle in flight (safety and maintenance monitoring) support autonomous operation, perform a post-flight vehicle inspection and provide data for a reflight history database. (Safety monitoring is the integration of sensors and their accompanying algorithms into the controllers so that the anomalous system operation can be identified and action taken to either resolve the anomaly or safe the vehicle. Maintenance monitoring is the accumulation and analysis of operating data to be used in performing post-flight analysis and turnaround maintenance).

VHMS Support to Flight Operations.

During the operational flight phases, diagnostics are required to support five major options:

1. Abort Mission (Destruction Avoidance)
2. Alter Mission
3. Complete Mission In Degraded Mode
4. Reconfigure
5. Enter Fault Isolation Mode

In the past, realtime vehicle health monitoring systems have characteristically been BIT/BITE or "redline/GO-NOGO" systems providing an annunciation of a monitored fault indication with limited information or data provided for analysis or prognosis. In redline systems, at the time a parameter passed a certain level, action would be taken to prevent damage or possible destruction to the system or vehicle. With some types of failure modes, there may be changes in operating parameters or configuration that, if made, would resolve a particular anomaly and assure full or partial mission capability.

Current BIT/BITE systems are generating false alarm rates (85%), cannot duplicate faults (25%), and retest OKs (15%) which significantly limit confidence and usefulness. Expanded capability for in-flight system anomaly resolution can be obtained through the development of an integrated diagnostic system consisting of sensors, signal processors, algorithms and control logic.

VHMS Support to Ground Operations.

Deficiencies in current ground processing systems account for excessively high operations costs, excessive length of turnaround cycle, excessive failures generated during ground power-up operations, and inadequate reliability. BITE has had lower front-end emphasis than operational system elements due to large development and circuitry costs. Consequently, it seldom performs to specified requirements. Inadequate BITE results in false equipment pulls, cannot duplicates, and retest OKs. The immediate impact of this inadequacy is an increase in required spares and maintenance time.

The development of a ground processing system in concert with a vehicle health monitoring system would be beneficial. Collection and storage or telemetering of inflight data in a maintenance database increases the ability to perform vehicle diagnostics and prognostics. With this information, critical component inspections and tests could be performed in place, as needed, as opposed to performing removal inspections and tests on a scheduled basis. This decreases processing and turnaround time by eliminating unnecessary maintenance. Additionally, incidental damage to other components that would occur as part of a launch processing system would reduce the "standing army" required to process launch vehicles.

Health Monitoring System Implementation.

The commercial and military approach to vehicle health monitoring system implementation is to develop a BITE/monitoring architecture which is not vehicle specific but which can be tailored and applied to different flight vehicles from C-17 to LHX. The PLS vehicle will also exploit the advances gained from programs like Pave Pillar/Pave Pace and MASA and adopt the databus technologies, LRU BITE advancements, and developed software.

The PLS health monitoring processor is dual redundant on the board but not dual board redundant. A master-slave concept is employed with the VHMS controller polled periodically by the PLS master flight control computer complex. Status messages are interpreted by the main computer complex and remedial action ordered or a crew alarm is initiated. The defective component is determined by the resident expert system and an assessment is made whether the repair is feasible during flight. The results of the assessment are saved in optical disk memory for telemetry downlinking or for vehicle post-flight analysis.

Of major significance is the interface at each subsystem. The health monitoring system must be part of every subsystem, with the appropriate limit detectors, status registers, comparators and self-check routines being inherent in the subsystem design. The monitoring system then polls each subsystem at a programmed rate (key critical monitors having priority interrupt ability) and a status message is assemble for transmittal to the main computer during the master executive machine cycle. An approach similar to this is used successfully on the checkout complex for the Block II GPS production.

Figure 5-55 shows the JIAWG methodology for vehicle diagnostics and fault coverage. The concept uses three distinct management levels in the top-down definition; system,

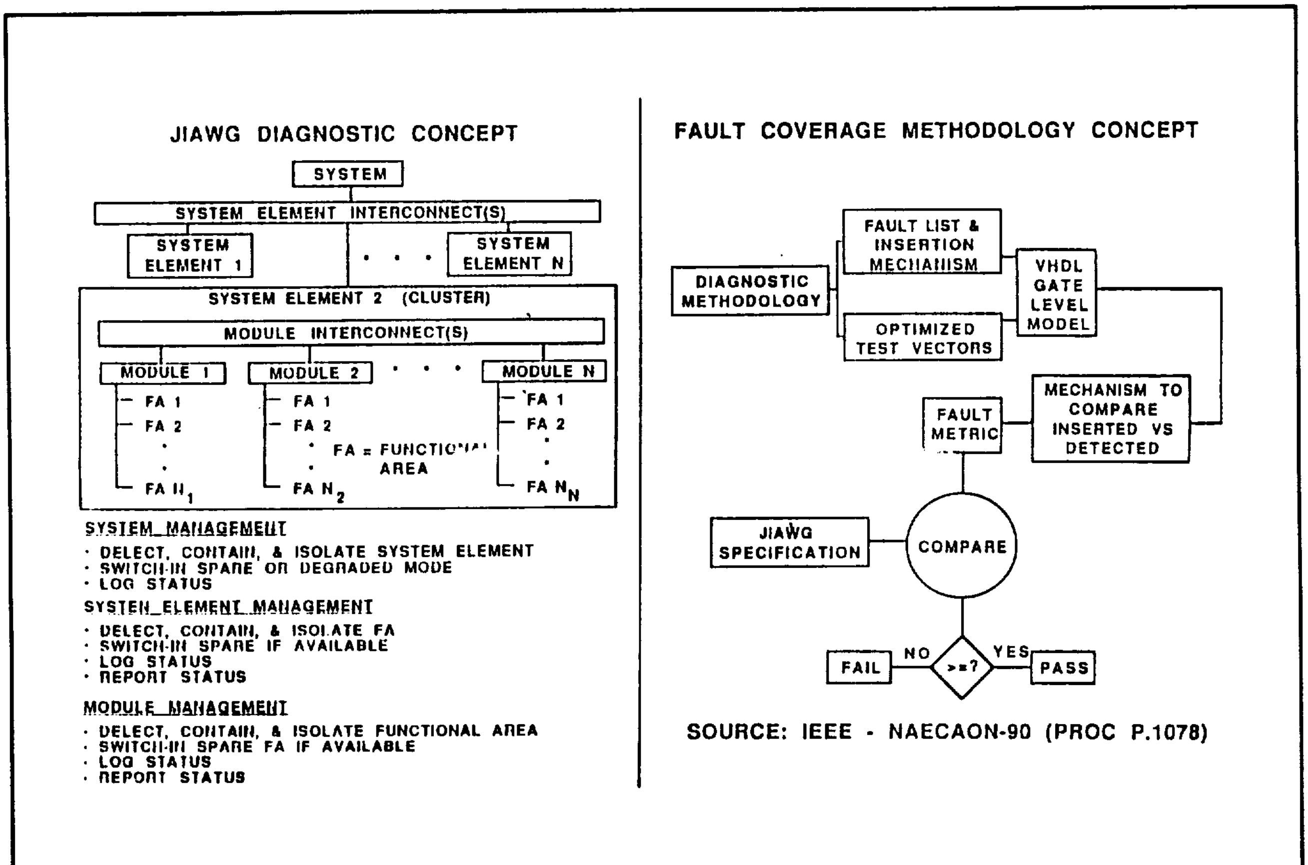


Figure 5-55. JIAWG Methodology for Vehicle Diagnostics

system element, and module management level. The fault coverage methodology consists of a combination of common procedures and tools, with fault metrics derived and verified by use of design-unique, gate-level models to be compared to the JIAWG-specified values for compliance verification. The use of tools such as fault lists, optimized test vectors, and insertion and comparison mechanisms is implied. Corresponding techniques are also used for the mechanical LRUs under the health monitoring system coverage.

For commercial aircraft, an ARINC committee had proposed a health monitoring capability which uses a central maintenance computer. The concept (Figure 5-56) features a display unit and control panel with optional data link via hardcopy printer and telemetry groundlink. Shared use of multifunction cockpit display/control units is anticipated unless a separate plug-in, hand-held computer is used. Printer and ground datalink capabilities are provided by ground plug-in modules if not on the vehicle. An air/ground datalink is integrated with a telemetry and control link to provide additional timely or critical information. The central maintenance computer has the same modules as the other vehicle computers to ensure spares commonality. The proposed connectivity between the central maintenance computer and vehicle subsystems is via both data and health buses.

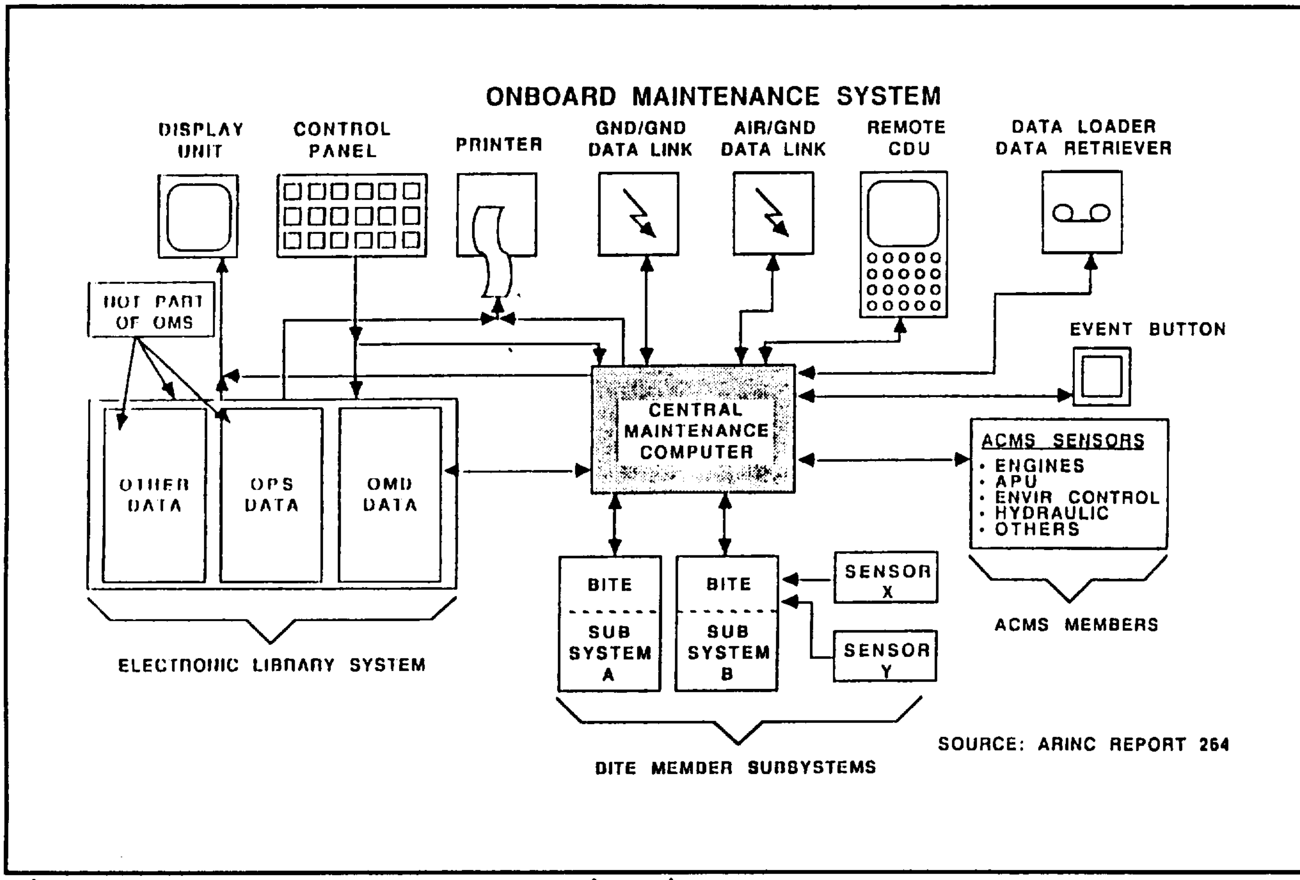


Figure 5-56. ARINC Health Monitoring System Concept

Figure 5-57 is a third approach to implementing a health monitoring capability. The objective of this system model is to identify the required functions of existing and future systems. The BITE/monitoring interfaces are defined in terms of system resources rather than actual hardware characteristics. The "Operational" area applies to the preflight, inflight and postflight operational phases. Its principal function is to reconfigure the vehicle and alert the crew to failures which are detrimental to vehicle or mission. The "Diagnostic" area supports the objective of two-level maintenance to minimize unique GSE. This approach presupposes that an adequate BITE infrastructure exists. Like the concept in Figure 5-56, the vehicle multifunctional controls and displays can be employed in a diagnostic mode to provide a high level of interactive technician support.

At this time, no selection has been made. Each approach is similar to the others and meets the same functionality. The final selection will be based on the state-of-the-art in health monitoring systems design in the commercial and military avionics industry at that time.

Displays And Controls. The PLS uses advanced flat panel or "glass cockpit" technology with a heads up display on the pilot's side of the cockpit. Conventional gyros and air data sensors on

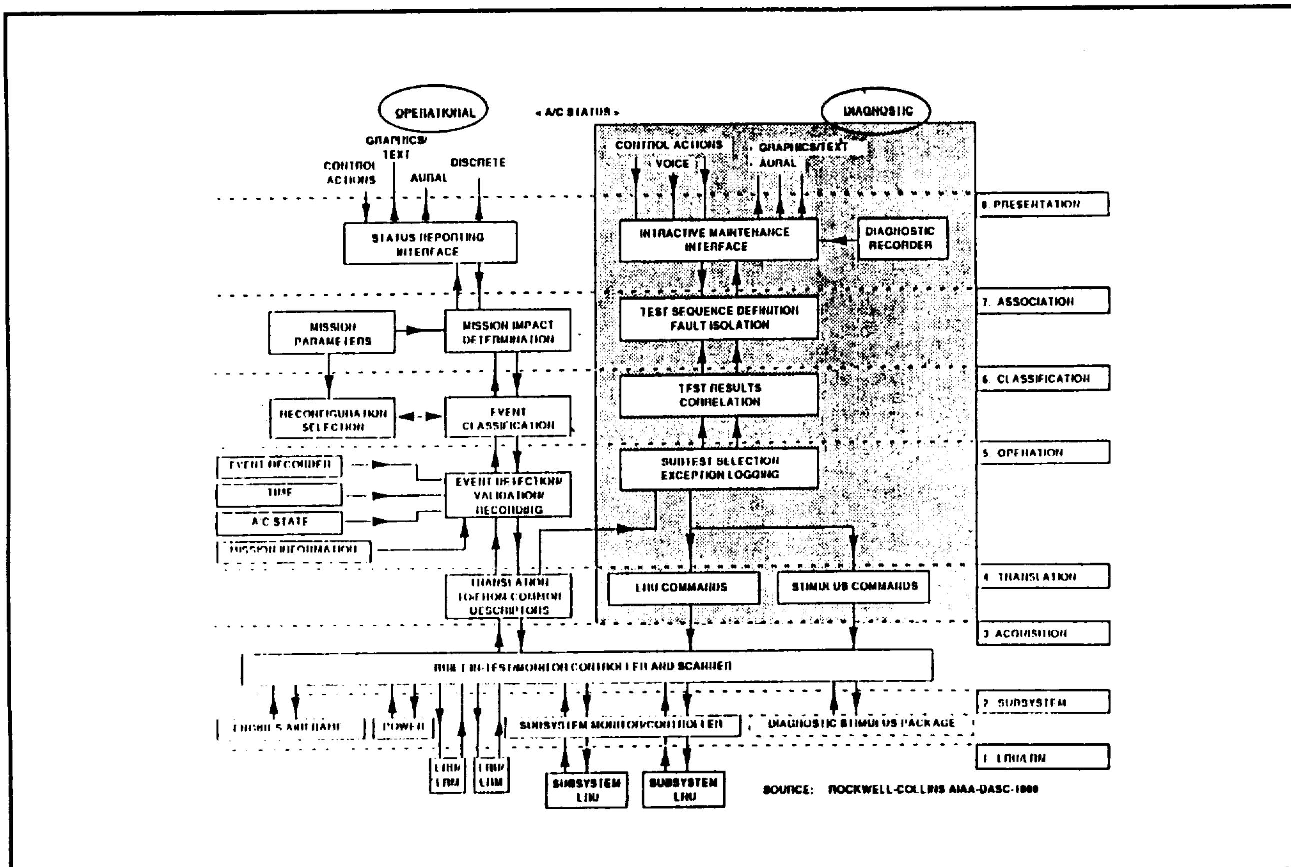


Figure 5-57. Alternate Concept for Health Monitoring Capability

the co-pilot's side cover major electrical discontinuities such as a lightning strike or a cosmic ray shower. The panel displays will be software configured to display the required instrumentation for the active phase of the flight.

5.1.11 Environmental Control System

The environmental control system primarily monitors and regulates cabin air quality (pressure, temperature, humidity, composition) and the equipment thermal environment to assure that specified limits are not exceeded. The system uses closed-loop servo processes and provides sensor initiated status data transfer to the master communication system computer for data distribution and to the onboard maintenance system computer to support maintenance analysis.

ECLS System Description. The selected environmental control system concept (Figures 5-58 and 5-59) avoids the use of freon, radiators, cold plates, lithium hydroxide canisters, and other high power and high maintenance components by implementing a "passive" approach thermal control. The heat generating equipment is installed in the vehicle to achieve a heatsink transfer to the aluminum cabin structure or extension frames. For the avionics installation, the water loop of the air conditioning system is routed near the cabin wall to which the avionics equipment is

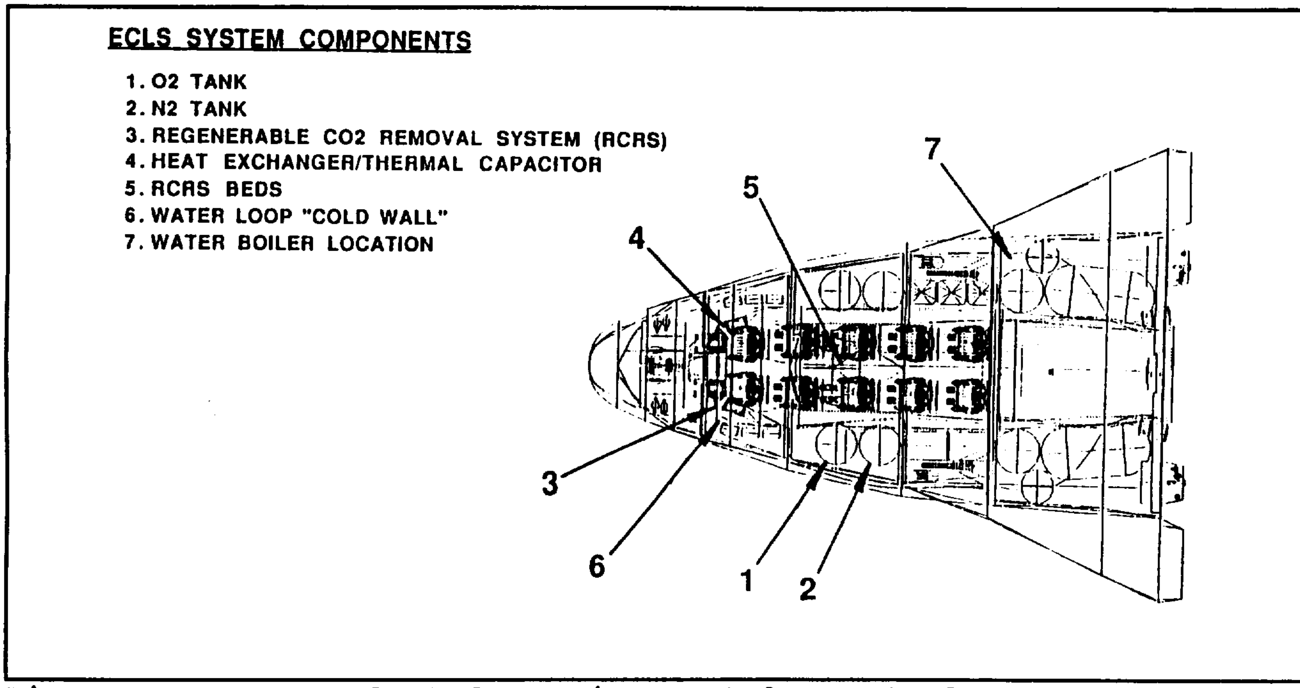


Figure 5-58. Selected Environmental Control System Concept Installation.

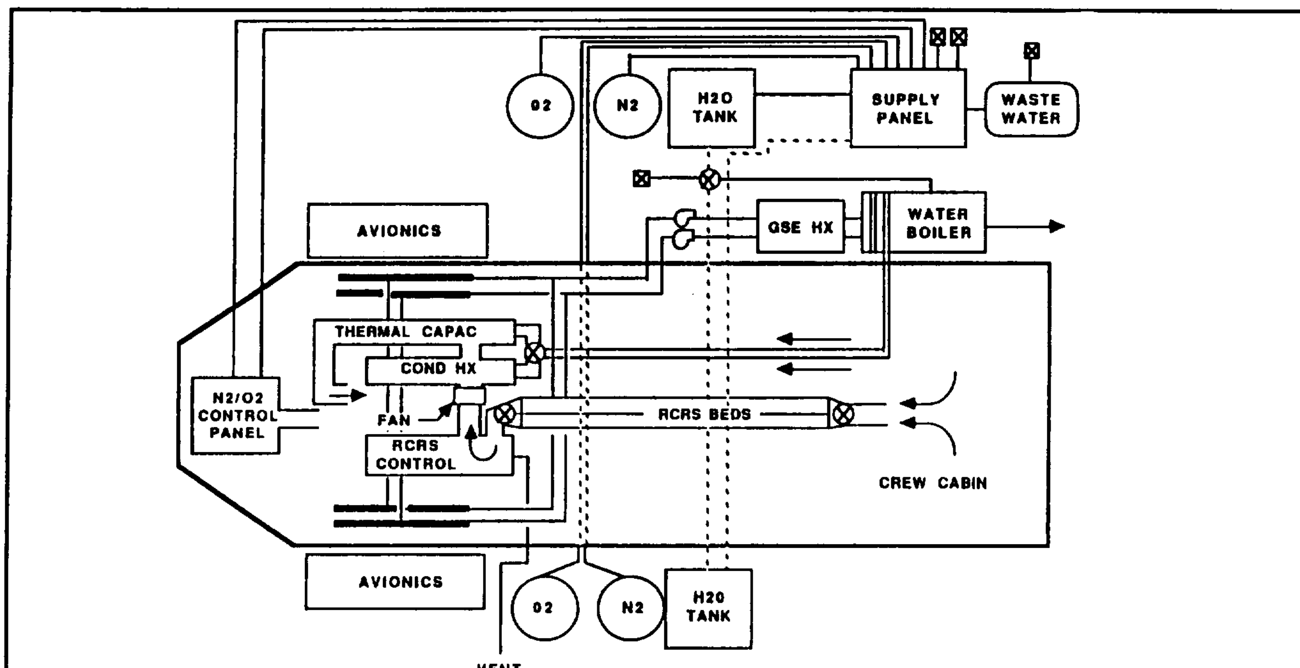


Figure 5-59. Selected Environmental Control System Concept Schematic.

mounted. This "cold wall" ensures the effective transport of heat generated by the avionics into the cabin cooling system and ultimately to the water boiler.

The RCRS system replaces the cumbersome lithium canister changeout for CO2 removal.

5.1.12 Personnel Accommodations

The nominal PLS mission profile has the personnel in the vehicle for nine to twelve hours during the ascent/rendezvous period and approximately nine hours for the return flight. These two relatively short periods permit a tolerable level of habitability for the small PLS vehicle. Trade studies were performed to optimize the cabin volume within the manufacturing and design objectives.

Alternate Crew Cabin Shapes. The "reference" crew cabin shape for the PLS vehicle uses a constant circular radius of 38 inches for the majority of the cabin length. There is a transition length between the seating areas and the narrow transfer tunnel for Space Station docking. The total volume available to the crew is about 500 cubic feet. For nominal mission segments lasting only a few hours, this is acceptable but more room would be better. The problem, as stated in the evaluation criteria, is "examine modifications to the reference crew cabin shape and internal layout for improved human factors acceptability." The approach for the trade study is to increase the habitability with approaches which a) minimize the change to the vehicle outer moldline but increase the cabin cross-sectional area, b) modify the seat location and aisle width and c) modify the cabin/tunnel transition area. For each option, an assessment of the manufacturing and operations impacts is made.

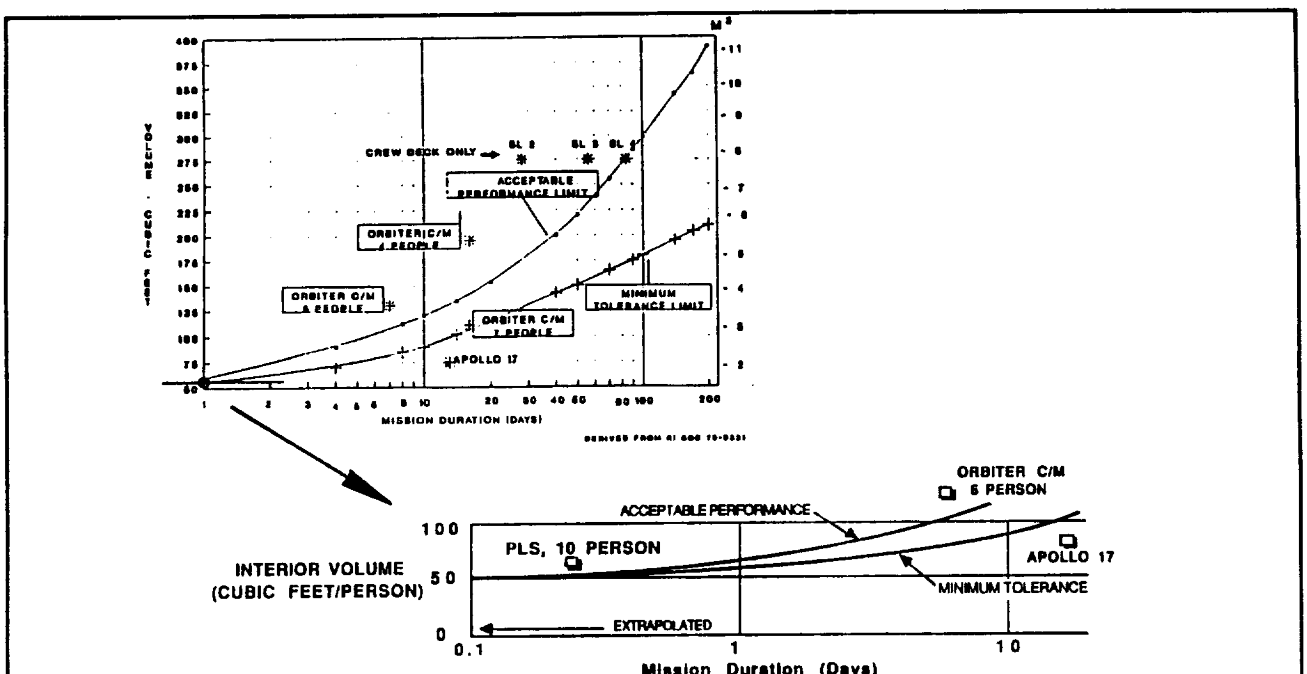


Figure 5-61. Habitability Effects on Space Vehicle Crews

Figure 5-61 presents data on the relative performance capabilities of space vehicle crews in terms of interior volume and mission duration. Table 5-10 lists the pertinent factors and their relative importance when evaluating a given cabin area for habitability. These criteria will be used in the evaluation of the crew cabin shape and layout options.

Table 5-10. Habitability Criteria

	SAFETY	PERFORMANCE	COMFORT	
VOLUME	♥♥♥	♥♥♥	♥♥♥	♥♥♥ ESSENTIAL
ORIENTATION	♥♥♥	♥♥♥	♥♥♥	♥♥ HIGHLY DESIREABLE
VISION	♥♥♥*	♥♥♥*	♥♥	♥ DESIREABLE
FLIGHT SUITS	♥♥♥	♥♥♥	♥♥♥	♥* COMMANDER/PILOT
OXYGEN	♥♥♥	♥♥♥	♥♥	
PRIVACY		♥♥	♥	
MEDICAL	♥♥	♥		
FOOD/WATER	♥♥	♥♥	♥♥	
COLORS		♥	♥	
WINDOWS	♥♥♥*	♥♥♥*	♥	

Ranked Parameters	Representative Components
1. Safety	Suit (Anti-G), Oxygen, Orientation, Injured Crew Egress, Medical Equipment.
2. Function/Performance	Spacial Location of Controls, Seat Adjustment.
3. Comfort	Volume, Visual Field, Color, Ventilation/Temperature.

The optional cabin shapes are defined by offsets from the reference moldline as determined from the baseline computer geometry. These offsets are used to calculate increases to the habitable volume. For the options which modify the location of the seats, cabin cross-sections are generated which show both seats and passengers plus proposed ingress/egress paths. Information is gathered from human factors databases which shows the limits of tolerable volumes by level of crew activity and duration of occupancy.

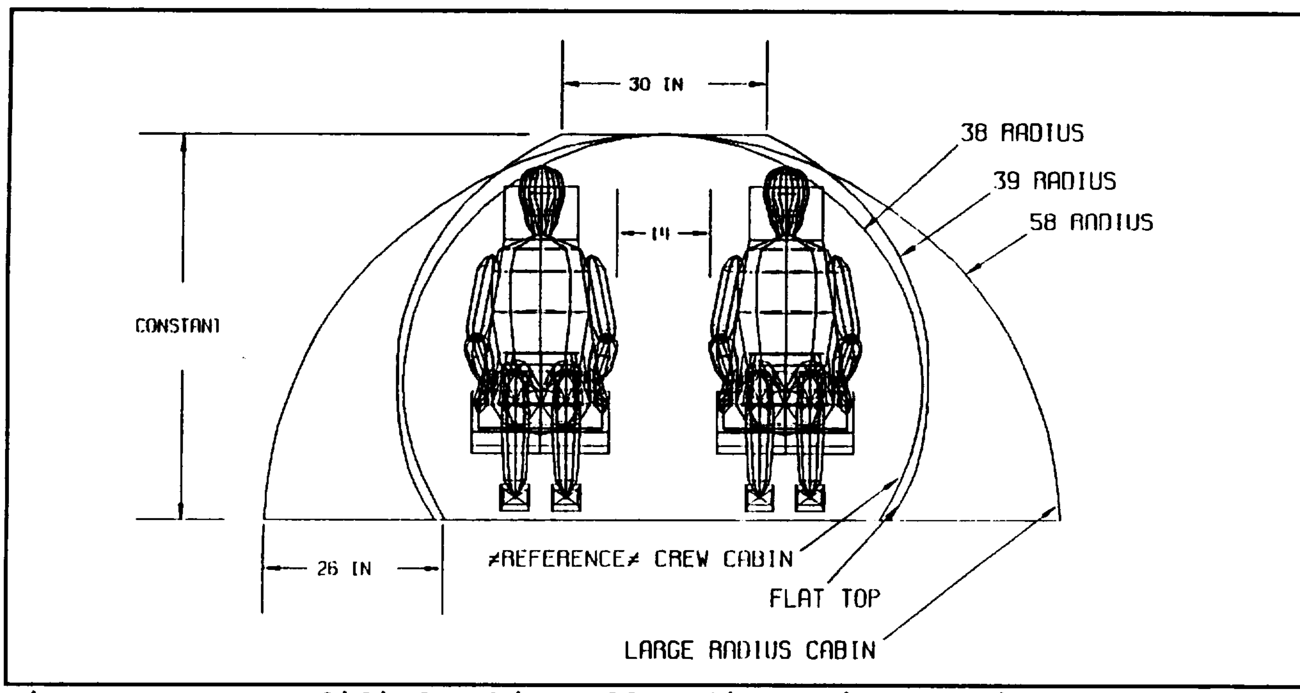


Figure 5-62. Modified Cabin Wall Radius (First Option)

<u>Modifications To Crew Cabin Radius.</u> The reference cabin inboard profile is maintained by constraining the upper centerline to its original location. Additional interior volume is provided

by increasing the cabin circular radius. A cross-section through the midpoint of the cabin is shown in Figure 5-62. The increase in interior volume is at the expense of space available for the subsystems which must reside between the cabin wall (pressure vessel) and the outer moldline. Aft of the cabin midpoint, the integration of the larger cabin radius (58 inches) is easy to accomplish since the vehicle is increasing in width. On the forward side, however, the large radius quickly exceeds the width of the vehicle and radically alters the basic moldline contours.

A second approach to modifying the cabin radius involves adding a flat section to the top of the cabin and joining it to larger radii (39 inches) on either side. This adds a small amount to the total cabin volume without significantly changing the cabin shape. Since the top of the cabin is the area where an ingress/egress ladder is installed and integrated with the cabin structure, the effect of the flat section would probably not add much weight to the vehicle. A more important impact would be to the manufacturing effort, since now the ring frames can not be produced in one piece but are divided by,and must be fastened to, the flat section of cabin skin. A modification of this approach uses the flat section only in the region of the last row of passenger seats in conjunction with the aisle width study. This option is shown in Figure 5-63.

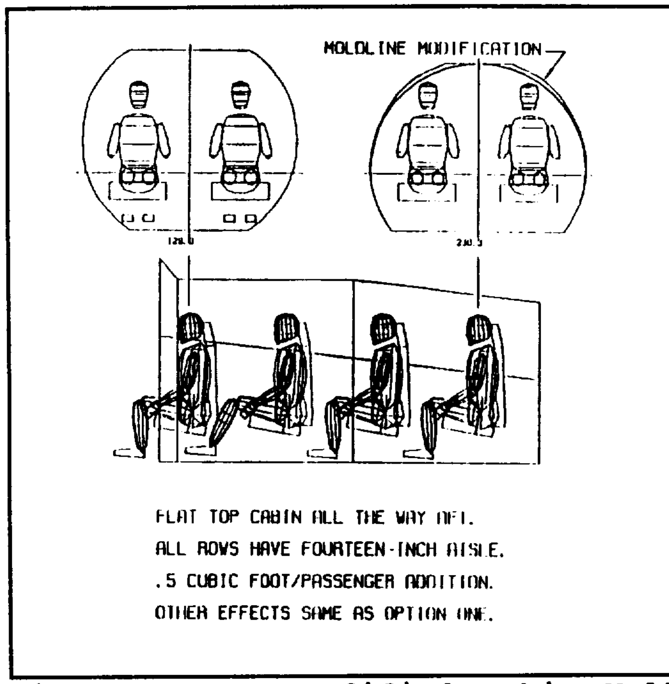


Figure 5-63. Modified Cabin Wall Radius (Option Three)

A third option adds headroom and internal volume by raising the centerline vehicle contour and providing a larger cabin radius. This approach conserves the reference subsystem volume and increases the cabin area but significantly alters the original PLS moldline shape. This option is shown in Figure 5-64.

<u>Seat Location And Aisle Provision.</u> The second part of the trade study addresses the location of the passenger seats. By moving the seats outboard, a center aisle is created to allow easier passage by personnel on orbit and especially for rapid egress during a pad emergency.

The reference structural concept for the cabin interior has the passengers seated close to the vehicle centerline to shorten the load path from the seats to the single center keel. This approach minimized the structural elements for transferring the launch and abort loads of the passengers and maximized the outboard stowage volume and headroom but made front-to-rear traverse difficult. The optional arrangement creates a fourteen-inch center aisle by using a dual keel structure in the center of the cabin. Headroom is reduced, especially in the aft row of passenger seats but the habitability and rapid egress capability of the design are much improved.

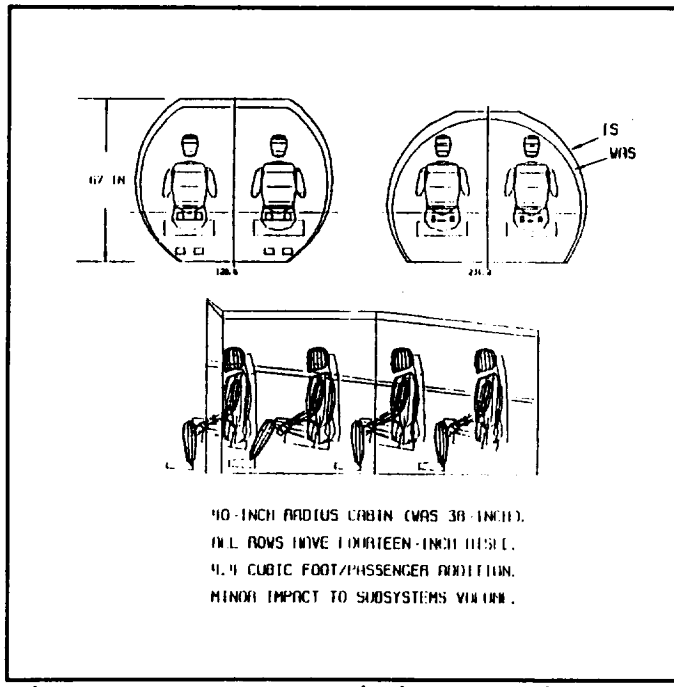


Figure 5-63. Modified Cabin Wall Radius (Option Two)

<u>Increased Transfer Tunnel Volume.</u> This option produces the largest increase in internal volume with no change to the reference vehicle moldline. The objective of the reference design was to maximize the subsystem volume in the aft section of the vehicle since the reference monopropellant concept required large tankage. By anticipating a change to a more efficient propulsion concept, more of the aft area could be allocated to usable cabin volume.

Rather than use a minimum forty-inch diameter transfer tunnel, the cabin volume was allowed to increase to the full moldline depth and a tapered width from 76 to 44 inches. This increased the available cabin volume by a full 16% to about 580 cubic feet. Additional benefits of this change are the much simplified cabin shape resulting from the elimination of the transition section of the reference cabin. The alternate structure is easier to manufacture because it is a simpler shape and has fewer elements. The launch and abort loads from the aft structural interface are distributed over a larger area and the new shape has much more direct load paths. The new shape also supports the alternate PLS missions by accommodating the installation of an internal airlock in the transfer area of the crew cabin. This design alternate is shown in Figure 5-65.

This study is unique in that any combination of the layout options defined above can be incorporated in the final design. Each provides some benefit to the habitability of the crew cabin volume. For some options the benefits are small but the impact to manufacturing complexity, structural integrity of the pressure vessel and weight growth are not inconsequential. The preferred concept adopts the designs which offer the larger habitability improvements with the smaller structural impacts.

The concept for the preferred vehicle retains the original moldline contours and achieves higher level of habitability by

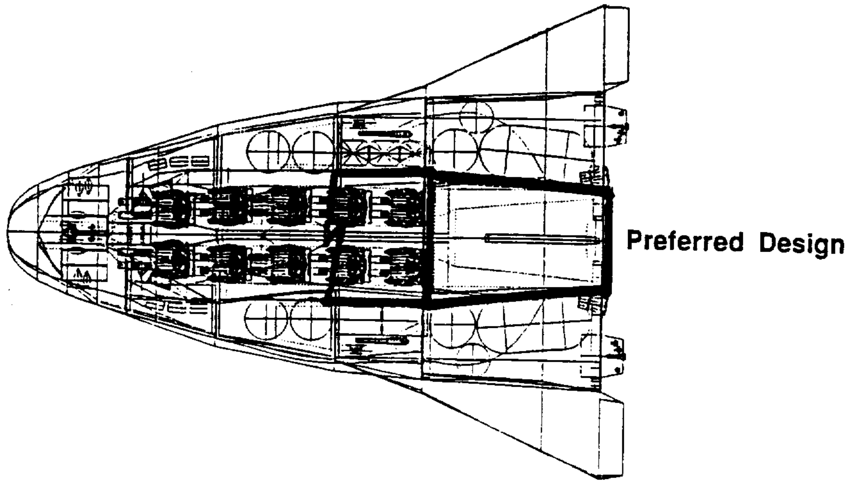


Figure 5-65. Modified Aft Tunnel Concept

incorporating a wide aisle in the cabin and the larger transfer tunnel configuration.

Cabin Interior Layout. Figure 5-63 shows a view of the cabin seating arrangement, proximity of the top hatch and the large rear transfer tunnel. The amount of space between passengers is illustrated in the figure as is the relationship between the passenger seats and the central keel and carrythrough frames. The aft tunnel area is provided as a privacy area for personal hygiene as well as the "cockpit" for the Space Station docking operation.

Passenger Seat Design. In addition to the general safety, comfort and reliability requirements and the capability to recline for the return of deconditioned Space Station personnel, the seats are to withstand the same loading conditions as Shuttle, namely;

1. A 280-pound ground step load over all seat surfaces.
2. A 125-pound on-orbit pushoff load.
3. A load of 600 pounds on the seat headrest as a prelaunch ingress/egress load.
4. A load of 840 pounds on the seatback for a prelaunch rescue.
5. Crash g-loads of +20/-3 longitudinal, +3/-3 lateral and +10/-4 vertical.

The factor of safety on the seat structure is 1.4.

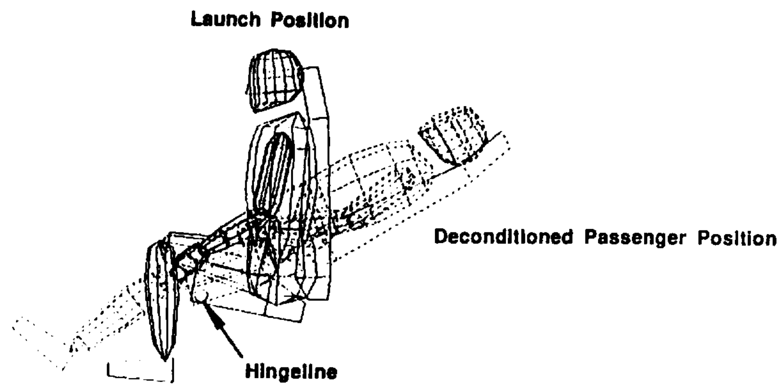


Figure 5-66. Seat Design Shown in Launch and Entry Positions

The PLS seat design is shown in Figure 5-66 in the launch and reclined reentry positions. For launch, the angle of the seat pan is 6-degrees down and the angle of the seatback is 2 degrees forward per specification. For the reclined position, the back angle must allow up to 66 degrees aft tilt. To achieve this amount of angular motion with a seat pitch of only 34 inches, the seat pan must be hinged at the front as well as where the seat pan and seatback meet. This concept allows the seat pan to rotate upwards so that when the seatback is tipped aft, there is clearance above the next passenger to the rear. The forward hingeline on the seat pan defines the location of the seat support structure, which coincides with the interior, carry-through portion of the major extension frames on the vehicle.

Since the seating concept must provide a five-point safety harness, the motion of the seat pan means that the harness must be attached to the seat rather than directly to the cabin structure. With the seat's proximity to the deep keel and frames in the cabin, this requirement presents no design concern.

In a crash situation, it is desirable for the seat to be able to stroke large amounts in the vertical and forward directions to attenuate the forces of the crash. The forward attenuation can be achieved by a plastic deformation of the lateral frame to which the seat pan is attached. The vertical stroke can accommodated by local failure-in-bearing of the keel and side web material where the aft edge of the seat pan is attached to these structural elements. This vertical stroke can be as much as 10 inches without contacting the feet of the rear passenger. This capability is not available in seat designs which rely on legs to transfer loads to floor structure and provides an extra measure of survivability, especially in the tail-first water landing scenario.

Rapid Egress Capability. For on-pad emergencies which require the evacuation of the PLS vehicle, there is an issue regarding the ability of the ten occupants to exit the vehicle within a specified time. The limited cabin volume, close seat spacing and small top hatch all contribute to this concern.

The total exit time is driven by the bottleneck in the egress operation. The assumption is made that the bottleneck is the top hatch which is a 32-inch wide by 40-inch tall oval. (This assumption is made because the seats described above and other interior aspects are more easily modified than the hatch size, and because, since unharnessing by the crew occurs simultaneously, a queue behind the hatch will probably form.) The opening area of the hatch is approximately six square feet and the hatch is hinged at the bottom so it swings downward when opened on the vertically-oriented vehicle to create a small platform.

A figure of two minutes had been discussed as a reasonable target for the emergency exit time. To assess the PLS design's ability to meet this time (as far as hatch adequacy is concerned) and examination of the similar activity on commercial aircraft is made. It is an FAA requirement to be able to evacuate an aircraft within 90 seconds. To derive an exit area parameter, it is assumed that an aircraft which holds 150 people has two aft and two forward emergency exits and two over-wing hatches for a total exit area of 110 square feet. This area provides 1.1 square feet per person, per minute in meeting the 90-second requirement. Applying this parameter to the top hatch opening area of the PLS gives and estimated emergency egress time of 1.8 minutes, or just under the target. This crude airline comparison is conservative when considering that the PLS evacuees are trained astronauts. If the top hatch is correctly sized, then the seat design and location must assure that an adequate exit path is available to the passengers. This path must also support the situation in which one or more of the seated passengers "freezes" in the emergency.

The primary method or traversing the (vertical) crew cabin, from top hatch to aft tunnel, is by using the ladder attached to the "ceiling" of the cabin. To enter the vehicle, a person steps over the threshold of the top hatch and turns 180 degrees to climb down (or up to the flight deck). Since the top of the cabin slopes six degrees with respect to the vehicle centerline, the ladder is not vertical but more naturally inclined. A normal ladder traverse requires enough room for the climber to swing his thigh through 100 degrees or so. This requires over two feet of clearance between the rungs of the ladder and the fixed seat structure. Figure 5-67 shows the climber adjacent to the second row of passenger seats at the level of the hatch opening. The "view looking up" shows ample clearance with seated passengers. This clearance is provided by the 14-inch aisle design.

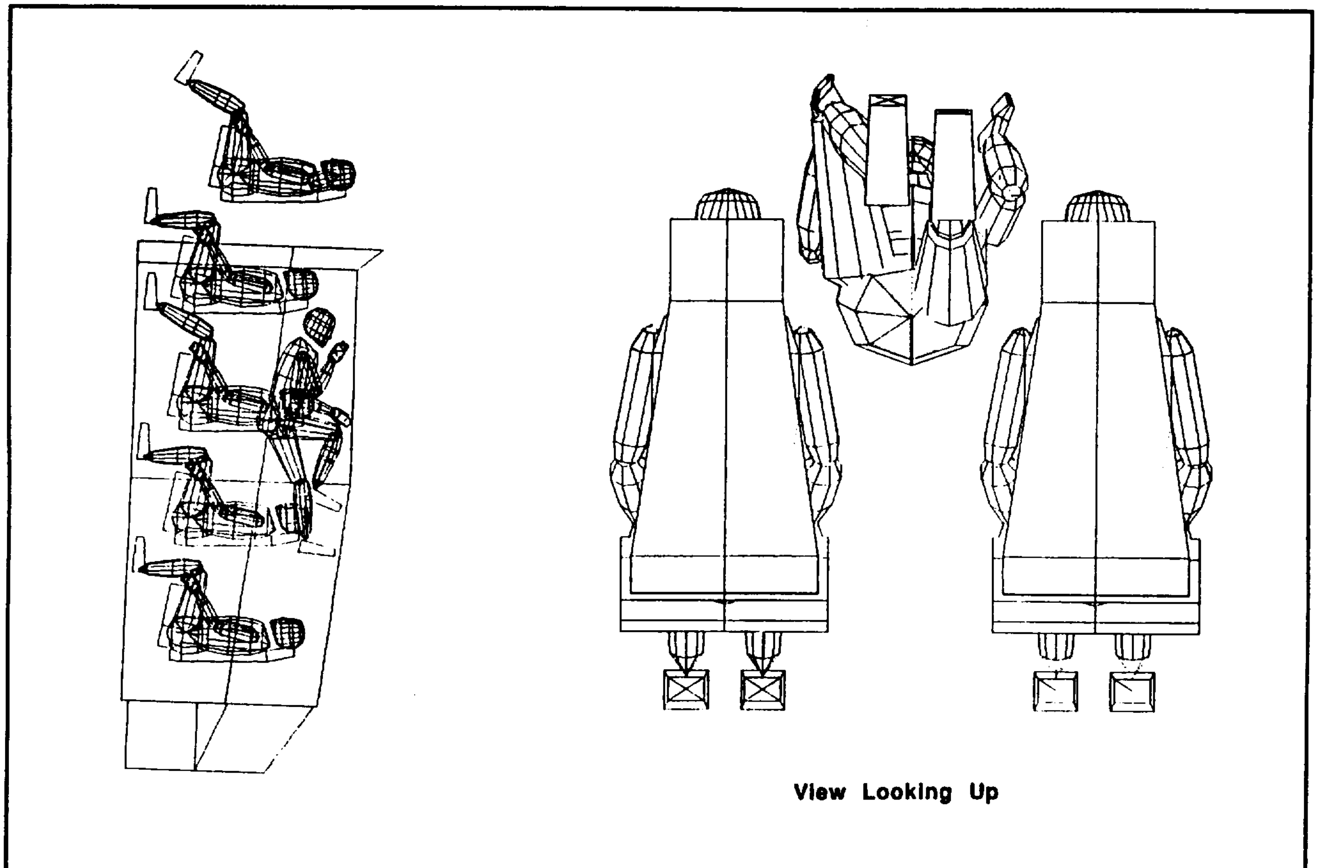


Figure 5-67. Wide Aisle Supports Egress Requirement.

What happens when the crew members arrive at the hatch in the emergency scenario? If the mobile service tower has been rolled back, the Shuttle experience suggests the use of a slidewire to get to a bunker. The Titan launch pad has a permanent service tower close to the vehicle. The PLS-end of the slidewire can be attached to a swing arm, located on this tower, and deployed when the evacuation signal is given. The design of the PLS-issue partial pressure suit can incorporate a harness to which is attached a short length of line and a slidewire carabiner. When the crew member reaches the top hatch, he pulls the carabiner and line from its pocket on his suit, clips onto the slidewire and jumps through the hatch opening. The swing arm is located so as not to preclude the hatch opening or to require a special deployment sequence.

The following is a possible timeline for the crew emergency egress scenario: at T=0, the command is given to evacuate the vehicle. Simultaneously, the swing arm mechanism is activated to position the slidewire near the top hatch. The pilot is first through the top hatch and ensures that the swing arm and slide wire are properly positioned. The commander remains on the PLS flight deck to serve as a gate keeper. He looks aft into the passenger area and directs traffic or gives instructions to facilitate the evacuation process. By T=5 seconds, all passengers have unbuckled their harnesses. Each rolls to his side and reaches overhead with his outboard hand to grasp the ladder. At about T=8 seconds one of the passengers in the first row is proceeding through the hatch opening after the pilot. The remaining passengers leave their seats and climb up the ladder in an orderly, rapid sequence. The advantages of having the seats close to each other near the vehicle centerline are that the reach to the ladder is shorter and that a slip would be easier to recover from. The 1.8 minute evacuation time estimated above allocates a very generous ten seconds per passenger for his activity at the hatch opening. In practice, it would probably be possible to evacuate the PLS in considerably less time than the two-minute allowance.

5.1.13 Recovery and Auxiliary Systems

Among the groundrules for the PLS vehicle specified during Task 1 of the AMLS/PLS study was the fundamental requirement to provide assured crew safety over the broadest range of mission and flight regimes as possible. This requirement lead to the inclusion of the LES in the reference vehicle configuration.

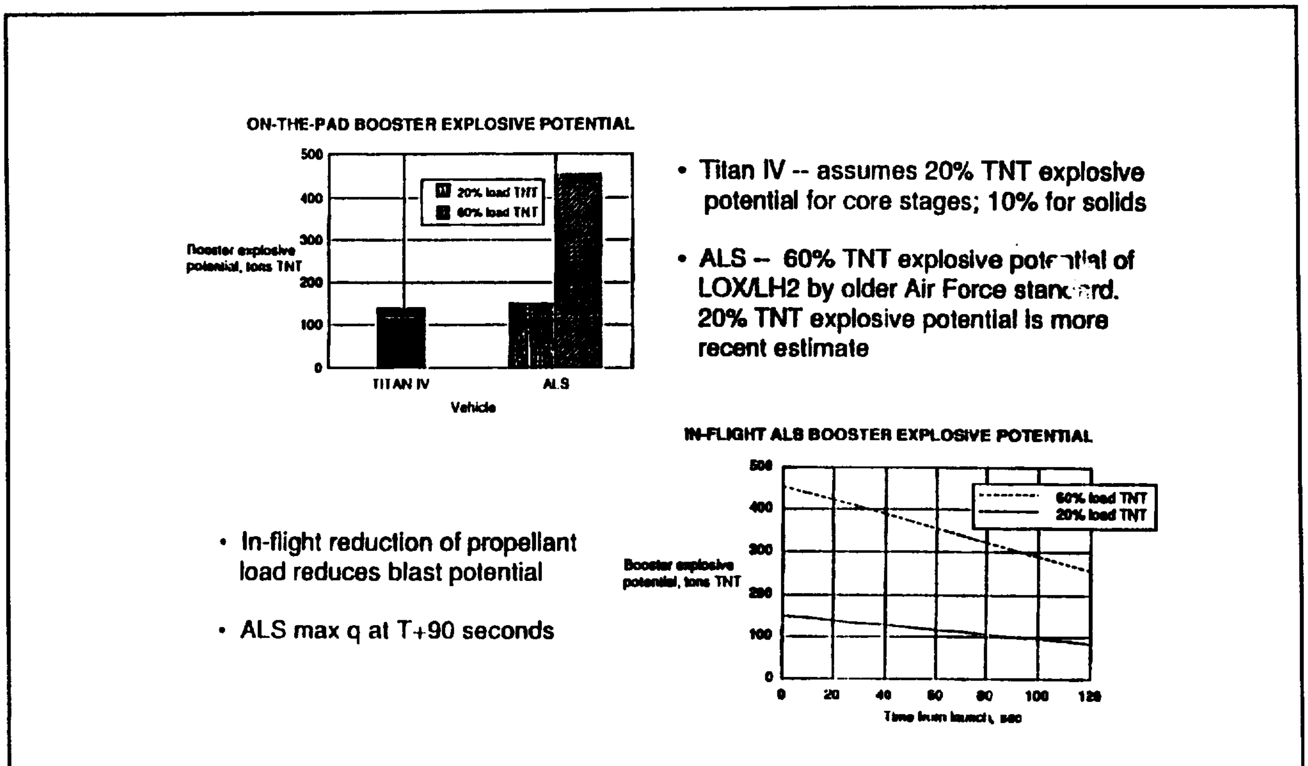


Figure 5-68. Explosive Potential for On-Pad Explosions

<u>Launch Escape System And Abort Definition.</u> An abort condition which receives much design attention is the case of an abort from the launch pad. In this scenario, the booster has all of its propellant and therefore has the maximum explosive potential. Figure 5-68 quantifies the explosive potential of both the Titan IV and ALS boosters for the launch pad and ascent situation.

Martin Marietta has provided information on the possible Titan failure modes which apply to the PLS abort issue. The conditions at the Titan pad are considered relatively safe and stable up to T-minus 31.7 seconds when the transfer of power is made from the ground to the onboard systems. At T-minus 20 seconds, the onboard destruction system is armed. At T-minus 1 second or at SRM/Core Stage 0 ignition is when the real potential for a PLS abort begins. The major possibilities for a catastrophic event lie in the propulsion and control systems. They are not truly a danger until engine ignition, followed by subsequent liftoff (at 150 milliseconds).

From ignition through ascent, potential conditions exist which could initiate an abort. As mentioned above, the major potential for catastrophic failure lies in the propellant and control systems.

The Titan has two types of propulsion systems, Solid Rocket Motors and a Liquid Fueled Core Stage. Failure in either of these systems can lead to a catastrophic event. The warning time is determined by the type of failure. An solid booster motor failure may present a warning in only milliseconds if the failure is structural. If the failure is a slow degradation, adequate time may be available. Within the last 20 years, a Titan 34D experienced an booster motor case failure at T-plus 8 seconds and the time for destruction was in milliseconds -- insufficient warning time for a PLS escape. A liquid propulsion system failure should be detectable through instrumentation and should allow adequate time for a PLS abort if the failure warrants. A liquid failure may cause the shutdown of an engine or engines which may not be catastrophic but may jeopardize the success of the mission. Within the last 20 years, a Titan 34D had a propulsion failure characterized by a low thrust indication. The vehicle remained stable but the propellant pump ceased, resulting in the shutdown of the engine.

According to Martin Marietta, a structural failure of the booster is thought to be the least likely event to occur. The Titan has a self-destruct system called ISDS which senses structural breakup and starts a sequence which terminates in vehicle destruction a 20 to 30 milliseconds after initiation. The PLS could not escape without more warning time.

The booster flight control systems are also a source of potential failure. The engine thrust vector actuators control the flight path of the vehicle as well as the loads on the launch vehicle structures. These actuators are commanded by guidance and driven by hydraulic systems. A hydraulic failure or mechanical failure in the actuator could result in a catastrophic event. A vehicle at high velocity and high aerodynamic loading which experiences a sudden change in trajectory or the initiation of a tumble would be destroyed. If one or two actuators failed, guidance might be able to compensate. Under these conditions, the PLS may be able to escape. Within the last 20 years, a Titan 3C experienced a hydraulic failure, then a loss of control and deviation from the planned flight path.

Other failure modes include avionics and instrumentation. These failures are not likely to cause a catastrophic event and should allow time for abort preparation, planning and execution.

When the crew is onboard the PLS and the PLS systems are in operation, the onboard computers will be monitoring the Titan health through the interface connector at skirt station 2492. The software will rapidly analyze conditions which are off-nominal, identify faults and predict the degradation of the Titan systems, mission completion probability, PLS survivability and the need for an abort. If an abort is required, the PLS (without human intervention) will sound the crew alarm and execute the crew abort sequence. If an abort is imminent, the PLS software will sound the alarm and provide time for crew preparation before executing the abort sequence -- manual override enabled. The role of the vehicle health monitoring system in the LES is to:

1. Detect off-nominal Titan IV conditions.

2. Determine true conditions.

3. Determine the threat of the conditions to the crew, PLS vehicle or mission and command the appropriate action for the current phase of the mission profile.

Figure 5-69 presents the results of the blast overpressure analysis in terms of distance from the point of booster explosion. These data are used to determine the amount of warning time required by the PLS/LES as shown in Figure 5-70. The following are specified as LES design and performance requirements:

1. Eight-g abort motor thrust capability.

2. Abort motor thrust vector control authority (preliminary requirement).

3. Stable powered flight and thrust termination.

<u>LES Design Concepts.</u> One LES concept places three, large solid rocket motors within the PLS booster adapter. This concept is shown in Figure 5-71. The motors are identical and are installed in a thrust structure within the adapter so that the nominal thrust vector from each motor is directed toward the combined vehicle center of mass. When the abort signal is received, the vehicle and LES separate from the booster and the lower half of the adapter along the line indicated in the figure. This additional separation plane achieves three design objectives. First, from a performance standpoint, the escaping vehicle is not encumbered by the full adapter weight. Second, the shape of the retained adapter segment can be tailored to meet certain aerodynamic objectives. The third advantage is an operational benefit since the complex, pyro-broken, separation plane can be a factory joint which leaves only a simple, structural interface at the rear of the adapter for the launch site field joint. Because the LES employs a pusher-type SRM configuration (in contrast to the

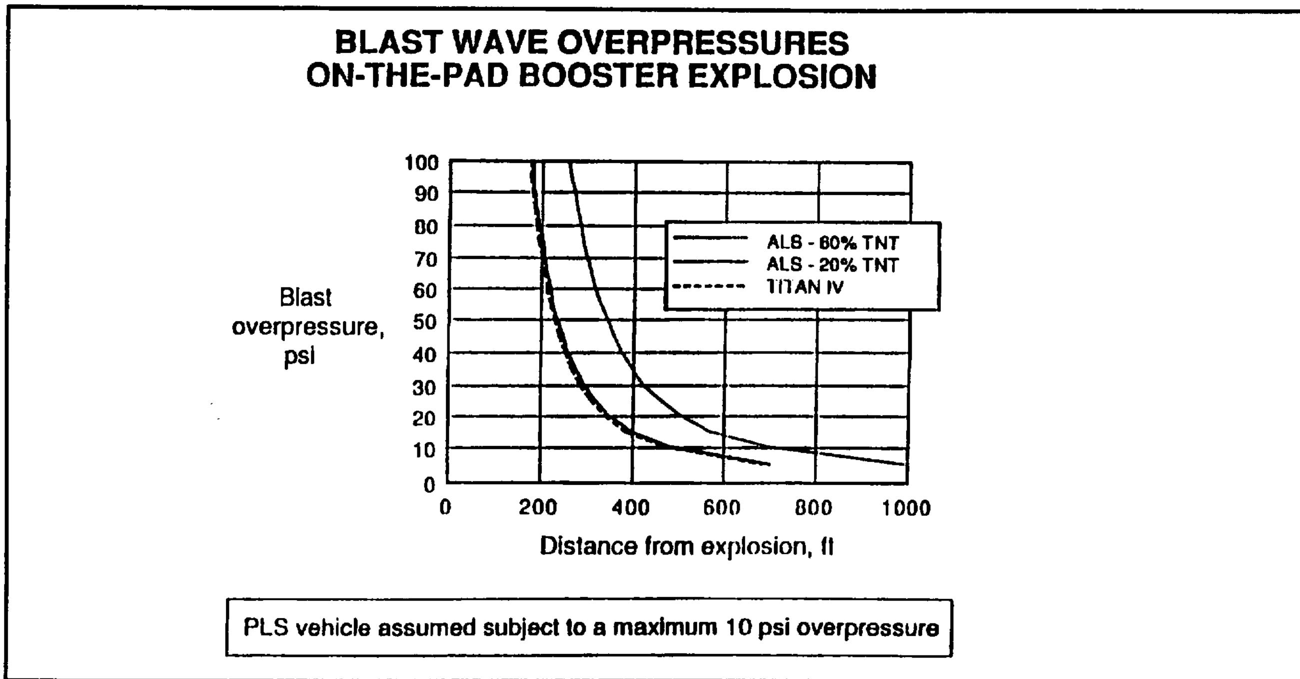


Figure 5-69. Blast Overpressure Analysis

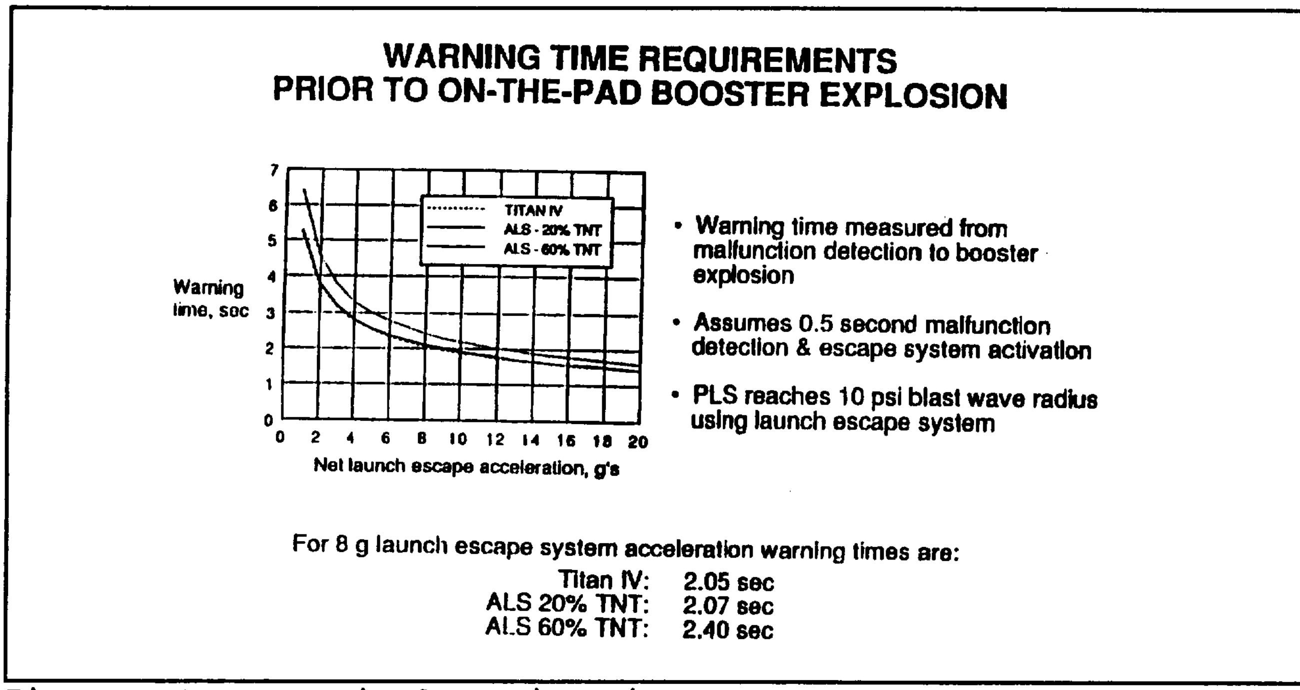


Figure 5-70. Required Warning Time

Apollo tractor system) the motors must have a steering capability. Table 5-11 presents SRM design and dimensional data for this concept.

An alternative concept relocates the same three solid rocket motors on the belly of the PLS. The objective of this change is to make available the area immediately aft of the rear docking hatch on the PLS so it is compatible with anticipated advanced DRM

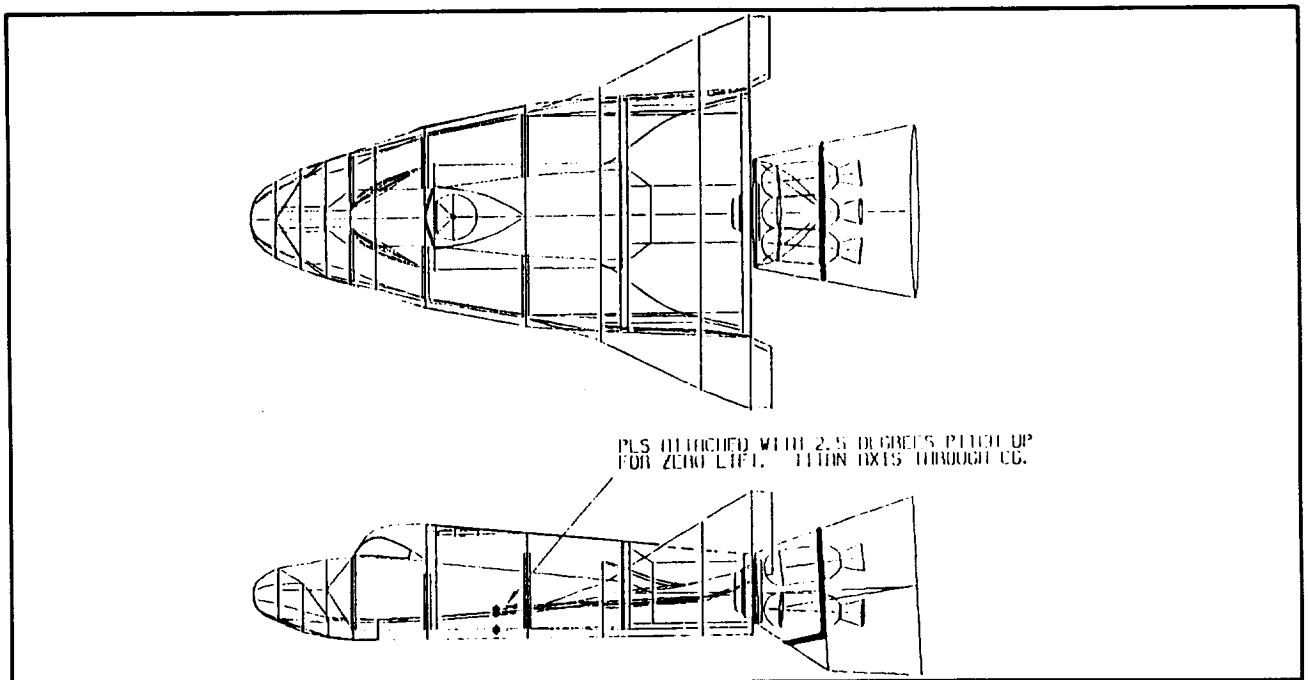


Figure 5-71. Launch Escape System - Initial Concept

requirements such as the attachment of a satellite service module. This concept is illustrated in Figure 5-72. The belly location is selected because the three solids are too large to mount easily to the outside of the adapter and because they appear to work better aerodynamically just ahead of the flat ramp of the adapter structure. As with the reference LES concept, the motor thrust vector is aligned with the combined c.g., but in this case the vectoring is accomplished with an angle bias on each of the motor nozzles. Design issues that this placement raises are the penetration by the LES structure through the thermal protection system and the probable addition of equipment (swing struts, springs, etc.) to assure that contact with the PLS vehicle is precluded after motor burnout and separation.

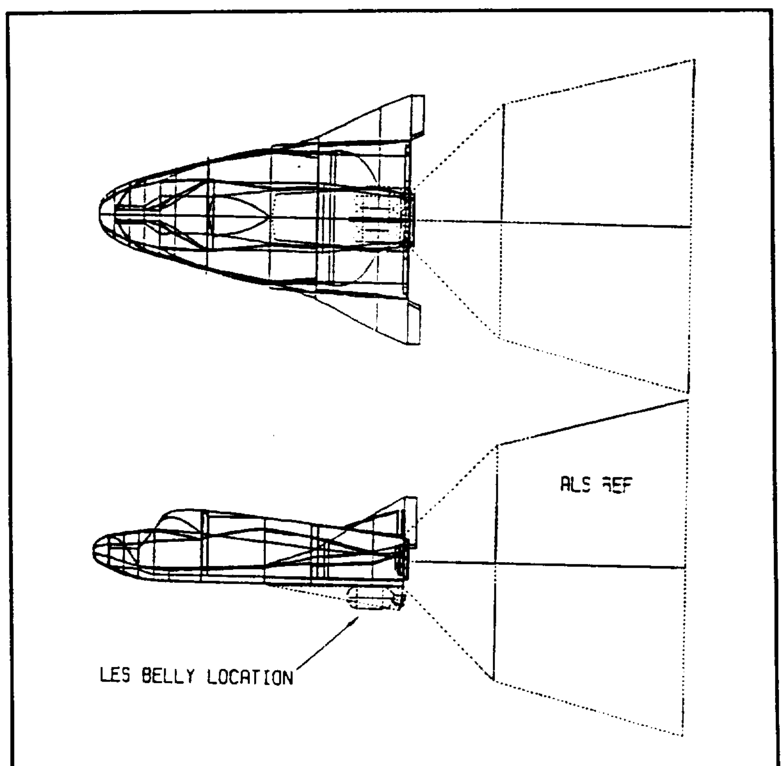


Figure 5-72. Alternative LES Concept

Table 5-11. SRM design and Dimensional data

Item	Value
Delta-V (ft/s) =	700
Motor Mass Fraction =	0.87
Motor Density (lb/ft3) =	108
Motor Isp (sec) =	260
Aero "K" factor =	0.8
Configuration =	PLS
g's =	8
Vehicle Weight *(lb) =	21,000
Total Sys Weight (lbs) =	31,000
No. of Motors =	3
Thrust per Motor =	82,700
Weight per Motor (lb) =	2,600
Volume per Motor (ft3)=	26.01
Burn Time (sec) =	8.17
Total Impulse (Lb-sec)=	2,028,000
L/D ratio =	3

* Includes Escape Structure Weight

Cylinder Dia (in)	length (ft)	Probable Selection
8	75.99	
9	60.04	
10	48.64	
11	40.19	
12	33.77	
13	28.78	
14	24.81	
15	21.62	
16	19.00	
17	16.33	
18	15.01	
19	13.47	
20	12.16	
21	11.03	
22	10.05	
23	9.19	
24	8.44	
25	7.78	√
26	7.19	√
27	6.67	√
28	6.20	√
29	5.78	√
30	5.40	√
31	5.06	

A third (and preferred LES design alternate) locates narrower and longer solids on the outside of the adapter structure as shown in Figure 5-73. The advantages of this configuration are that supplementary mission equipment can be attached to the PLS vehicle inside the adapter and that the smaller SRMs generally are sheltered from the aerodynamic/thermodynamic loading by the aft end of the PLS vehicle. Table 5-12 gives motor sizing data for this configuration. Figure 5-74 is used to illustrate the effect of LES motor Isp or altitude requirement on the abort system weight.

<u>LES Operation.</u> The separation of the PLS vehicle and LES from the booster occurs along a line near the midpoint of the booster adapter. The mechanical separation is accomplished with a pyrotechnic device such as "super-zip". The firing of the LES rocket motors is coordinated with the separation sequence to minimize the rocket blast effects on the booster propellant tank.

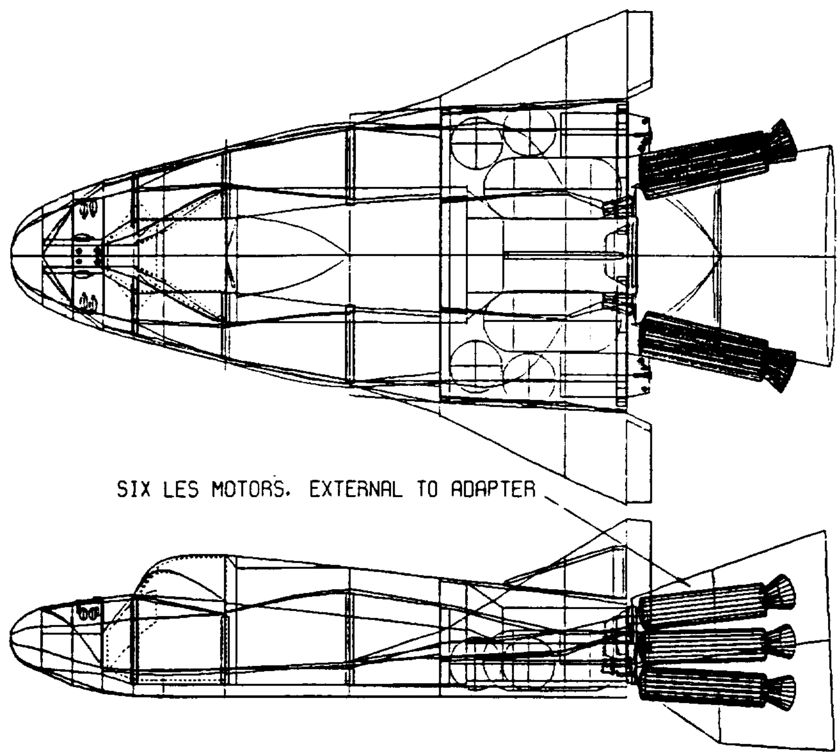


Figure 5-73. Preferred LES Design Concept

To maintain vehicle attitude control during the powered stage of the LES operation, the thrust vector control system on the rocket motors (preliminary requirement) is commanded by the PLS GN&C system. The location of the motors and shape of the adapter section retained by the LES are selected to support the control authority requirements of the PLS/LES combination.

The expended LES assembly is jettisoned from the PLS vehicle as soon as practical. The separation system is the same as used for the nominal booster separation and precludes post-separation contact between the adapter hardware and the PLS vehicle.

The recovery system for the PLS vehicle is a conventional three-main chute cluster arrangement composed of 120-foot diameter ringsail parachutes, deployed by a mortar-fired, conical ribbon pilot chute for each main. The parachute system is sized for a 30-foot per second sink rate. Figure 5-75 presents parametric data used to size the PLS parachute system. The effects of individual parachute failure on PLS landing velocity is shown in Figure 5-76.

The best arrangement for the parachute installation appears to be the co-location of all three chutes on one side of the vehicle even though this complicates the vehicle c.g. development. The

Table 5-12. SRM Motor Sizing Data for Preferred LES Concept

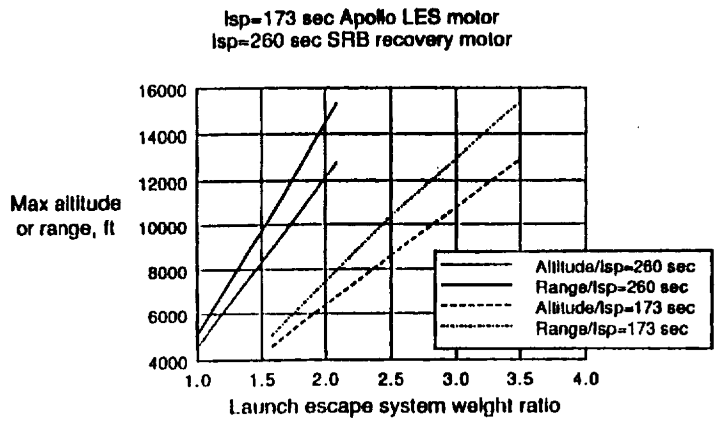
Item	Value
Delta-V (ft/s) =	700
Motor Mass Fraction =	0.87
Motor Density (lb/ft3) =	108
Motor Isp (sec) =	260
Aero "K" factor =	0.8
Configuration =	PLS
g's =	8
Vehicle Weight *(lb) =	21,000
Total Sys Weight (lbs) =	31,000
No. of Motors =	6
Thrust per Motor =	41,350
Weight per Motor (lb) =	1,300
Volume per Motor (ft3)=	13.00
Burn Time (sec) =	4.00
Total Impulse (Lb-sec)=	992,400
L/D ratio =	5

* Includes Escape Structure Weight

Cylinder Dia (in)	length (ft)	Probable Selection
4	151.99	
5	97.27	
6	67.55	
7	49.63	
8	38.00	
9	30.02	
10	24.32	
11	20.10	
12	16.89	
13	14.39	
14	12.41	
15	10.81	
16	9.50	
17	8.41	√
18	7.51	√
19	6.74	√
20	6.08	
21	5.51	
22	5.02	

location selected at this time (pending confirmation of final vehicle c.g. calculation) is above the main landing gear on the port side of the vehicle. The access panel above the chutes is pyro-separated from the vehicle as required by the chute deployment sequence. The risers which attach to the hard points on the crew cabin follow the front carrythrough frame to the centerline of the vehicle. The two forward riser lines are installed under the TPS blanket along the top of the cabin to the points where they attach to the two forward riser fittings near the top hatch. This configuration (see Figure 5-77) provides a four-point attachment of the chute cluster to the PLS to control the vehicle suspension angle and water entry angle.

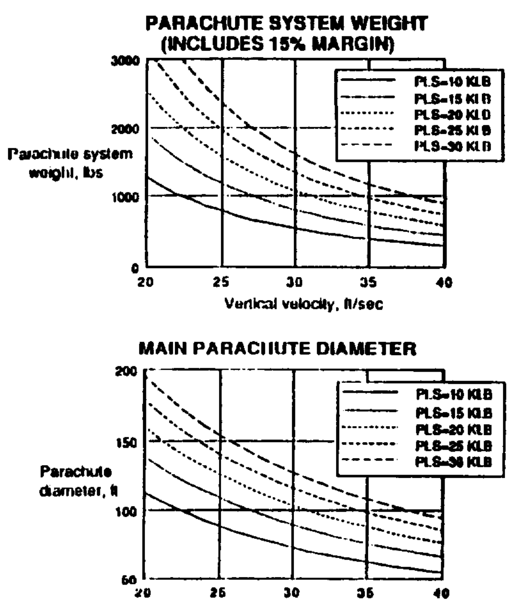
Upon impact with the water, the graphite/polyimide heatshield structure will probably fail by fracturing rather than a plastic deformation failure characteristic of a metallic structure. The heatshield design consists of bonded-in, hat section, longitudinal stiffeners and frames which coincide with the frame extensions from the crew cabin. These longitudinal and cross-ship structural elements will act as rip stops if a fracture begins as a result of a hard impact. (The fracture characteristics of Gr/Pi at 500



Baseline LES motor designed for 5000-ft range with Isp=260 sec

Conclusion: Reduced Isp or increased on-the-pad altitude/range requirement significantly impacts LES weight.

Figure 5-74. Effect of LES Motor Isp on Abort System Weight



- Based on Apollo parachute system
  - 2 drogue chutes
  - 3 pilot chutes
  - 3 main ringsail chutes
- Subsonic main chute deployment for launch and landing abort situations
- 15% margin on system weight
- Main parachutes sized for 30 ft/sec vertical touchdown velocity
- For 25 Klb PLS suspended weight
  - Parachute system wt = 1335 lb
  - Each main chute = 116 ft diameter

Figure 5-75. Parametric data for Parachute System

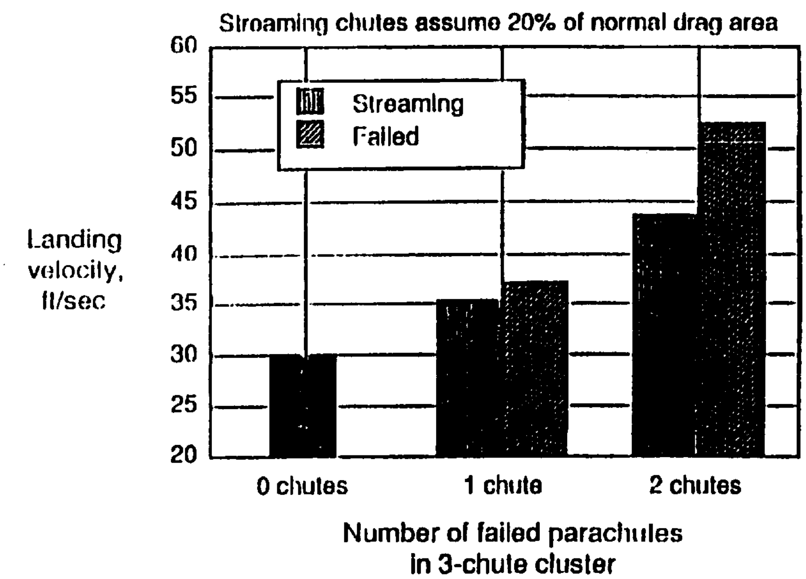


Figure 5-76. Parachute Failure Effects on PLS Landing Velocity

degrees F are similar to those at room temperature.) The Apollo capsule had several crushable "bumpers" between the heatshield and the pressure vessel on one side to absorb the forces of a hard landing. In the same way, the PLS heatshield will be sacrificial in nature to absorb or distribute high impact loads.

The heatshield attachment concept for minimizing heat shorts is compatible with the desire to isolate the forces of impact. The few links from the heatshield to the crew cabin will fail during impact, absorb energy and reduce the transmissibility of forces to the crew cabin. The design and orientation of these links will not pose a puncture hazard to the crew cabin in the hard landing case.

The PLS vehicle is suspended from the parachutes so that it enters the water tail first. For the primary structure, this attitude presents the smallest, most rugged structure -- the aft hatch. The aft secondary structure and the externally located subsystems benefit the cabin impact by absorbing energy.

Subsystems, such as minimum survival ECS, recovery communications and survival equipment must be located or protected to preclude damage in a worst case impact. Particular attention must be given to the design of the canisters which contain the flotation bags. This equipment must be functional after the water landing. An assessment of the other subsystem hardware must be

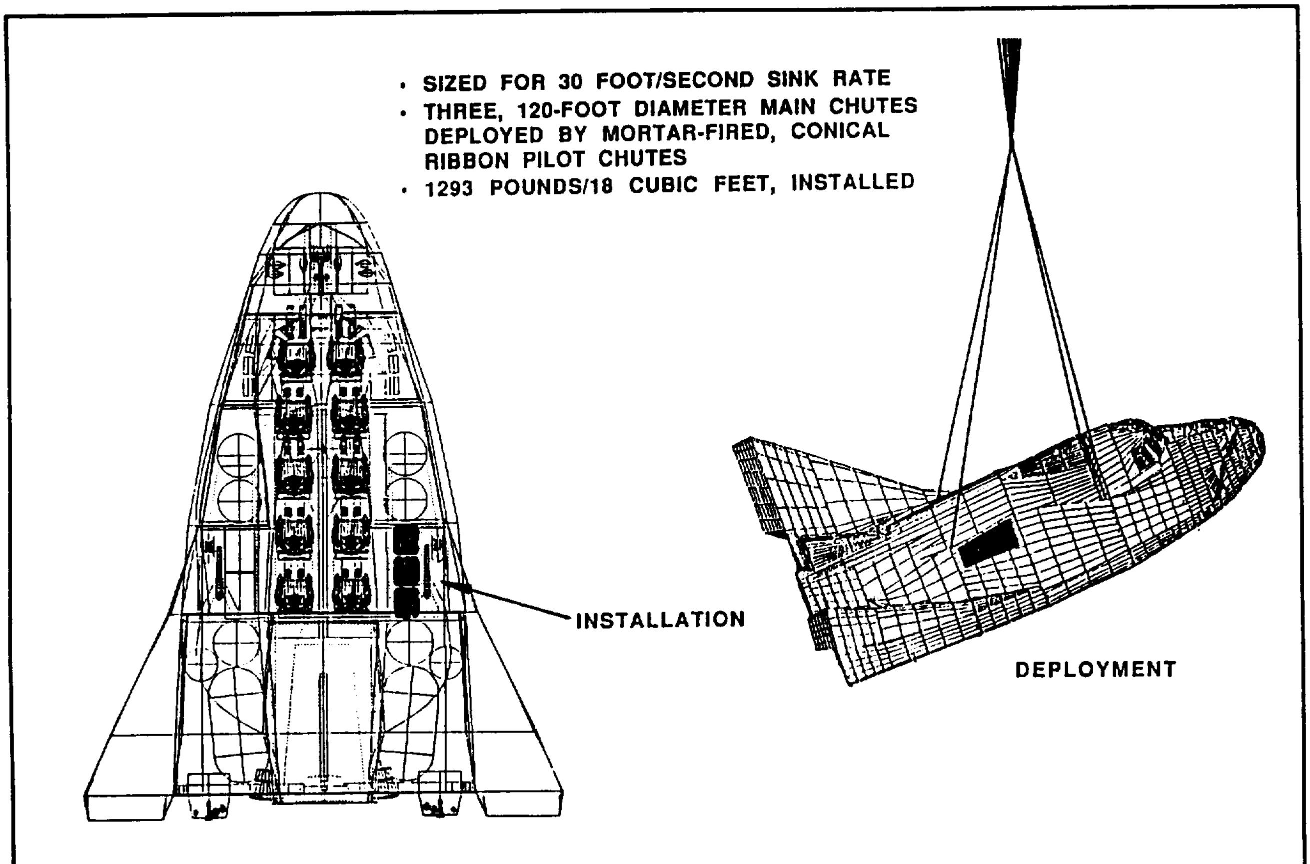


Figure 5-77. Four-Point Attachment of the Parachute Cluster

made to identify and mitigate possible hazards regarding secondary damage to the crew cabin.

A significant evaluation criteria on the PLS vehicle design is the ability of the vehicle to stay afloat in the range of expected sea states in a ditching scenario. Recovery at ocean locations far from land will involve waiting several hours for recovery vessels or aircraft to arrive at the splashdown site. The survival of the crew requires that the pressure vessel remain intact and seaworthy. The two areas of design which support this requirement are 1) the physical characteristics of the configuration which assure structural integrity of the vehicle during and after the water impact, and 2) the provision of supplementary flotation devices which maintain the vehicle condition until recovery is complete.

The physical attributes of the PLS were discussed above. The pertinent aspect of the structure is the integrity of the pressure vessel. For flotation, a study of subsystem component volumes was performed to calculate the amount of flotation provided by the pressure vessel and subsystems following splashdown. The results are illustrated in Figure 5-78. A supplementary flotation system is shown deployed from the aft end of the vehicle. This system assures that the aft hatch is always above the water line. If wave or wind action inverts the PLS, the lower pair of flotation bags again keeps the aft hatch above the water level.

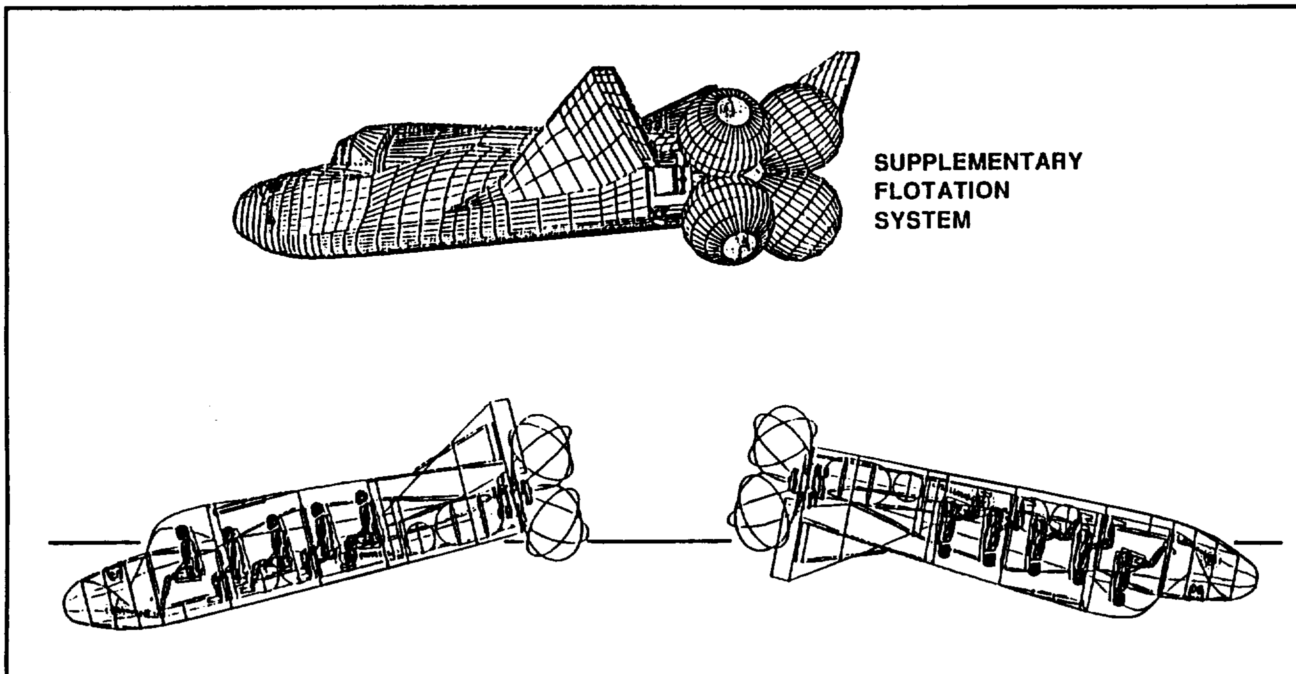


Figure 5-78. PLS Water Flotation Characteristics

5.2 SOFTWARE DEFINITION

The costs associated with the development of software for advanced space systems represent a large fraction of the total program cost. This trend is accelerated both, by the high cost of software and the increased complexity and functionality of the space systems as higher levels of autonomy and self-diagnosis are mandated.

A software requirements estimate was made for the PLS to support the total system life cycle costing task. The estimate was derived from a set of functionality requirements assumed for the PLS. The total software estimate is reported for vehicle requirements and for ground processing requirements.

5.2.1 Vehicle Software.

The vehicle flight software resides in two separate computer systems, one dedicated to a master communication system and another dedicated to an on-board maintenance system (OMS). The master system handles all communication and telemetry between the PLS and the ground, vehicle and the Space Station and between the vehicle and TSRSS. The OMS incorporates BITE, BITE access, failure monitoring and fault detection, a condition monitoring system , on-board maintenance documentation and provides total integration of these functions. Both of the computer systems provide redundancy and fault detection/fault isolation/self healing.

The onboard segment of the PLS software is defined as that portion of the software which operates totally within the onboard data processors of the PLS vehicle. Estimates were developed for the following components of the onboard software:

1. Guidance, which includes the requirements of ascent, on-orbit maneuvers, de-orbit and entry, aerodynamic descent, abort and other contingency flight.

2. Navigation, composed of inertial navigation and Kalman filter components.

3. Flight Control, consisting of inertial maneuver, attitude control and jet selection logic.

4. Sensor Processing, including GPS, Microwave Landing System, IMU, Star Tracker and Air Data Probe.

5. Sequencing, which controls effector arming, pyro arming, crossfeed selection, consumables dumping and separation sequences.

6. Utilities for math operations and coordinate transformations.

7. Executive Services, providing context maintenance, data conversion, interrupt handling and I/O management.

8. Communication, including telemetry, forward link, intercom voice and video.

9. Displays and Controls for HUD, dedicated displays, CRT drivers, caution and warning and hand controllers.

10. Database Management for user query, application update and retrieval.

11. Systems Management for power distribution, environmental control and consumables monitoring.

12. Redundancy Management, which enables system reconfiguration.

13. Fault Detection and Isolation, composed of BITE data consolidation, fault trees and system event monitoring.

14. Fault Correlation and Failure Reporting, composed of fault history analysis, a query manager, a report formatter and a flight data recorder.

15. Maintenance and Integration Operations, which supports ground servicing with a menu manager, on-line procedures data, testing algorithms and GSE interfaces.

Figure 5-79 summarizes the lines of code estimated for the PLS vehicle.

CSC NAME	SIZE (KSLOC)	EXISTING	MODIFIED	NEW
GUIDANCE	38.6	11.58	0	27.02
NAVIGATION	8.4	3.36	.84	4.2
FLIGHT CONTROL	13.0	1.3	0	11.7
SENSOR PROCESSING	8.6	0	0	8.6
SEQUENCING	8.0	0	0	8.0
UTILITIES	2.0	1.8	0	.2
EXECUTIVE	36.0	0	0	36.0
COMMUNICATION	17.2	0	0	17.2
DISPLAYS AND CONTROLS	35.0	10.5	0	24.5
DATABASE MANAGEMENT	5.0	5.0	0	0
SYSTEM MANAGEMENT	22.4	4.48	2.24	15.68
REDUNDANCY MANAGEMENT	8.0	.8	1.6	5.6
FAULT DETECTION & ISOLATION	30.0	6.0	1.5	22.5
FAULT CORRELATION & FAILURE REPORTING	30.0	9.0	3.0	18.0
MAINTENANCE OPERATIONS	30.0	0	0	30.0
INTEGRATION OPERATIONS	20.0	0	0	20.0

Figure 5-79. Estimated Lines of Software Code

5.2.2 Ground Software

In support of the vehicle flight software, the ground processing software also resides in two separate computer systems. A ground communication system handles all communication and telemetry including commands and a ground maintenance system compliments and supports the OMS to enhance the total maintenance process.

Software is required to support the PLS avionics development. Included in this software is that which is necessary to support the automated design effort of the flight profiles and envelopes for the various phases of the PLS mission. This software capability uses existing software for the Shuttle program to the extent possible including: rendezvous and docking, separation and descent, entry and landing, consumables, propulsion, power and ECLSS. The products developed by this software are a trajectory tape, a simulator data pack, I-loads, data tapes to specific users and various reports (e.g. range safety, US Spacecom, crew charts and flight data file). An automated mission timeline capability is developed by augmenting current mission timeline techniques from the Shuttle program to meet the PLS flight rate requirements. Specific capabilities are to: receive and load flight profile tapes, perform ascent and entry timeline development, integrate mission requirements, identify and analyze conflicts, identify procedure conflicts, perform timeline verification, generate crew activity products and tailor an operating system.

Additional ground software requirements support the PLS mission control center to provide realtime mission support and integrated training for data reduction, software enhancement of flight command and control and displays to monitor status during the PLS flight.

PLS systems simulation software provides the capability for end-to-end mission simulation for instructors, flight crew, passengers and flight controllers in a stand-alone or integrate environment with the PLS mission control center a computer-based, one-G trainer. The simulation software provides PLS familiarization, computer aided instruction, mission training and emergency training.

6.0 ACQUISITION PHASE DEFINITION

This section documents the acquisition phase definition - presents planning data for program phases A, B, and C/D. These data have been developed based upon accomplishing the specific major activities related to design, development, production, test, verification, safety, reliability, quality assurance, and management and control for both hardware and software. The PLS program master and manufacturing production schedules, including sub-tier schedules and the manufacturing flow and build plan are part of these data and are presented in this report.

6.1 PROGRAM DEVELOPMENT

One of the contributing efforts initiated early on in the study was to establish a top level set of program schedules that would provide a set of key milestones for all tasks, thus allowing the further development of subschedules unique to the specific requirements of each task area.

6.1.1 Master Development Schedule

The PLS Preliminary Master Program Schedule, Figure 6-1 establishes realistic and complete schedules for all of the following activities:

- Engineering
- Tooling
- Mission Operations
- Flight Test
- Operations Support
- Facilities
- Procurement
- Ground Test
- Production

The life cycle cost analyses and products reflect the milestones in this Master Program Schedule. The functional tasks areas; Subsystem Design, Manufacturing and Verification, and Operations and Support will each be constrained by the milestone established by this Master Program schedule.

The PLS Master Program Schedule reflects a number of ground rules:

1. All production work is based on a two shift, five day weeks schedule.

2. All production fabrication is support with one welding set of tooling and line. The one welding line supports two final assembly lines.

3. The schedules assume that the CAD, CAE, and CAM are in place and the associated schedule benefits can be realized.

The above ground rules were selected to produce a low cost fabrication and production operations. The over all schedule was developed to reduce risk and paralleling of major activities, such as development and final vehicle fabrication or flight testing.

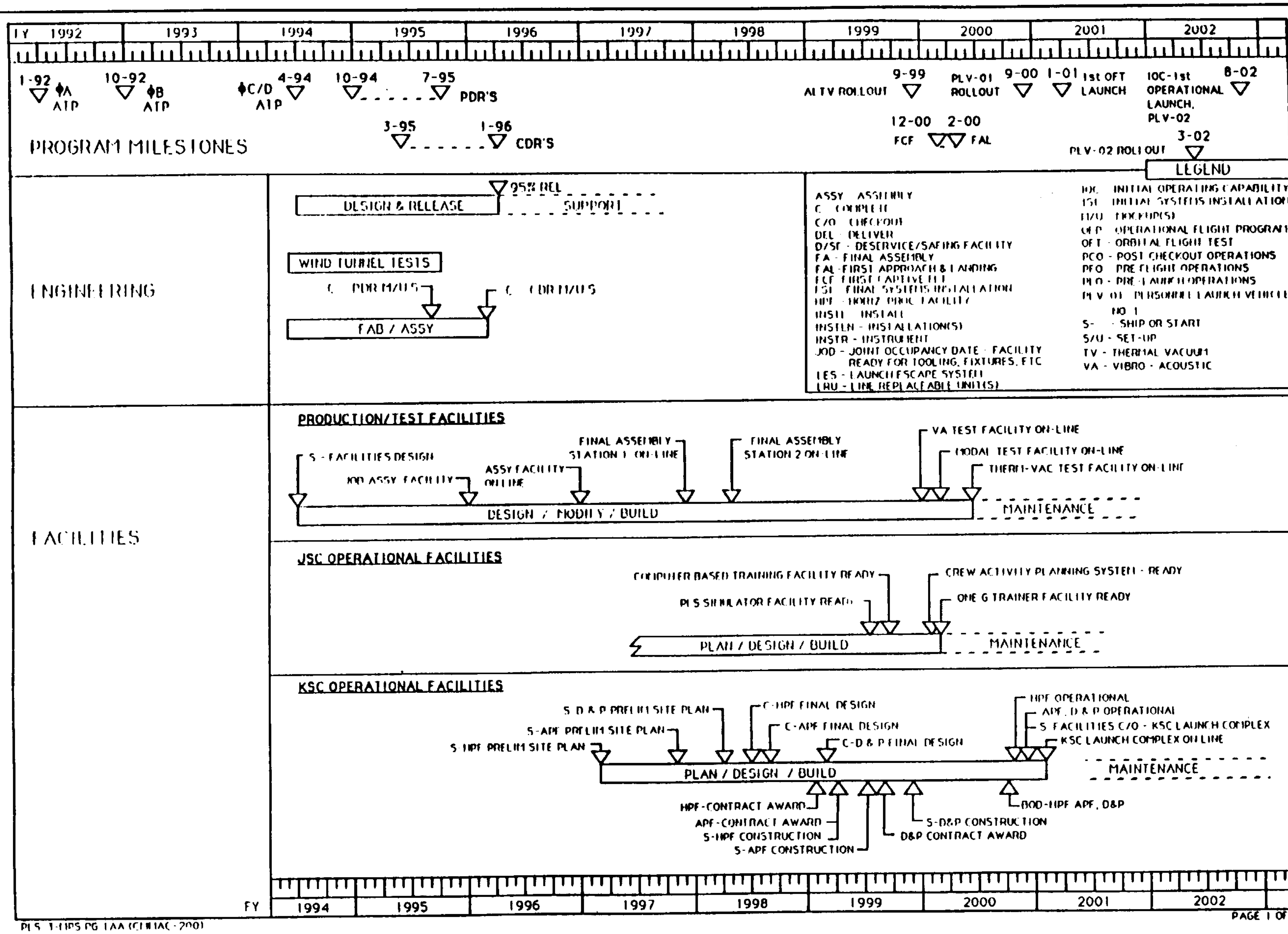


Figure 6-1. Preliminary Master Program Schedule

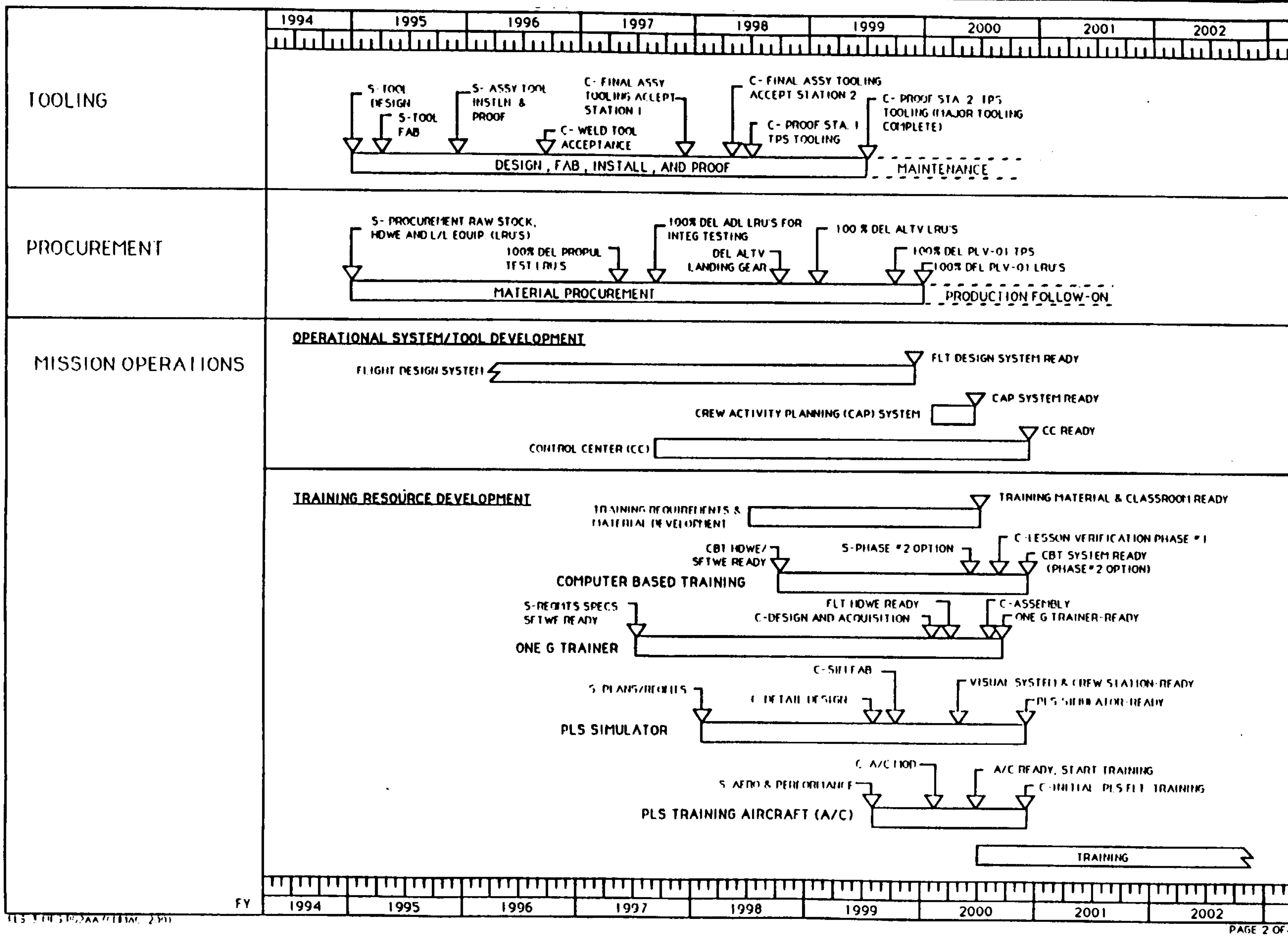


Figure 6-1. Preliminary Master Program Schedule (Continued)

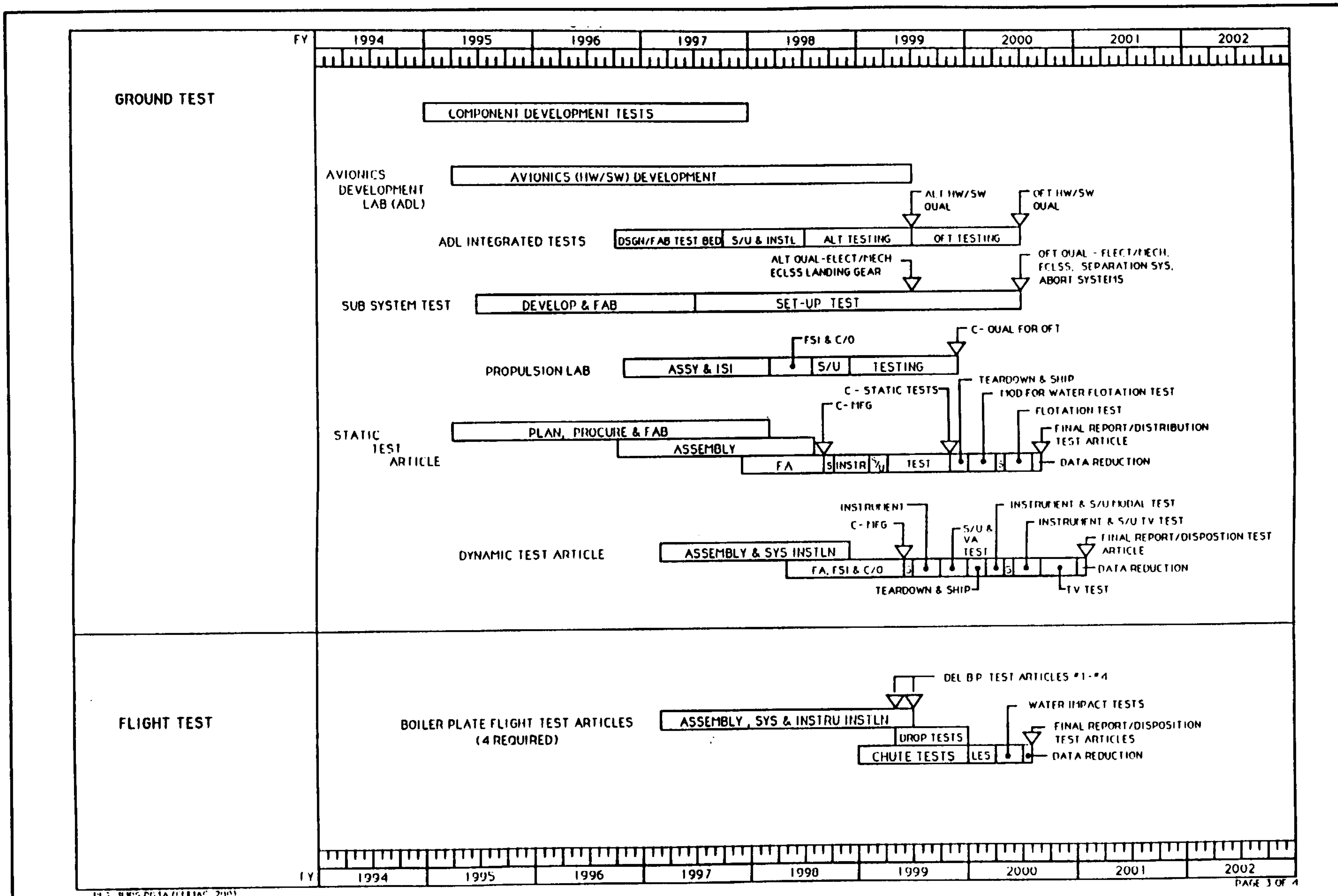


Figure 6-1. Preliminary Master Program Schedule (Continued)

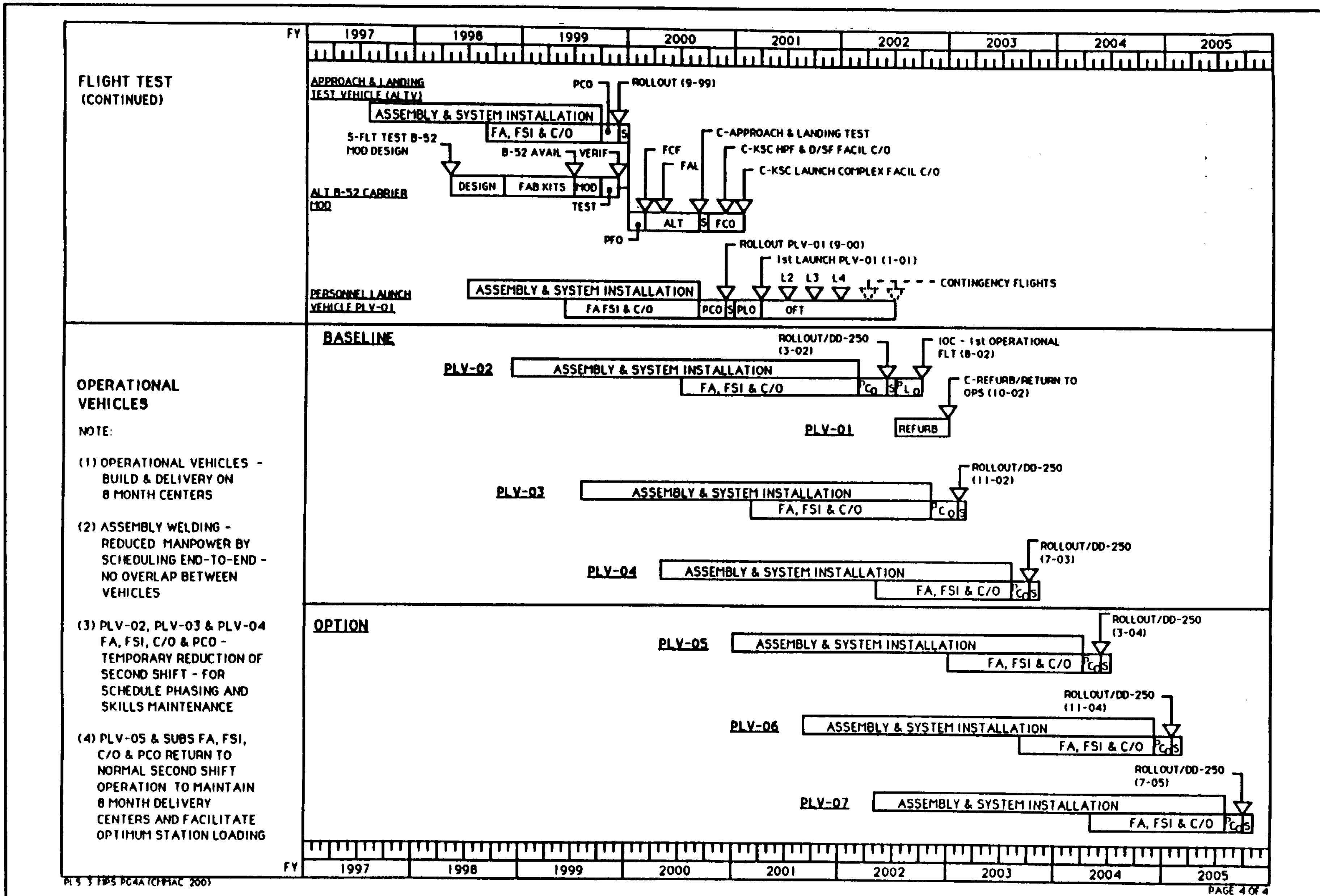


Figure 6-1. Preliminary Master Program Schedule (Concluded)

Program Milestones. The program milestones were based on an earliest Phase A start date of January, 1992, and a duration of nine months. The time reduction in the Phase A effort was considered possible when the results of this study are incorporated into the initial contract data package. The Functional Flow Block Diagrams for Production and Operations and Support are presented in Reference 4-1. The Phase B effort would be performed over a one and one-half year period to assure that the Phase C/D requirement documents and end item specifications are complete and mature. The Phase C/D activity will be discussed in more detail below.

Design, Development, Test and Engineering Schedules. With Phase C/D ATP in April, 1994, the program follows a conservative set of schedules in all functions until the final orbital flight test is completed in 2002. All functional program areas are reflected in the PLS Master Program Schedules (Figure 6-1).

Engineering - The engineering design effort is baselined in a number of Critical Design Review's (CDR's) between March, 1995 and January, 1996. The mock-ups necessary to validate and control the final design will also be developed between ATP and the completion of the CDR's. All related engineering analysis and required wind tunnel test to support the design development will be completed during this time frame.

Facilities - The facilities need for each of three major areas, Production, KSC, and JSC, are addressed in this section,.

Production: The production facility needs for the final assembly lines to support the fabrication schedule are presented. See Section 6.1.3 for additional detail.

KSC: The major new facilities for the PLS program are all located at launch site. The expanded schedules for each facility, horizontal processing facility, adapter processing facility, and deservicing/pyro facility are presented in more detail in Figures 6-2 through 6-4.

JSC: The facility requirements at JSC for Mission Operations are more an expansion of existing facility instead of construction of new facilities. The Mission Operations support requirements are supported by the schedules. See Reference 4-1 for additional data.

Tooling - The major tooling requirements for the welding and final assembly lines are presented. The lead, production and set up time for tooling was based upon experience with the Sabreliner, B-1B, X-31 and Shuttle programs. The tooling schedules support the fabrication need dates.

Procurement - The major material procurements data are presented alone with the LRU and equipment need date for the major test articles. The lead times presented are consistent with those presently being experienced on the Shuttle program.

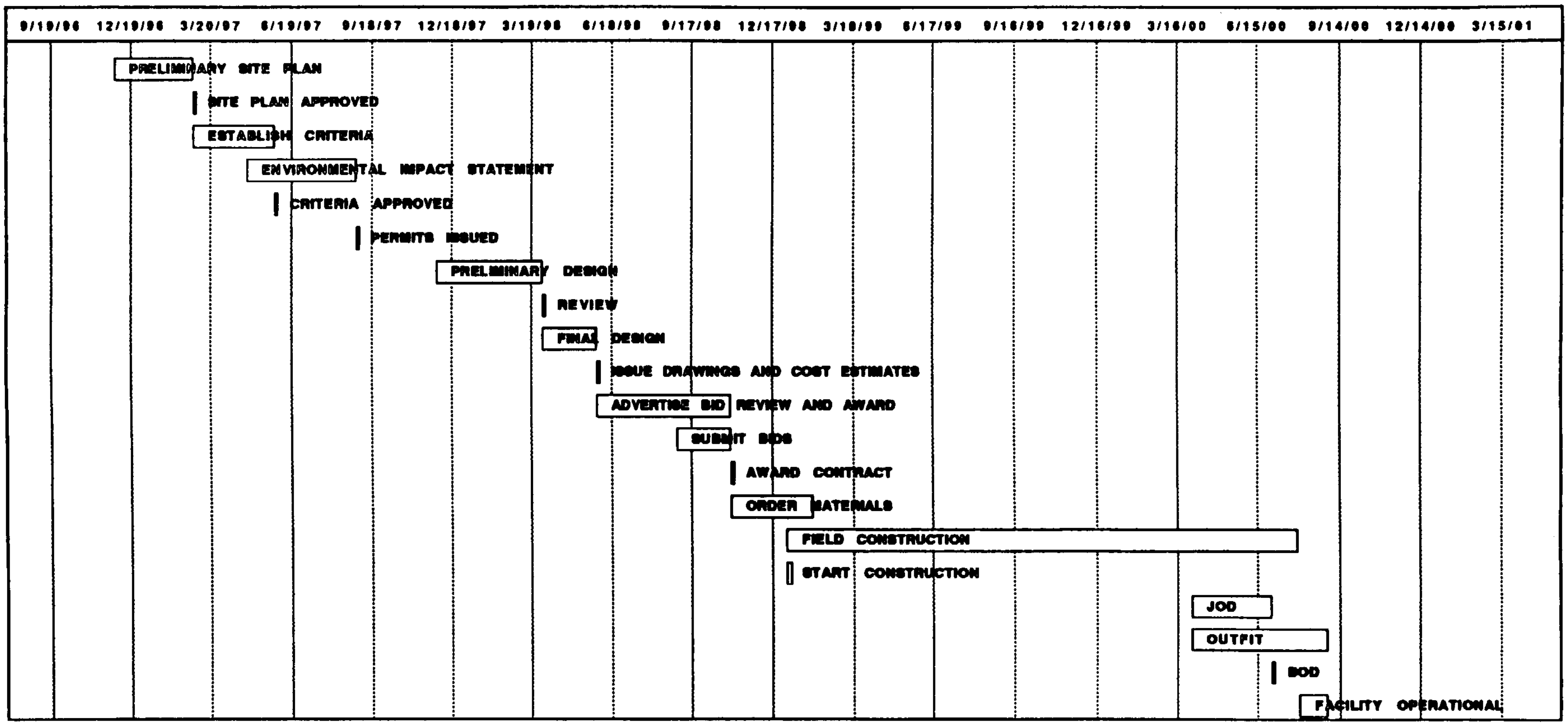


Figure 6-2. Horizontal Processing Facility Construction

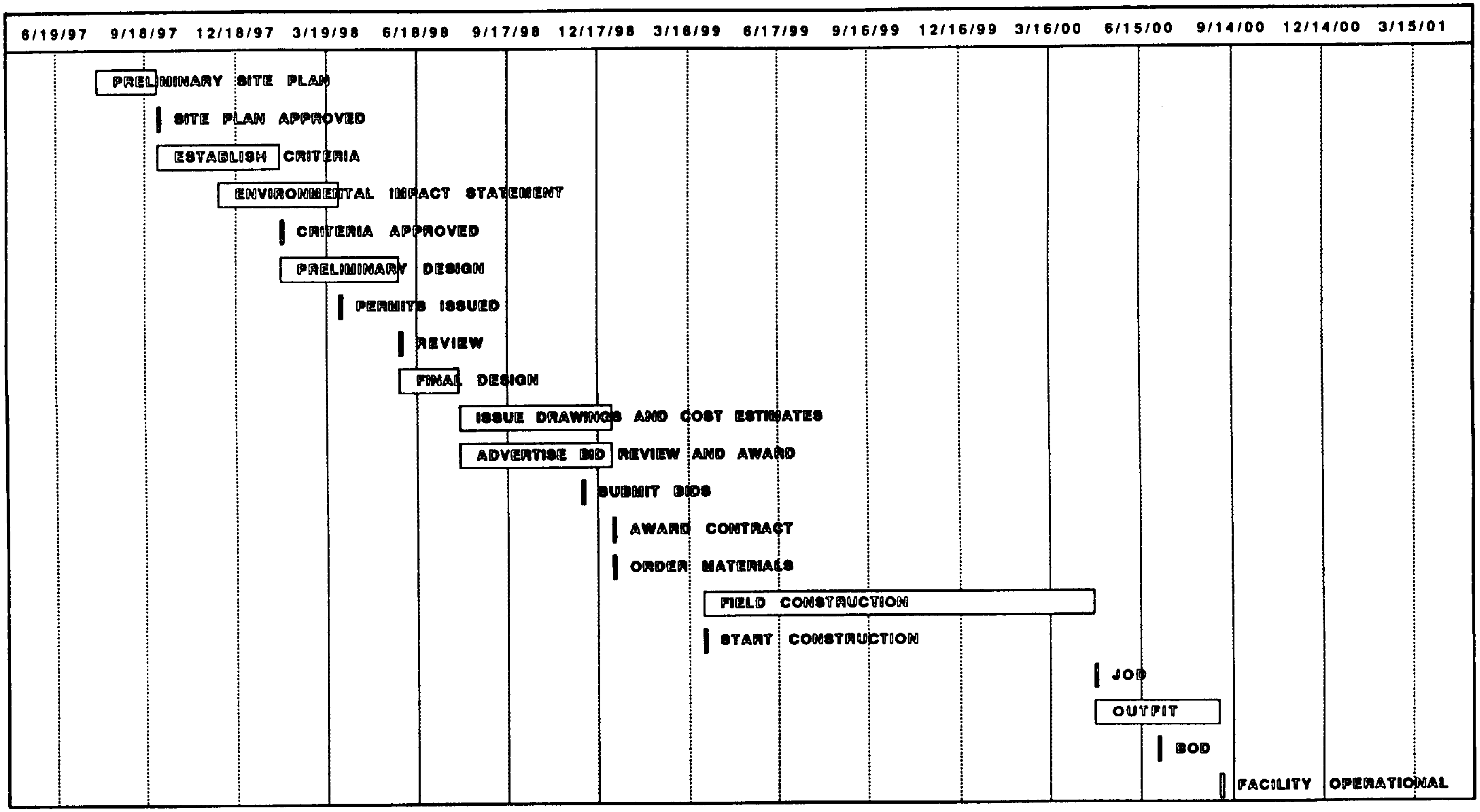


Figure 6-3. Adaptor Processing Facility Construction

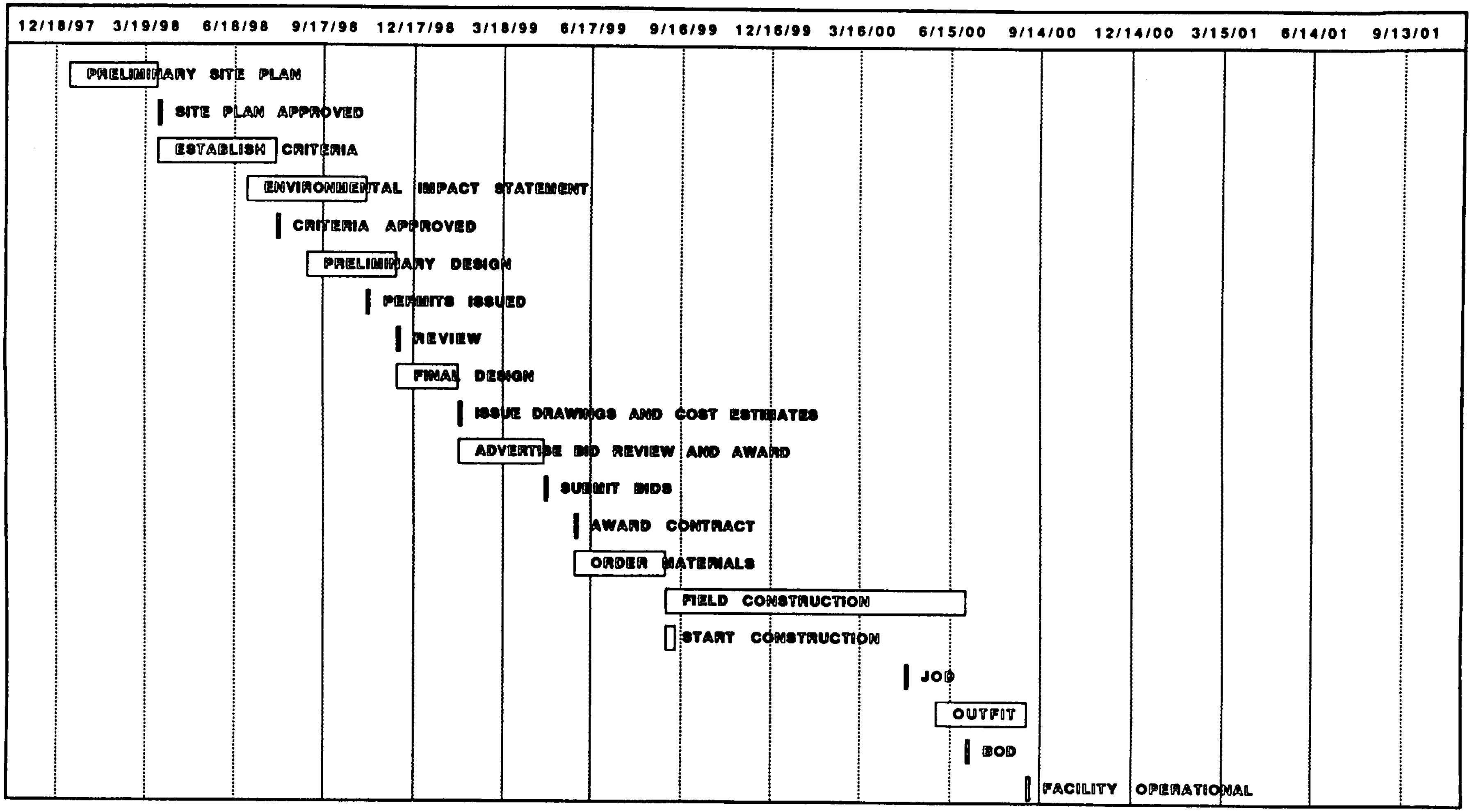


Figure 6-4. Deservice and Pyro Facility Construction

Mission Operations - The capability to develop, plan and conduct a mission requires the development of NASA/JSC capabilities to provide mission design, training and monitoring and control during the actual mission.

Operation System/Tool Development - The major systems, Flight Design System, Crew Activity Plan (CAP) System and Control Center (CC) will be on line and ready to support the first OFT mission on January 1, 2001.

Training Resources Development - The following functions will be available to support not only crew training but the training of flight controllers and instructors.

Training requirements and material development
Computer based training
One G trainer
PLS Simulator
PLS training aircraft

Ground Test - The component development tests are scheduled to be completed prior to any subsystem installations in any major test articles. All Avionics Development Laboratory tests are to be completed prior to Approach and Landing Tests (ALT) and Operational Flight Tests (OFT) flights, including the flight software.

The sub-system development tests will be completed prior to the first subsystem installation into a flight test article. The associated subsystem qualifications testing and completion of analysis of the results for the ALT an OFT flight will be complete prior to the schedule IOC fight date.

All test articles in the PLS program have multiple usage and each is discussed below:

Static Test Article (STA) - After completion of the required static test to confirm the design loads, the test loads will continue to be incrementally increased until failure in major structural areas occur. The static test article will be used to perform the water floatation tests after the static load testing is complete. Any damage resulting from the static load test will have to be repaired to insure that the crew cabin will maintain its pressure integrity. A possible future trade would be to determine which of the two vehicles, dynamic or static would best satisfy the flotation test requirements, since static testing will damage the test vehicle.

Dynamic Test Article (DTA) - The dynamic vibro-acoustic and modal tests will be followed by a Thermal Vacuum Test (TVT). This vehicle will contain very few subsystems and cannot be manned during the TVT. The test will allow verification of the thermal math models and prove-out the passive cooling system design with the use of thermal simulators.

Flight Test - The flight test program is composed of tests of the parachutes/launch escape system, approach and landing performance, and orbital flight tests. These tests use the following test articles.

Boiler Plate Flight Test Articles (BPFTA) - The four boiler plate vehicles will support all the parachute drop test and launch escape system test to verify and qualify the launch escape system, separation system, and parachute design. A total of 25 three-parachute tests will be performed (some each on land and water), 23 of which will be drop test and two launch escape system motor firings and dynamic parachute deployment at maximum altitude. Some damage is expected and if need be, some repair will be performed on the boiler plates to insure the completion of the parachute and launch escape system test on schedule.

ALT Vehicle - The ALT test program includes full-scale low-speed tests launched from a B-52. These test include unpowered subsonic and landing tests. Supersonic drop test may be possible with some solid booster thrust augmentation, but were not addressed during this study. The tests are designed to verify the atmospheric handling qualities, landing performance, and the guidance and control and autoland systems.

Orbital Flight Test Vehicle (OFTV) - The orbital flight tests complete the flight test series by verifying the overall capability of the full system. The orbital flight tests comprise six flights, two of which are contingencies to assure that data from four flights are received. The Orbital Flight Test Vehicle (PLV-01) will be refurbished and added to the operational fleet. The mission planning elements to be in place for the first OFT flight are presented on Figure 6-5. Mission preparations for this flight are presented on Figure 6-6.

<u>Flight Test Program</u>. The parachute, PLS vehicle and system (including the launch vehicle) will be tested and verified during the following flight test program.

Parachute, Water Impact and LES Tests - The following qualification sequence for the PLS parachute system was obtained from Pioneer, the developers of the orbiter drag chutes. The PLS parachute design is based on a existing design, which is only sized to satisfy the PLS requirements.

Five bomb drops and 25 full three parachute tests are scheduled in the preliminary development schedules. The bomb drops would be with single parachutes and a dead weight equal to one-third of the expected PLS maximum weight. All parachute drop tests will be made from a large type air

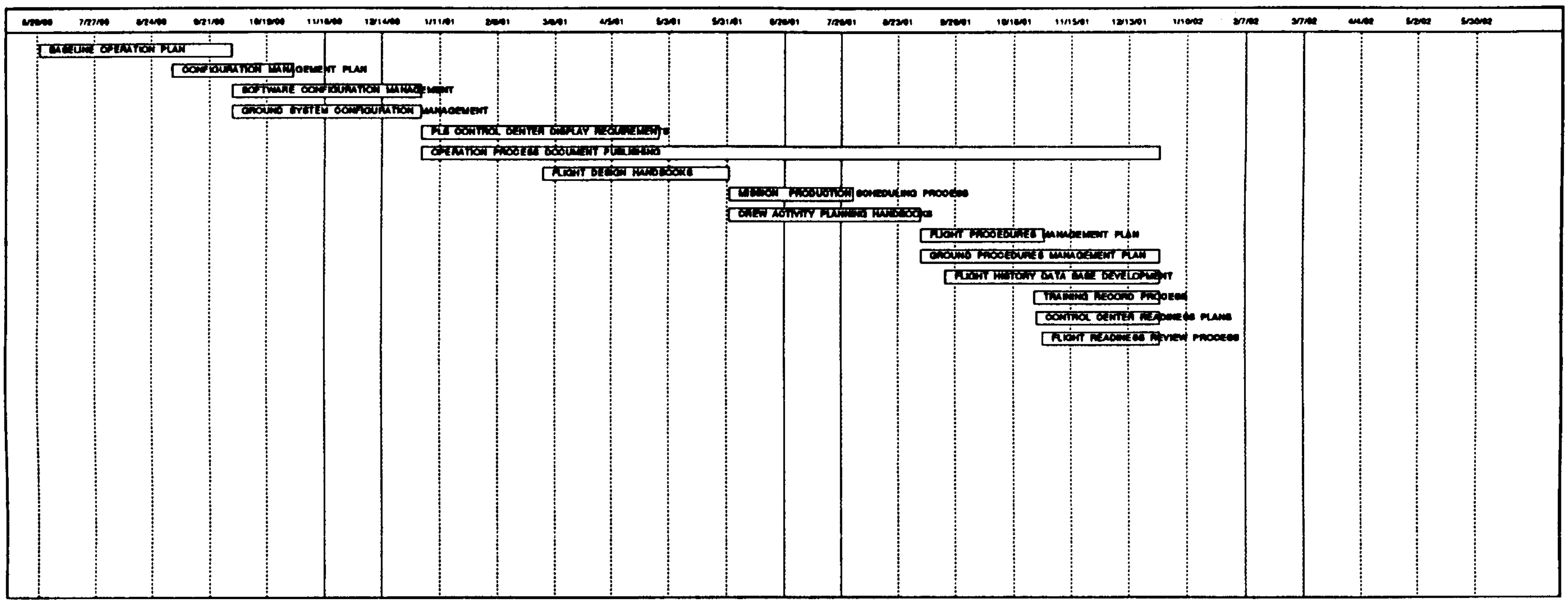


Figure 6-5. Mission Planning Preparations

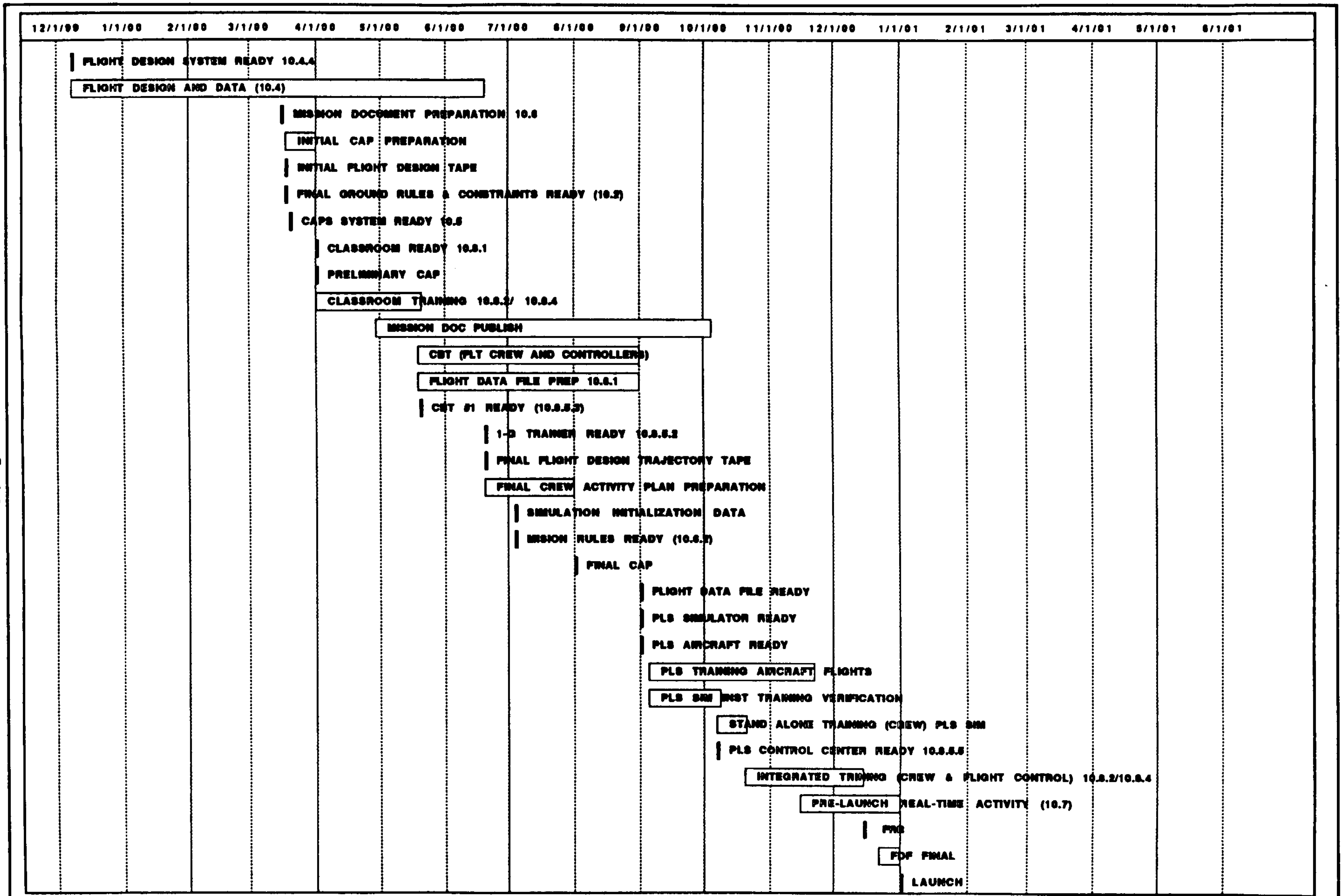


Figure 6-6. OFT-1 Mission Preparation Schedule

transport, like a C-5 or C-17.

Twenty-three of the three parachute tests will be drop tested with full up parachute system, motors, drogues, and parachutes. These tests will use one of four PLS boiler plate vehicles, which will have appropriate instrumentation for the drop tests. The boiler plate PLS vehicles will be repaired as required to complete the parachute test program. To demonstrate robustness in the parachute system design, two chute drops, simulating a parachute failure, would be performed. Water impact tests will be performed in an appropriate water tank facility following the parachute drop tests.

Two of the 25 three parachute tests will be part of two full up Launch Escape System tests, which will include an instrumented PLS boiler plate vehicle, full booster adapter and systems, SRM's and parachutes and their systems.

The PLS boiler plate test vehicles will be available to perform other tests and fit checks as they become defined by definition of the PLS system follow-on studies.

Approach and Landing Test (ALT) - The ALT program validates the following PLS system capabilities in a very controlled environment.

Autoland Performance
Landing Gear and Brake Performance
Low Speed Aerodynamic Control Authority
Cross Wind Landing Sensitivity
C. G. Envelope Sensitivity
Maximum Weight Vehicle Performance
Final Approach Energy Management

The ALT test article will be a PLS flight vehicle with additional instrumentation and the following non-functioning systems removed:

Propulsion systems
Docking elements and systems
Portions of the ECLSS and power systems
Passenger seats and provisions

The PLS vehicle will also have a unique crew cabin since it must interface with a pylon on the B-52 wing. Also one ejection seat will be installed to provide abort capability below parachute capability. This vehicle was not considered for refurbishment, but this question should be addressed during some future trade study.

The PLS design can only accommodate one ejection seat due to space limitations, two seats would block access to the cockpit by the crew. Designs to allow moveable or removable ejection seats would have high design, qualification, and program risk. This could be a future area where a trade study could be performed.

The following proposed ALT flight test profiles are presented for evaluation and are not meant to be final or firm.

Flight No. 1 - Straight in landing with no maneuvering using autoland, light braking and little or no cross wind.

Flight No. 2 - Maneuvering with autoland, hard braking and nominal cross winds.

Flight No. 3 - Maneuvering with crew landing, hard braking and maximum cross wind.

Flight No. 4 - Straight in landing with aft C.G. and crew landing, hard braking and nominal cross winds.

Flight No. 5 - Maneuvering with aft C.G. and crew landing, hard braking and nominal cross winds.

The ALT flight test program can take on various configurations of flight operations per flight. The above complement is just one set to consider. Once the ALT flights are completed, the ALT flight test vehicle will be shipped to KSC to provide path finder checkout of the Horizontal Processing and De-servicing/Pyro facility. The ALT program will allow the first OFT flight to be manned without ejection seats.

Orbital Flight Test (OFT) - The OFT program of the PLS vehicle verifies it is operational by validating the following analytical models developed to describe flight performance and environment:

Aerodynamics
Thermal - TPS/TCS*
Vibration/Acoustics*
Load*
Venting
Aero/thermal

Those models marked with a "*" are limits the vehicle cannot exceed in flight since they have been verified by ground testing. The OFT program also establishes crew confidence in the PLS flight worthiness design, operations, performance and handling quality.

The test results from each flight may result in changes: to the control loop lead, lag or gain; to follow-on flight test requirements; or operational flight limits. Final test results could also effect the subsystem design, like supplemental or reduced TPS requirements in local areas.

Each OFT flight will accomplish a number of tests in addition to collecting data to verify engineering math models. Proposed flight plans for four OFT flights are presented in Table 6-1. Special test requirements for ascent, orbit and entry/landing of each of the four missions are also included in this table.

Table 6-1. Preliminary OFT Plan.

	ORBITAL FLIGHTS*			
	1st	2nd	3rd	4th
FLIGHT REGIME (28.5 DEG INCL)	• MANNED (2 CREW) • BENIGN LAUNCH • 24 HOURS • MIN. CROSS WIND LANDING AT EDWARDS AFB • NO BRAKING • NOM C.G.	• MANNED (2 CREW) • 24 HOURS • NOM CROSS WIND LANDINGS AT EDWARDS AFB • BRAKING • FORWARD C.G.	• MANNED (4 CREW) • 72 HOURS • NOM CROSS WIND LANDINGS AT EDWARDS AFB • HARD BRAKING • FORWARD C.G.	• MANNED (4 CREW) • 72 HOURS • AFT C.G. • NOMINAL CROSS WIND LANDING AT KSC • HARD BRAKING • MAX WEIGHT
SPECIAL FLIGHT TEST REQUIREMENTS ASCENT:	NONE	• ELV INDUCED $I\sigma\alpha$ & β	• ELV INDUCED $I\sigma\alpha$ & β	NONE
ORBIT:	• OMS/RCS MANEUVERS	• OMS/RCS MANEUVERS • TOP, BOTTOM, TAIL TO SUN	• OMS/RCS MANEUVERS • RENDEZVOUS • SSF DOCKING OPTION	• RENDEZVOUS • SSF DOCKING/ BERTHING • SOLAR SOAK • SSF RMS ATTACHMENT VERIFICATION
ENTRY/LANDING:	• CROSS RANGE TO EDWARDS AFB <700 NM • AUTOLAND	• AERO MANUEVERS • AUTOLAND	• AERO MANEUVERS • AUTOLAND	• AERO MANEUVERS • LARGE CROSS RANGE • PILOT LANDING

* TWO ADDITIONAL ORBITAL TEST FLIGHTS REQUIRED FOR CONTINGENCY

For math model verification, four flights of data are required. Six flights have been scheduled in the Preliminary Master Program Schedule, as contingency to assure that four flights of data are obtained. Test requirements areas presented in Table 6-2 are established by the functions listed in this table. These functional areas establish the majority of the OFT flight test requirements.

The PLS test article will be a production vehicle with a special data and communication system installed and six passenger seats and provisions replaced by batteries and ECLSS consumables. These modifications will satisfy the collection of post test flight data requirements and the 72 hour mission capability. Even with additional batteries, a power down configuration during sleep cycles will have to be performed to accomplish a 72 hour mission.

Center of gravity control during the flights will be by ballasting. Also, all launches should be into a rendezvous compatible orbit with the Space Station Freedom to provide an additional abort mode during the test flights. The OFT vehicle will retain a portion of the data system for finalizing any OFT open items when reconfigured for operational flights.

Fleet sizing. Fleet sizing analyses provide the definition of the fleet size that will have to be produced during the

Table 6-2. Functional Areas Providing Test Requirements.

THERMAL/STRUCTURE:	SUBSYSTEM THERMAL DESIGN, TPS/TCS CAPACITY ASSESSMENT, MOLDLINE PENETRATION CONTROL
THERMAL/AERO:	ASCENT, ENTRY, TRANSONIC HINGE MOMENTS, TPS, LOCAL FLOW SEPARATION
STRUCTURES:	LOAD & STRESS EVALUATION, SHOCK/VIBRATION/ ACOUSTICAL, VENTING
PROPULSION:	PERFORMANCE, OPERATIONS, VIBRATION, STABILITY
MECHANICAL:	PLUME IMPINGEMENT, LANDING GEAR/BRAKING/STEERING
ECLSS:	PERFORMANCE, CABIN TEMP/HUMIDITY SURVEY, FLASH EVAPORATORS, POST LANDING CABIN CONTROL
AVIONICS:	GN&C PERFORMANCE, TELE & COMM, DATA STORAGE, TRACKING, AUTOLAND ANTENNA PATTERNS, ON-BOARD HEALTH MONITORING ASSESSMENT
CREW/PASSENGERS:	ACOUSTICAL NOISE, CABIN ATMOSPHERE, HABITABILITY, D&C COMPATIBILITY, WASTE MANAGEMENT
MISSION:	DOCKING/PROXIMITY OPERATIONS, CROSS RANGE, MISSION DURATION, PERFORMANCE, LAUNCH/FLIGHT/ENTRY CONDITIONS
POWER:	BATTERY PERFORMANCE/STABILITY

manufacturing production run. The traffic and personnel exchange models provided by LaRC can be satisfied by eight missions a year with one vehicle. This Section presents the analysis results and justification for a final fleet size of four vehicles, when attrition and continuous support are considered.

Assumptions - The resulting fleet size was based on the following assumptions:

- Space Station crew stay of 180 days
- Initial Orbiter crew exchange every 90 days
- Full Space Station crew (eight or more) change out would exceed the Shuttle Capability
- Space Station operability requires crew overlap
- Lunar/Mars personnel transferred on dedicated flights
- All crew rotations performed by the PLS when the Space Station crew size reaches 24

The most efficient use of the PLS is to exchange eight crew on every flight. With 24 crew at station, six flights with eight passengers each will exchange the 48 personnel required to keep the Space Station fully staffed and not exceed the 180-day rotation requirement. One PLS flight every 60 days.

Traffic model summary - The flight rate build up after the IOC flight in August, 2002, is presented in Table 6-3. This flight rate support the personnel traffic model, assuming orbiter support in the early years, with six eight passenger flights in 2007 for Space Station crew exchange and two four passenger flights for Lunar/Mars crews. By 2009, the Lunar/Mars crew requirements grow to two eight passenger flights per year. 141 flights of the PLS occurs by the end of 2020 with the above considerations. Further analysis to integrate the PLS and orbiter flights in the earlier year (2002 to 2008) may reduce the number of flights required by the PLS during these seven years.

Table 6-3. PLS Flight Rate Build-up.

YEARS (FISCAL)	SSF FLIGHTS PER YEAR	LUNAR/MARS FLIGHTS PER YEAR	PASSENGER SIZE	TOTAL FLIGHTS
2002 (IOC Aug 2002)	1		4 - 8	1
2003	4	2	4 - 8	6
2004 & 2005	5	2	4 - 8	14
2006 - 2008	6	2	4 - 8	24
2009 - 2020	6	2	8	96
			TOTAL:	141

System Attrition - In-flight aborts of the PLS vehicle do not necessarily denote loss of vehicle. Catastrophic vehicle loss represents a small fraction of the total number of aborts, and approaches zero. The current Mission Success reliability forecasts for PLS, from the study, is 0.978. At present, no single point failure has been identified in the PLS design that will cause loss of vehicle. However, a number of PLS failures, occurring singularly or in combination with others, may cause mission abort. Should this happen, the PLS simply would safely return to earth for subsequent repair and reuse.

PLS loss also may occur as a result of catastrophic failure of the Titan IV or other Launch Vehicles. The most recent projections for Titan IV Launch Vehicle Reliability would result in 2 to 4 loss of mission failures per hundred launches. The PLS should be capable of escaping from all but a no-warning Titan IV explosion occurring on the launch pad or in flight. A Titan IV no-warning explosion is considered to be a very unlikely event. Accordingly, it is unlikely that a PLS loss of this type will occur at a rate greater that one per thousand launches. The assumption for vehicle loss, but not crew loss is one per hundred launches, resulting from water landing or runway landing damage.

Fleet size - The fleet size is determined by the mission requirements and realistic attrition estimates.

One PLS vehicle will satisfy the current traffic model requirements.
The attrition rate of one vehicle loss per hundred flight, requires two additional vehicles for the 141 flights by the year 2020.
One PLS backup vehicle to allow continuous mission operations under any circumstances.

A fleet size of four vehicles will satisfy both the traffic model and a realistic attrition rate, plus continuous support. All four PLS vehicles, in the fleet, would be flown to support the traffic model to maintain full operability of the fleet. Non-use of a vehicle has been proven to create problems that are not understood and result in unexplained anomalies. The traffic model can be satisfied by four vehicles using a 41 day calendar turnaround schedule, which is the minimum manpower requirement per reference 5.

6.1.2 Production

Manufacturing and system validation plans identify the production requirements, time lines (critical paths), issues/risks, facilities (requirements and recommendations), major equipment (including engine test stands, mock ups test beds, iron birds, and simulations laboratories), testing and test articles, and integration approaches for the PLS.

Acquisition Phase. The PLS objective is to design a safe, durable, low life-cycle-cost vehicle. Obtaining this objective starts by emphasizing producibility and maintainability in the preliminary design concepts. The design will be driven by operations and maintainability requirements and assured by an integrated system engineering, a total quality management approach and an integrated reliable & maintainable process.

The first efforts associated with the development of our operations concept were to develop a series of functional flow block diagrams (FFBD's) that would capture the operational functions associated with the PLS. The addition of the DDT&E blocks associated with "capabilities development" and operational flight test (OFT) verification provided the important links to the pre-production and operational periods that are necessary ingredients in our "design for operations" philosophy. Operations lower level flows are found in Reference 4-1.

PLS program management has placed operations, maintainability and producibility in priority position of importance. This system will be producible within the boundaries of being first maintainable and operable. The key word is "access"! See Figure 6-7. The best examples of this are: the removable heat shield; the exterior systems access panels; and the manufacturing access opening in the crew cabin. Back-face heat shield removal during the operations phase of the program is assured by not installing the heat shield until after all the systems are installed on the vehicles exterior. Therefore, if the heat shield can be installed last it will be able to be removed first, if and when required. The removable systems access panels provide access during manufacturing and during operations.

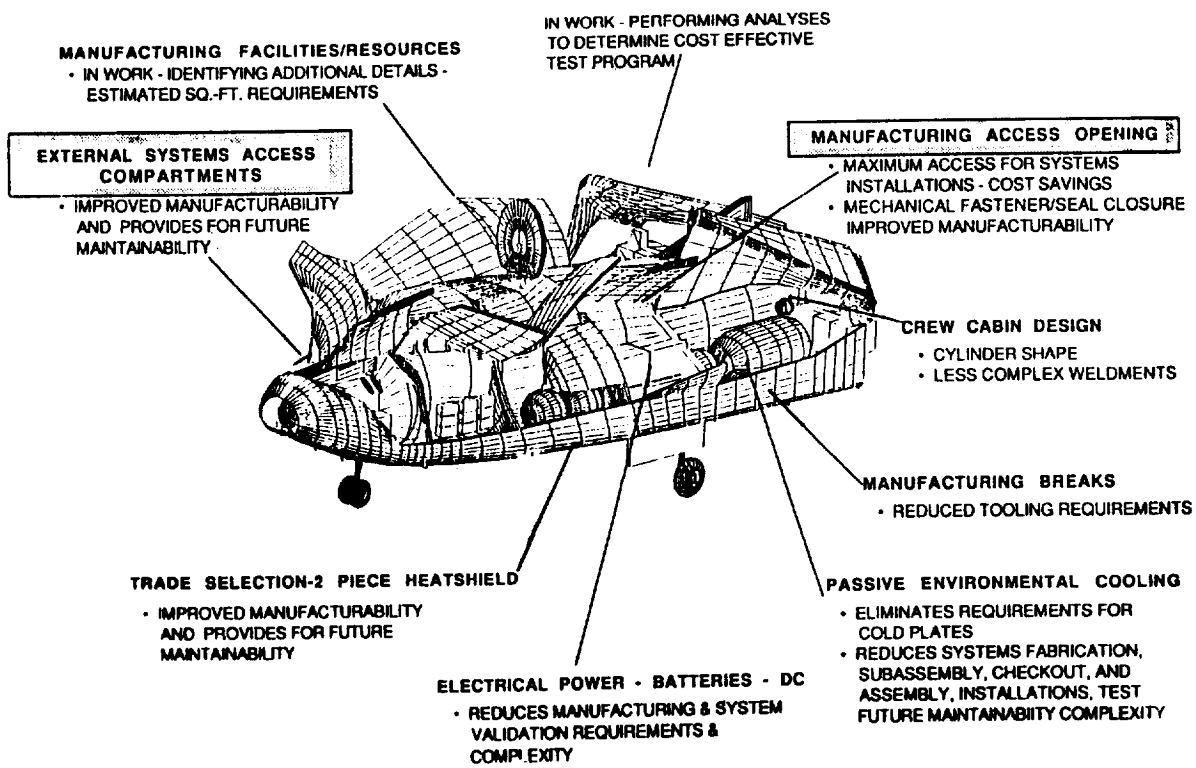


Figure 6-7. Producibility Influence on Vehicle Design

The manufacturing access opening in the crew cabin will provide significant intangible benefits to the PLS Program, as a similar access opening in the Space Shuttle Orbiter crew module. Additional benefits can be derived through the mechanical closure/opening, if and when it would be required to disassemble the transfer tunnel from the crew cabin. Improved manufacturability through producibility of design is enhancing future operations maintainability.

Fabrication of all PLS vehicles and test articles in one production run is cost-effective for the program, since each Shuttle Orbiter was built with a personnel turnover rate of 70 percent. Continuous build will require only one facility and rate tooling setup, a minimum amount of retraining, and a one-time procurement of items including those with long lead times. Early planning will assure the operational spares requirements are included in the production order. The cumulative results of these actions will result in a cost effective manufacturing program and would support DRM-1 requirements.

Fast turn around requires accessibility. To comply with that requirement, most avionic systems and other systems historically requiring operations attention are located on the exterior crew cabin structure, within accessible exterior compartments. In addition to accessibility, the systems will use mature, state-of-the-art techniques, including self-test. To further comply with the accessibility requirement, the heat shield back face structure is a removable two piece structure, as previously stated. This enhances access and provides the means for lower vehicle access during manufacture and for field operations maintainability and inspection.

The electric system is direct current, thus simplifying or eliminating heat-producing conversion devices. The actuation systems are electro-mechanical, avoiding APU/hydraulic problems that have plagued the Shuttle.

Manufacturing Flow and Build Concept. The crew cabin fabrication illustrated in the Manufacturing Flow and Build Plan, Figure 6-8, begins with single degree forming of aluminum sheet stock. This is followed by the necessary machining operation prior to welding the sub-assembly pieces together. Circular design of major sections reduced the tooling complexity and reflects the manufacturing influence on design to minimize production, operations and cost. The PLS preferred concept incorporates welded ring stiffeners and mechanically attached frames for operational maintainability and ease of inspection. The transfer tunnel is fabricated separately and mechanically attached to the forward crew cabin, thus allowing easy manufacturing access to the crew cabin for interior fabrication prior to attachment of the transfer tunnel. The crew cabin assembly is completed with installation of the rear carry through frames. The bottom heat shield is not attached until all the sub-system elements have been installed in the systems compartments and the systems individually checked out. These methods of attachment/assembly will also facilitate future operations maintainability.

Manufacturing Master Schedule. The Manufacturing Master Schedule, Figure 6-9, was developed using program-level milestones that support customer-indicated requirements and an analysis of integrated task time estimates. The analysis establishes optimum support for the manufacturing program in all areas, including engineering, facilities, material procurement, manpower loading and tooling, and application of comparative measurements of historical Space Shuttle performance.

System support hardware will be fabricated via a blended schedule to maintain systems used throughout the vehicle. All other scheduled bars are major components and hardware groups of overall vehicle and stand alone with their own flow plans, such as the crew cabin in Figure 6-10. Final assembly and checkout will be the point in time when the major vehicle components are mated, allowing systems integration and subsequent testing. Flow plans will be established for each major component/hardware group shown on the master schedule, providing an orderly time phasing for

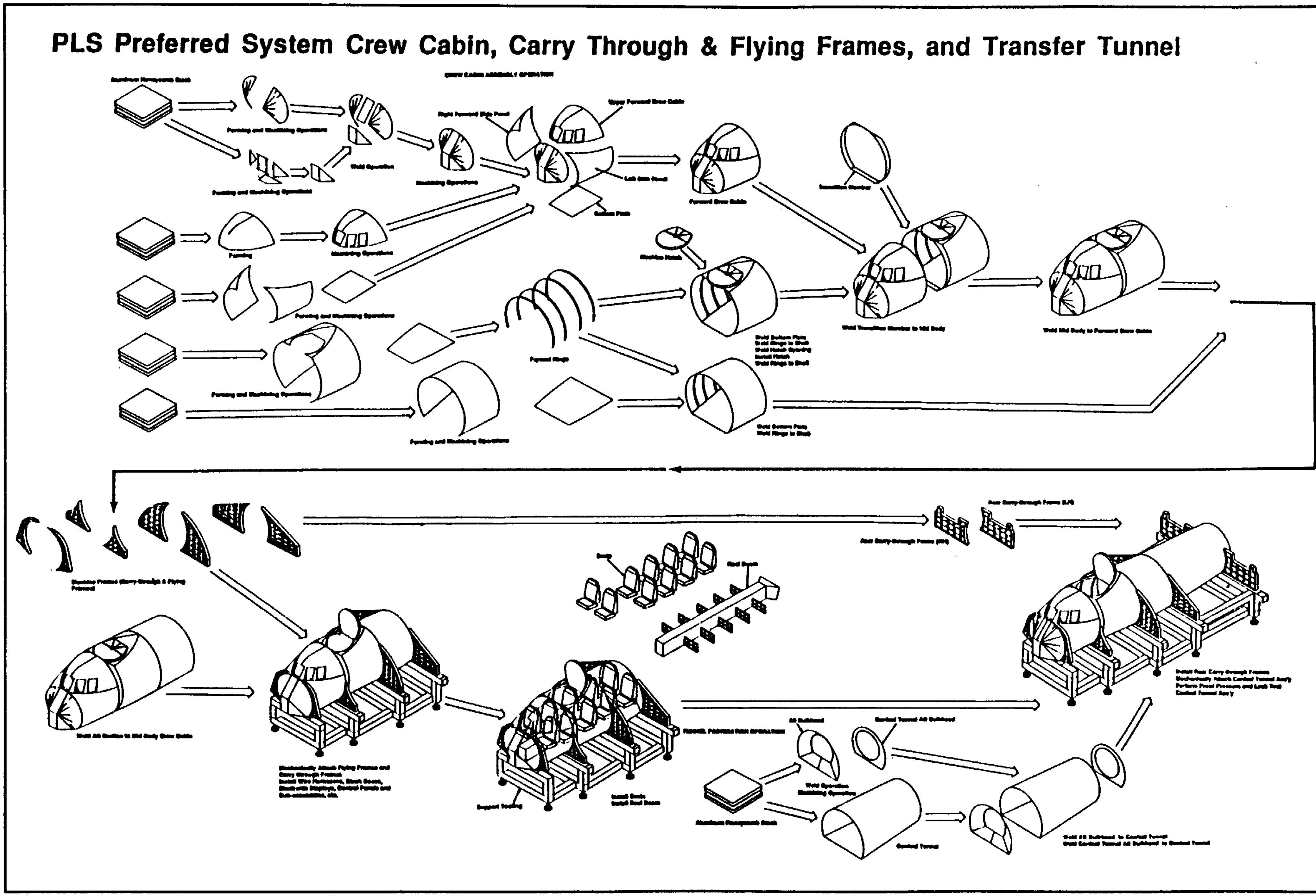


Figure 6-8. Manufacturing Flow and Build Plan

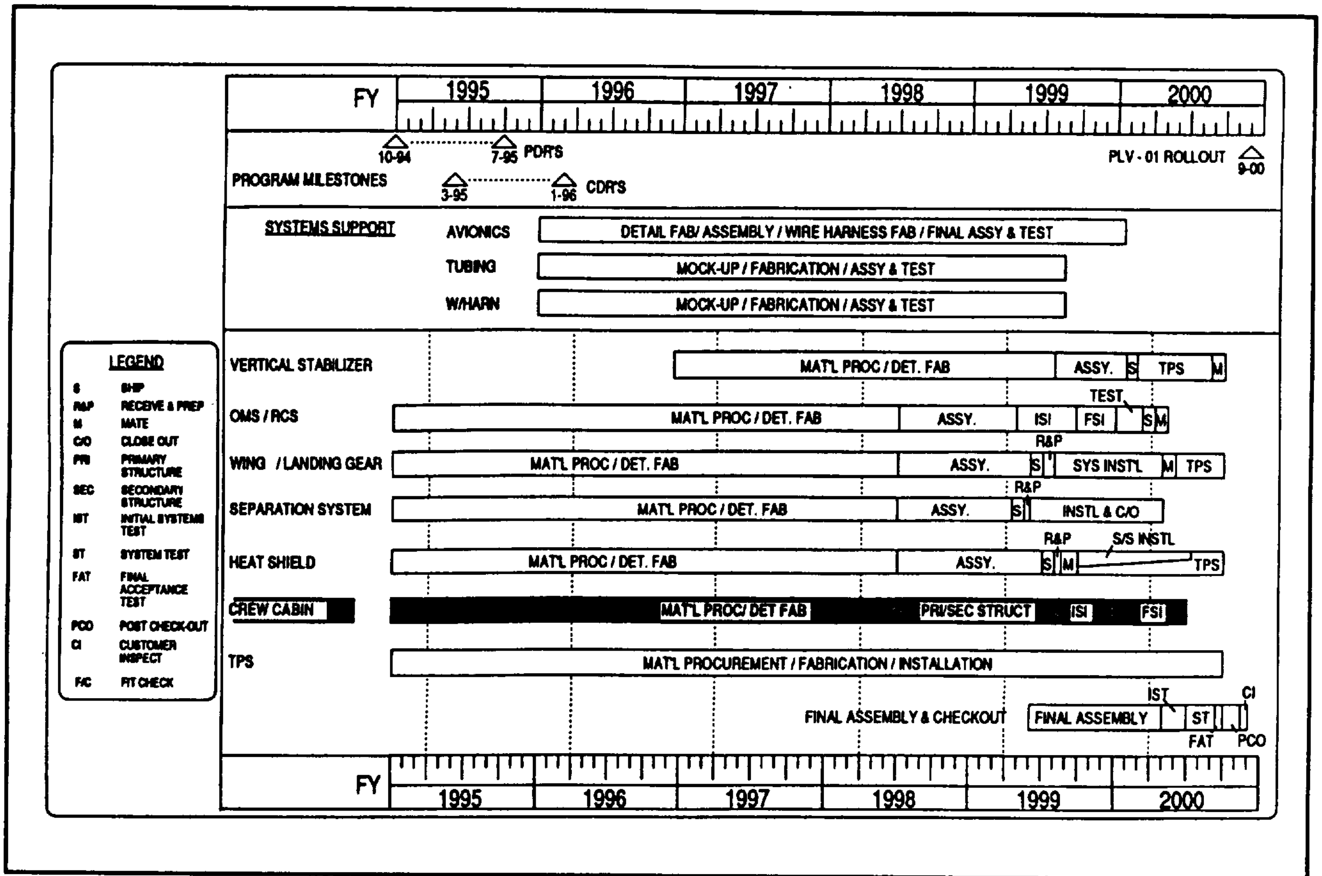


Figure 6-9. Manufacturing Master Schedule

Manufacturing production activities. The schedules developed will serve as the basis for manpower loading, material need, and facility and equipment usage. Optimum control points will be established to provide performance controls, change control, and status information to be measured against the master schedule.

6.1.3 Test and Verification

Manufacturing and fabrication of flight hardware includes the verification of system operation, both individually and integrated. The system will be validated during the operational phase of the contract. The following definitions are being used by the study:

Verification - All tests (and/or checkout) performed prior to validation of the system

Validation - Certification of the system performed during the operations phase of the contract (such as space vehicle flight readiness review sign-off or flight worthiness-aircraft certification)

Philosophy. The testing philosophy is to achieve system validation without overkill. The aircraft industry approach to verification and validation is being closely reviewed to determine the most efficient and effective manner of achieving validation.

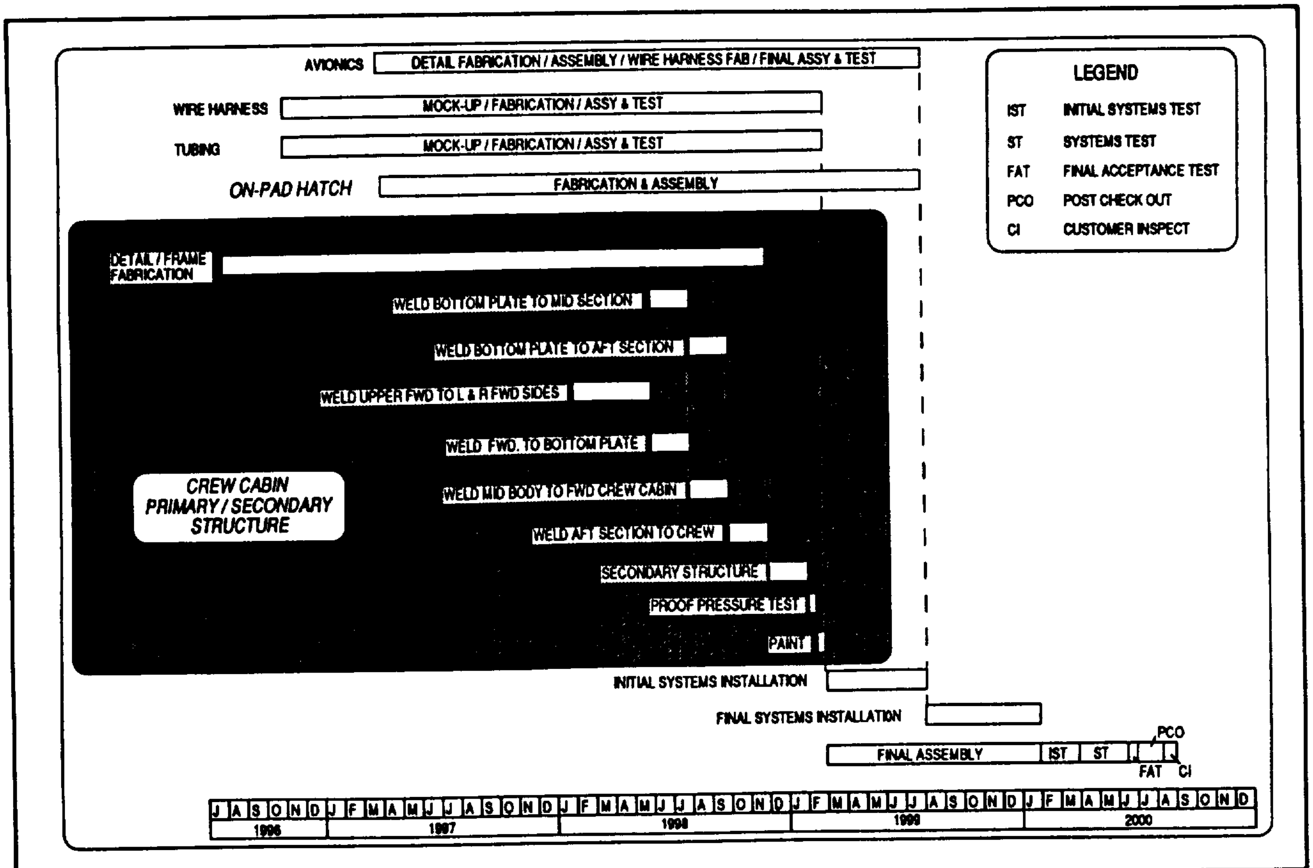


Figure 6-10. Crew Cabin Flow Plan

Emphasis on development testing will provide assurance of a sound product and high confidence in a successful qualification test program. A high-fidelity integration test will lead to a flight worthy, low-risk system. Flight testing will be orchestrated to provide evidence that flight boundaries and requirements can be safely achieved. Guided by lessons learned, the testing concept developed for PLS supports the basic philosophy of assuring a safe, durable, cost-effective PLS.

PLS preferred concept sub-system test article requirements are presented in Figure 6-11 for development, qualification, integration, and the approach and landing tests.

Facilities. Based upon the information in the work break down structure, the MFBP's, the recommended test plans, available make/buy information, and the program and manufacturing schedules, potential major sites and facilities for manufacturing and test are identified in Tables 6-4 and 6-5. The level of effort did not include cost of the facilities/modifications or new facilities construction. Basic initial facilities area requirements for: Rockwell International, Space Systems Division: Downey and Palmdale, CA, and Kennedy Space Center (KSC), FL; North American Aircraft: Tulsa, OK, Palmdale and El Segundo, CA; Johnson Space Center (JSC), Houston, TX; Edwards Air Force Base (AFB), CA; and White Sands, NM were determined and are shown in Table 6-6.

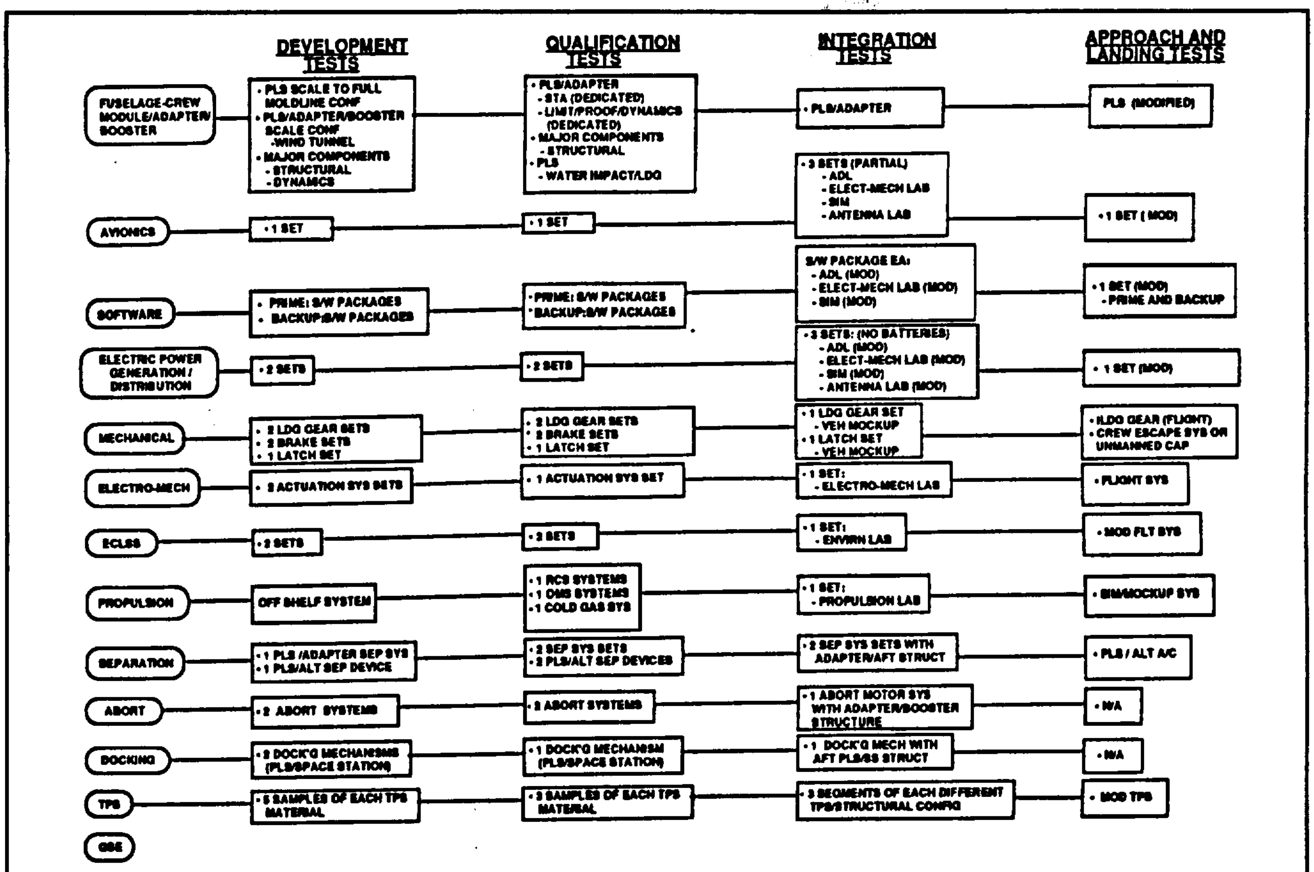


Figure 6-11. Test Article Requirements for System Validation

A successful, cost effective PLS program starts with a design that has demonstrated it can meet the requirements. The demonstration process will take the forms of analysis, similarity, demonstration, and test. Tests and demonstrations require test articles that support the verification process. These test articles will range from simple, built up segments to a complete vehicle demonstrating compliance of a totally integrated system.

Development tests generally will consist of testing subassemblies, complete assemblies, and major components, providing confidence that the test article subjected to the qualification test will successfully meet requirements. Most of the structural articles undergoing qualification tests will be rendered non-flight worthy and thus classified as test dedicated. Mechanical and avionic equipment undergoing qualification testing will also be classified as non-flight worthy without major refurbishment. Integrated testing will be orchestrated to demonstrate that the systems perform together as required under stipulated environment.

The Approach and Landing Test (ALT) for the preferred program entails dropping a modified PLS vehicle from an appropriate aircraft to demonstrate approach and landing qualities. The ALT will involve two forms of approach: free-fall without power and free-fall with add-on rockets to provide a supersonic approach.

Table 6-4. Manufacturing and Operations Facilities

KEY: FABRICATION = F ASSEMBLY = A TEST = T OPERATIONS = O

FACILITIES - TYPES	REMARKS
FABRICATION	
• STRUCTURES • DETAILS • COMPONENTS	• DOWNEY/LAD/TULSA/BUY-TBD
• SYSTEMS (HW/SW) • DETAILS • COMPONENTS	• DOWNEY/ANAHEIM/GOV'T FURNISHED EQUIP/BUY-TBD
ASSEMBLY/INSTALLATIONS	
• SUBASSEMBLY • STRUCTURES • DETAILS • COMPONENTS • SYSTEMS • DETAILS • COMPONENTS	• DOWNEY/TULSA/PALMDALE/BUY-TBD
• ASSEMBLY • STRUCTURES • DETAILS • COMPONENTS • SYSTEMS • DETAILS • COMPONENTS	• DOWNEY/TULSA/PALMDALE/BUY-TBD
• INITIAL SYSTEMS INSTALLATIONS • FINAL ASSEMBLY • FINAL SYSTEMS INSTALLATIONS	• PALMDALE/KSC

FACILITIES - TYPES (Cont'd.)	REMARKS
TEST	
• WIND TUNNEL	• AVAILABLE-NASA
• SOFTWARE	• AVAILABLE-DOWNEY
• ELECTRONICS/AVIONICS	• AVAILABLE-DOWNEY
• SIMULATORS	• AVAILABLE-W/MOD DOWNEY
• LANDING GEAR/ACTUATIONS SYSTEMS	• AVAILABLE-W/MOD DOWNEY
• ENVIRONMENTAL CONTROL LIFE SUPPORT SYSTEMS (ECLSS)	• AVAILABLE-W/MOD DOWNEY
• PARACHUTE	• AVAILABLE-W/MOD WHITE SANDS
• ADAPTER (SEPARATION)(MID/FORE)	• AVAILABLE-W/MOD WHITE SANDS
• DEVELOPMENT TEST	• AVAILABLE-W/MOD DOWNEY
• PROOF PRESSURE TEST (CREW MODULE)	• NEW - DOWNEY/PALMDALE
• QUALIFICATION TEST	• AVAILABLE-W/MOD DOWNEY
• VIBRO ACOUSTIC TEST	• AVAILABLE-W/MOD DOWNEY
• STATIC TEST	• AVAILABLE-W/MOD PALMDALE
• MODAL TEST	• AVAILABLE-W/MOD DOWNEY
• THERMAL VACUUM	• AVAILABLE- SEAL BEACH
• AVIONICS DEVELOPMENT LAB (ADL)	• AVAILABLE-W/MOD DOWNEY
• PROPULSION TEST (OMS/RCS)	• AVAILABLE-W/MOD WHITE SANDS
• INTEGRATED SYSTEM TEST	• AVAILABLE-W/MOD PALMDALE
• ABORT (SEPARATION)	• AVAILABLE-W/MOD WHITE SANDS
• WATER BUOYANCY/IMPACT	• NEW-PALMDALE/JSC
• ORBITAL	• AVAILABLE-W/MOD KSC/EDWARDS
• TRAINER	• AVAILABLE-W/MOD JSC
• STRUCTURAL	• AVAILABLE-W/MAJ MOD PALMDALE

FACILITIES - TYPES (Cont'd.)	REMARKS
OPERATIONS	
• KSC LANDING STRIP	• AVAILABLE-STS, KSC, FL
• HORIZONTAL PROCESSING FACILITY	• NEW-KSC, FL
• KSC LAUNCH COMPLEX	• MOD EXPENDABLE LAUNCH VEHICLE (ELV) PAD 41 & KSC CENTER, KSC, FL
• DESERVICE FACILITY	• NEW BUILDING AT STRIP-KSC, FL
• TPS SUPPORT FACILITY	• PART OF HORIZ PROCESS FAC-KSC, FL
• FLIGHT DESIGN FACILITY	• NEW-HOUSTON, TX
• CREW ACTIVITY PLANNING SYSTEM (CAPS) FACILITY	• NEW-HOUSTON, TX
• PLS CONTROL CENTER FACILITY	• NEW-HOUSTON, TX
• PLS SIMULATOR FACILITY	• NEW-HOUSTON, TX
• 1-G TRAINER FACILITY	• NEW-HOUSTON, TX
• COMPUTER BASED TRAINER FACILITY	• NEW-HOUSTON, TX
• KSC ADAPTER PROCESSING FACILITY	• NEW-KSC, FL

07/12/90

Table 6-5. Hardware and Software Facilities Requirements

KEY: FABRICATION = F ASSEMBLY = A TEST = T OPERATIONS = O

MOCK-UPS/SIMULATORS/ TEST ARTICLES/COMPONENTS/ VEHICLES	H/W-S/W COMPLETE DATE	FACILITIES REQUIREMENTS: TYPE FACILITIES REQ'D	FACILITIES REQUIREMENTS: EXISTING/ LOCATION	FACILITIES REQUIREMENTS: MODS REQUIRED/ LOCATION	FACILITIES REQUIREMENTS: NEW/ LOCATION
• MOCK-UPS	12/90	F/A	DOWNEY		
• CREW MODULE - INTERIOR/HATCH		F/A	DOWNEY		
• TUNNEL/HATCH		F/A	DOWNEY		
• TBD					
• TRAINERS		F/A/T			
• ONE-G	04/98	F/A/T		JSC	
• NEUTRAL BUOYANCY (WET)	[illegible]	F/A/T		JSC	
• HIGH-FI SIMULATOR	[illegible]	F/A/T		JSC	JSC
• TEST ARTICLES		F/A/T		DOWNEY/JSC	
• DEVELOPMENT COMPONENTS	10/97	F/A/T			
• ADL HW/SW - AVIONICS	03/90	F/A/T	DOWNEY		
• ADL INTEGRATION/QUAL	[illegible]	F/A/T			
• SUBSYSTEM	[illegible]	F/A/T			
• TBD					
• PROPULSION	[illegible]	F/A/T			
• WINDTUNNEL	[illegible]	F/A/T			
• STATIC TEST ARTICLE	[illegible]	F/A/T	— SEE DETAILS BELOW —		
• STATIC TEST	07/98	F/A/T		PALMDALE	
• MOD FOR WATER FLOTATION TEST	12/98	F/A/T		PALMDALE	
• DYNAMIC TEST ARTICLE	01/98	F/A/T	— SEE DETAILS BELOW —		
• MODAL TEST	01/98	F/A/T		DOWNEY	
• MOD FOR THERMAL VACUUM TEST	[illegible]	F/A/T		SEAL BEACH	
• FLIGHT TESTS				EDWARDS AFB/KSC	
• APPROACH & LANDING TEST VEHICLE	[illegible]	F/A/T/O	— SEE DETAILS BELOW —		
• MOD B-52 CARRIER	[illegible]	F/A/T		PALMDALE	
• APPROACH & LANDING TEST	[illegible]	T		EDWARDS AFB	
• MOD FOR WATER IMPACT TEST	[illegible]	F/A/T (Boilerplate)		[illegible]	PALMDALE/KSC
• FLV-01 ORBITER FLIGHT TEST VEHICLE	[illegible]	F/A/T/O	— SEE DETAILS BELOW —		
• PERFORM 4 FLIGHT TESTS	01/01-04/01-0?/01-01/02	T		EDWARDS AFB/KSC	
PATHFINDER VEHICLE (Facilities Fit-Checks)		T	[illegible]		
OPERATIONAL VEHICLES				KSC, FL	
• PLV-02	[illegible]	F/A/T/O		KSC, FL	
• PLV-03	11/98	F/A/T/O		KSC, FL	
• PLV-04	07/98	F/A/T/O		KSC, FL	
• PLV-05	09/04	F/A/T/O		KSC, FL	
• PLV-06	11/04	F/A/T/O		KSC, FL	
• PLV-07	07/98	F/A/T/O		KSC, FL	
• GROUND TEST ARTICLE (GTA) (1)(SHARED)LANDING GEAR(2)RCS) (2)OMS)(4)PRIME POWER)(4)ECAD) (4)(SHARED)ACTUATORS) (3)GNC)(3)COM & TRACK) (3)DATA PROC)(3)VHMS)(4)D&C) (5)ECLSS)(1)REC & AUX SYS) ADAPTER - (1)PROPULSION) (1)PRIME POWER)(1)ECAD) (1)AVIONICS)(1)AUXILIARY SYS)	08/97 through 03/98 (See above)	F/A/T/O			
• STATIC TEST ARTICLE (STA) (1)WING GROUP)(1)CENTER FIN) (1)BODY GROUP)(1)PARTIAL) (STRUCTURAL ELEMENT ONLY) REC & AUX SYS) ADAPTER - (1)BODY GROUP)	[illegible] (STA COMP 04/98) (Comp MOD for Float Boilerplate Test 03/99)	F/A/T/O		PALMDALE/ JSC	
• DYNAMIC TEST ARTICLE (DTA) (1)WING GROUP)(1)CENTER FIN) (1)BODY GROUP)(1)(SHARED) LANDING GEAR)(1)(SHARED) ACTUATORS)(1)(PARTIAL) (ALL SEATS NO PROV.)(PERS ACCOMD) (1)(PARTIAL)(STRUCTURAL ELEMENT ONLY)(REC & AUX SYS) ADAPTER - (1)BODY GROUP)	[illegible] (VIBRO ACOUSTIC TEST 12/98) (MODAL TEST 01/98)	F/A/T/O (THERMAL VAC TEST [illegible])		DOWNEY/SEAL BEACH	
• APPROACH & LANDING TEST (ALT) VEHICLE (1)WING GROUP)(1)CENTER FIN) (1)BODY GROUP)(1)LANDING GEAR) (1)PRIME POWER)(1)ECAD) (1)ACTUATORS)(1)GNC) (1)COM & TRACK)(1)DATA PROC) (1)VHMS)(1)D&C)(1)ECLSS) (1)PARTIAL)(PILOTS SEATS & PROV) PERS ACCOMD)(1)REC & AUX SYS) ADAPTER - (2)BODY GROUP(FORWARD SKIRT & THRUST STRUCTURE ONLY)) (0.75)BODY GROUP(FORWARD SKIRT & THRUST STRUCTURE ONLY)) (2)PROPULSION)(2)PRIME POWER) (2)ECAD)(2)AVIONICS)(2)AUX SYS)	[illegible] Prep for WATER IMPACT Boilerplate TEST COMP 12/98	F/A/T/O	DOWNEY/ PALMDALE	EDWARDS AFB	

MOCK-UPS/SIMULATORS/ TEST ARTICLES/COMPONENTS/ VEHICLES (Con't.)	H/W-S/W COMPLETE DATE	FACILITIES REQUIREMENTS (Con't.): TYPE FACILITIES REQ'D (SEE LIST PG 1-8)	FACILITIES REQUIREMENTS (Con't.): EXISTING/ LOCATION	FACILITIES REQUIREMENTS (Con't.): MODS REQUIRED/ LOCATION	FACILITIES REQUIREMENTS (Con't.): NEW/ LOCATION
• FLIGHT TEST VEHICLE (FLV-1) (1)WING GROUP)(1)CENTER FIN) (1)BODY GROUP)(1)TPS/TCS) (1)LANDING GEAR)(1)RCS)(1)OMS) (1)PRIME POWER)(1)ECAD) (1)ACTUATORS)(1)GNC) (1)COM & TRACK)(1)DATA PROC) (1)VHMS)(1)D&C)(1)ECLSS) (1)PARTIAL)(PILOT SEATS ONLY + PILOT PROVISIONS)(PERS ACCOMD) (1)REC & AUX SYS)(1)PERS GEAR) (4)(4 FLIGHT TESTS)CONSUMABLES) ADAPTER - (4)BODY GROUP) (4)TPS)(4)PROPULSION) (4)PRIME POWER)(4)ECAD) (4)AVIONICS)(4)AUX SYS)	[illegible] (1st Launch 01/01)	F/A/T/O		EDWARDS AFB /KSC	
• WING GROUP			• (See Below)		
• WING STRUCTURES	9/98		• (See Below)		
• UPPER/LOWER SKINS		F/A	• DOWNEY/[illegible]		
• CONTROL SURFACES		F/A	• TULSA/DOWNEY		
• FRONT CARRYTHROUGH FRAME		F/A	• DOWNEY/[illegible]		
• AFT CARRYTHROUGH FRAME		F/A	• DOWNEY/[illegible]		
• TRANSFER TUNNEL		F/A	• DOWNEY		
• UPPER TRANSFER TUNNEL COVER		F/A	• TULSA/DOWNEY		
• TAIL GROUP	10/98		• (See Below)		
• CENTER FIN		F/A	• TULSA/DOWNEY		
• BODY GROUP	4/98		• (See Below)		
• CREW CABIN			• (See Below)		
• PRIMARY STRUCTURE		F/A	• DOWNEY		
• SECONDARY STRUCTURE		F/A	• DOWNEY		
• WINDOWS		F/A	• TBD		
• UPPER HATCH ASSEMBLY		F/A/T	• DOWNEY		
• AFT HATCH ASSEMBLY		F/A/T	• DOWNEY		
• KEEL STRUCTURE		F/A	• DOWNEY		
• AVIONICS RACKS		F/A	• DOWNEY		
• HEAT SHIELD		F/A	• TULSA/DOWNEY		
• PROPULSION STRUCTURES			• (See Below)		
• [illegible]		F/A	• TBD		
• AFT RCS		F/A	• [illegible]		
• FORWARD RCS		F/A	• [illegible]		
• FRAMES		F/A	• [illegible]		
• FORWARD FLYING FRAMES		F/A	• DOWNEY		
• MID FLYING FRAMES		F/A	• DOWNEY		
• AFT FLYING FRAMES		F/A	• DOWNEY		
• DOORS		F/A	• TULSA/DOWNEY		
• NOSE DOORS		F/A	• TULSA/DOWNEY		
• FORWARD DOORS		F/A	• TULSA/DOWNEY		
• MID DOORS		F/A	• TULSA/DOWNEY		
• AFT DOORS		F/A	• TULSA/DOWNEY		
• THERMAL PROTECTION SYSTEM - EXTERNAL	8/98		See Below		
• NOSE		F/A	• TBD		
• HEATSHIELD		F/A	• PALMDALE/TBD		
• WING			See Below		
• UPPER SURFACE		F/A	• PALMDALE		
• LEADING EDGE		F/A	• TBD		
• LOWER SURFACE		F/A	• PALMDALE		
• TAIL - LEADING EDGE		F/A	• TBD		
• TAIL - SIDES SURFACE		F/A	• PALMDALE		
• INTERNAL - CREW CABIN		F/A	• TBD		
• INTERNAL - TRANSFER TUNNEL		F/A	• TBD		
• INTERNAL - EQUIPMENT BAYS		F/A	• TBD		

Table 6-5. Hardware and Software Facilities Requirements (Concluded)

KEY: FABRICATION = F ASSEMBLY = A TEST = T OPERATIONS = O

MOCK-UPS/SIMULATORS/ TEST ARTICLES/COMPONENTS/ VEHICLES	H/W-S/W COMPLETE DATE	FACILITIES REQUIREMENTS: TYPE FACILITIES REQ'D	EXISTING/ LOCATION	MODS REQUIRED/ LOCATION	NEW/ LOCATION
• LANDING GEAR - NOSE	01/99		See Below		
• WHEEL		F	• TBD		
• TIRE		F	• TBD		
• MECHANICAL		F/A/T	• TBD		
• STRUCTURE		F/A	• TBD		
• LANDING GEAR - MAIN	01/99		See Below		
• WHEELS		F	• TBD		
• TIRES		F	• TBD		
• MECHANICAL		F/A/T	• TBD		
• STRUCTURES		F/A	• TBD		
• BRAKES		F/A/T	• TBD		
• ANTI SKID		F/A/T	• TBD		
• PROPULSION, RCS	[illegible]	F/A/T	• TBD		
• PROPULSION, OMS	12/99	F/A/T	• TBD		
• PRIME POWER - BATTERIES		F/A/T	• TBD		
• ELECTRIC CONVERSION & DISTRIBUTION	[illegible]		See Below		
• POWER CONTROLLER		F/A/T	• TBD		
• PYRO INITIATION CONTROLLER		F/A/T	• TBD		
• WIRE HARNESS		T	• DOWNEY		
• SUPPORTS (STRUCTURE)		F/A	• DOWNEY		
• ACTUATORS	[illegible]		See Below		
• WING ELEVONS		F/A/T	• TBD		
• UPPER BODY ELEVONS		F/A/T	• TBD		
• LOWER BODY FLAP		F/A/T	• TBD		
• WING FOLDING		F/A/T	• TBD		
• NOSE LANDING		F/A/T	• TBD		
• MAIN LANDING		F/A/T	• TBD		
• TOP HATCH COVER		F/A/T	• TBD		
• CONTROLLERS	[illegible]		See Below		
• WING ELEVONS		F/A/T	• TBD		
• UPPER BODY ELEVONS		F/A/T	• TBD		
• LOWER BODY FLAP		F/A/T	• TBD		
• WING FOLDING		F/A/T	• TBD		
• AVIONICS	10/98		See Below		
• GN&C	10/98		See Below		
• INERTIAL MEASUREMENT UNITS (IMU's)		F/A/T	• TBD		
• GPS RECEIVERS		F/A/T	• TBD		
• ALTIMETERS		F/A/T	• TBD		
• AIR DATA PROBE		F/A/T	• TBD		
• AIR DATA SYSTEM ASSEMBLY		F/A/T	• TBD		
• HORIZON SENSORS		F/A/T	• TBD		
• HORIZON SENSOR HEADS		F/A/T	• TBD		
• COMMUNICATIONS & TRACKING	10/98		See Below		
• S-BAND ANTENNAS		F/A/T	• TBD		
• S-BAND PRE-AMPS		F/A/T	• TBD		
• S-BAND POWER AMPS		F/A/T	• TBD		
• S-BAND ANTENNA SWITCHES		F/A/T	• TBD		
S-BAND TRANSPONDERS		F/A/T	• TBD		
• SARSAT ANTENNA		F/A/T	• TBD		
• SARSAT TRANSPONDER		F/A/T	• TBD		
• UHF ANTENNAS		F/A/T	• TBD		
• UHF TRANSCEIVERS		F/A/T	• TBD		
• AUDIO TERMINALS		F/A/T	• TBD		
• HEADSETS & MIKES		F/A/T	• TBD		
• DATA PROCESSING	10/98		• See Below		
• COMPUTERS		F/A/T	• TBD		
• SIGNAL CONDITIONERS		F/A/T	• TBD		
• AEROSURFACE CONTROLLERS		F/A/T	• TBD		
• REACTION JET CONTROLLERS		F/A/T	• TBD		
• VEHICLE HEALTH MONITORING	[illegible]		See Below		
• SENSORS		F/A/T	• TBD		
• TELEMETRY UNIT		F/A/T	• TBD		
• HARNESS		F/A/T	• TBD		
• DISPLAYS & CONTROLS	[illegible]		See Below		
• CLOCK		F/A/T	• TBD		
• ELECTRIC TIME BASE GENERATOR		F/A/T	• TBD		
• DISPLAY/SELECTION CONTROL		F/A/T	• TBD		
• VOICE RECORDER		F/A/T	• TBD		
• HEAD-UP DISPLAYS		F/A/T	• TBD		
• DISPLAY CONTROLLERS		F/A/T	• TBD		
• SWITCHES		F/A/T	• TBD		
• PANELS		F/A/T	• TBD		
• SIGNAL CONDITIONERS		F/A/T	• TBD		
• CONTROLS		F/A/T	• TBD		

MOCK-UPS/SIMULATORS/ TEST ARTICLES/COMPONENTS/ VEHICLES (Con't.)	H/W-S/W COMPLETE DATE	FACILITIES REQUIREMENTS (Con't.): TYPE FACILITIES REQ'D (SEE LIST PG 1/1)	EXISTING/ LOCATION	MODS REQUIRED/ LOCATION	NEW/ LOCATION
• ENVIRONMENTAL CONTROL	[illegible]		See Below		
• CABIN PRESSURE & ATMOSPHERE			See Below		
• REGULATOR VALVES		F/A/T	• TBD		
• AIR TANKS		F/A/T	• TBD		
• AUX O2 TANKS		F/A/T	• TBD		
• FAN ASSEMBLY		F/A/T	• TBD		
• CO2 ABSORBER ASSEMBLY		F/A/T	• TBD		
• RELIEF BLEED VALVES		F/A/T	• TBD		
• CONDENSING HX		F/A/T	• TBD		
• SEPARATION VALVE QD		F/A/T	• TBD		
• EQUALIZATION VALVE & DUCTING		F/A/T	• TBD		
• HEAT TRANSFER & REJECTION SYSTEM			See Below		
• THERMAL CAPACITOR		F/A/T	• TBD		
• TUBING		F/A/T	• DOWNEY		
• VARIABLE ORIFICE		F/A/T	• TBD		
• VENT ISOLATION VALVE		F/A/T	• TBD		
• FLEX LINES		F/A/T	• TBD		
• PUMPS		F/A/T	• TBD		
• SEPARATION TANK ASSEMBLY		F/A/T	• TBD		
• FLASH EVAPORATOR		F/A/T	• TBD		
• WATER TANKS		F/A/T	• TBD		
• PERSONNEL ACCOMODATIONS			See Below		
• CREW SEATS	[illegible]	F/A/T	• TBD		
• PASSENGER SEATS	[illegible]	F/A/T	• TBD		
• STOWAGE BINS	[illegible]	F/A/T	• DOWNEY		
• WASTE WATER TANK	[illegible]	F/A/T	• TBD		
• WATER TANK SYSTEMS	[illegible]	F/A/T	• TBD		
• WASTE SYSTEMS	[illegible]	F/A/T	• TBD		
• FIRE DETECTION	[illegible]	F/A/T	• TBD		
• SEAT SUPPORT STRUCTURES	[illegible]	F/A/T	• DOWNEY		
• FOOD RESTRAINTS	[illegible]	F/A/T	• DOWNEY		
• FURNISHINGS	[illegible]	F/A/T	• DOWNEY		
• RECOVERY & AUXILIARY SYSTEMS			See Below		
• SEPARATION	01/99		See Below		
• STRUCTURE		F/A/T	• DOWNEY		
• PYROTECHNIC DEVICE		F/A/T	• TBD		
• DOCKING SYSTEM	01/99	F/A/T	• DOWNEY		
• PARACHUTE SYSTEM	01/99	F/A/T	• TBD		
• LATCHES	01/99		See Below		
• NOSE DOORS		F/A/T	• DOWNEY		
• MAIN DOORS		F/A/T	• DOWNEY		
• WING FOLDS		F/A/T	• DOWNEY		
• PERSONNEL & GEAR			See Below		
• CREW & PASSENGERS			• N/A		
• PARTIAL PRESSURE SUITS		F/A/T	• TBD		
• FOOD		F/A	• N/A		
• LIOH		F/A	• TBD		
• ACCESSORIES		F/A/T	• TBD		
• SURVIVAL/RESCUE		F/A/T	• TBD		
• MEDICAL		F/A/T	• TBD		
• HYGIENE		F/A/T	• TBD		
• MISCELLANEOUS		F/A/T	• TBD		
• CONSUMABLES/FLUIDS			See Below		
• PROPELLANTS		F	• TBD		
• GASSES		F	• TBD		
• ENVIRONMENTAL CONTROL		F	• TBD		
• LANDING GEAR		F	• TBD		
• ADAPTER SYSTEM			See Below		
• BODY GROUP	01/99		See Below		
• FORWARD SKIRT STRUCTURE		F/A/T	• DOWNEY		
• THRUST STRUCTURE		F/A/T	• DOWNEY		
• AFT SKIRT STRUCTURE		F/A/T	• DOWNEY		
• THERMAL PROTECTION SYSTEM	06/99		See Below		
• EXTERNAL (SPRAY ON)			See Below		
• FORWARD SKIRT		F/A/T	• DOWNEY		
• AFT SKIRT		F/A/T	• DOWNEY		
• PROPULSION - SOLID ROCKET MOTORS	06/99	F/A/T	• TBD		
• PRIME POWER - BATTERY	09/99	F/A/T	• TBD		
• ELECTRIC CONVERSION & DISTRIBUTION	09/99		See Below		
• WIRE HARNESSES (FORWARD & AFT)		F/A/T	• DOWNEY		
• AVIONICS - CAUTION & WARNING	09/99	F/A/T	• TBD		
• AUXILIARY SYSTEMS	09/99		See Below		
• STRUCTURE		F/A/T	• DOWNEY		
• PYRO DEVICE		F/A/T	• TBD		

Table 6-6. Facilties Floor Plan Areas - ROM Estimates

DOWNEY FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
FABRICATION		
SHEET METAL	3000	
MACHINE SHOP	10000	
PROCESSING	10000	
BONDING	8000	
WELDING	2000	
WIRE HARNESS	15000	
TOOLING	15000	
Sub-Total	*63000*	
ASSEMBLY		
ASSEMBLY	60000	
Sub-Total	*60000*	
	Total	123000
TEST		
ACCEPTANCE TEST	10000	
SOFTWARE	5000	
SIMULATORS	3000	
ENVIRONMENTAL CONTROL	3000	
QUALIFICATION TEST	3000	
VIBRO ACOUSTIC TEST	4000	
MODAL TEST	3000	
DEVELOPMENT TEST	10000	
LABORATORIES	30000	
Sub-Total	*71000*	
	Total	71000
MOCK-UP		
MOCK-UPS	15000	
Sub-Total	*15000*	
	Total	15000
OTHER		
OFFICE	150000	
WAREHOUSE	20000	
MISC	40000	
Sub-Total	*210000*	
	Total	210000
	TOTAL	419000
OFFICE	150000	
LABORATORY	71000	
MANUFACTURING	123000	
WAREHOUSE	20000	
OTHER	55000	
TOTAL	419000	

LAD FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
FABRICATION		
SHEET METAL	4000	
MACHINE SHOP	20000	
PROCESSING	5000	
Sub-Total	29000	
	Total	29000
ASSEMBLY		
SUB-ASSEMBLY	2000	
Sub-Total	2000	
	Total	2000
TEST		
INSPECTION	3000	
Sub-Total	3000	
	Total	3000
OTHER		
OFFICE	6000	
WAREHOUSE	2000	
MISC	1000	
Sub-Total	9000	
	Total	9000
	TOTAL	43000
OFFICE	6000	
LABORATORY	3000	
MANUFACTURING	31000	
WAREHOUSE	2000	
OTHER	1000	
TOTAL	43000	

TULSA FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
FABRICATION		
BONDING	12000	
Sub-Total	*12000*	
	Total	12000
ASSEMBLY		
ASSEMBLY	5000	
Sub-Total	*5000*	
	Total	5000
TEST		
ACCEPTANCE TEST	2000	
Sub-Total	*2000*	
	Total	2000
OTHER		
OFFICE	6000	
WAREHOUSE	1000	
MISC	1000	
Sub-Total	*8000*	
	Total	8000
	TOTAL	27000
OFFICE	6000	
LABORATORY	2000	
MANUFACTURING	17000	
WAREHOUSE	1000	
OTHER	1000	
TOTAL	27000	

PALMDALE FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
ASSEMBLY		
ASSEMBLY	30000	
Sub-Total	*30000*	
	Total	30000
TEST		
ACCEPTANCE TEST	10000	
Sub-Total	*10000*	
	Total	10000
OTHER		
OFFICE	40000	
WAREHOUSE	5000	
MISC	10000	
Sub-Total	*55000*	
	Total	55000
	TOTAL	95000
OFFICE	40000	
LABORATORY	10000	
MANUFACTURING	30000	
WAREHOUSE	5000	
OTHER	10000	
TOTAL	95000	

JSC FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
TEST		
BOUYANCY	3000	
HI-FI SIMULATOR	2000	
STATIC	3000	
TRAINER	4000	
TEST BEDS	2000	
Sub-Total	*14000*	
	Total	14000
OPERATIONS		
FLIGHT DESIGN	3000	
CREW ACTIVITY PLANNING	4000	
CONTROL CENTER	4000	
SIMULATOR	5000	
COMPUTER TRAINER	3000	
Sub-Total	*19000*	
	Total	19000
OTHER		
MOCK-UP	3000	
OFFICE	8000	
Sub-Total	*11000*	
	Total	11000
	TOTAL	44000

WHITE SANDS FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
TEST		
PARACHUTE	5000	
ADAPTER SEPERATION	5000	
PROPULSION	10000	
ABORT	5000	
Sub-Total	*25000*	
	TOTAL	25000

EDWARDS AFB FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
FLIGHT TEST		
FLIGHT TEST	10000	
Sub-Total	*10000*	
	TOTAL	10000

KSC FACILITIES ROM AREA ESTIMATE

	AREA (SQ-FT)	AREA TOTAL
OPERATIONS		
LANDING STRIP	TBD	
HORIZONTAL PROCESSING	109000	
LAUNCH COMPLEX	TBD	
DESERVICE	2550	
ADAPTER PROCESSING	10350	
Sub-Total	*121900*	
	TOTAL	121900

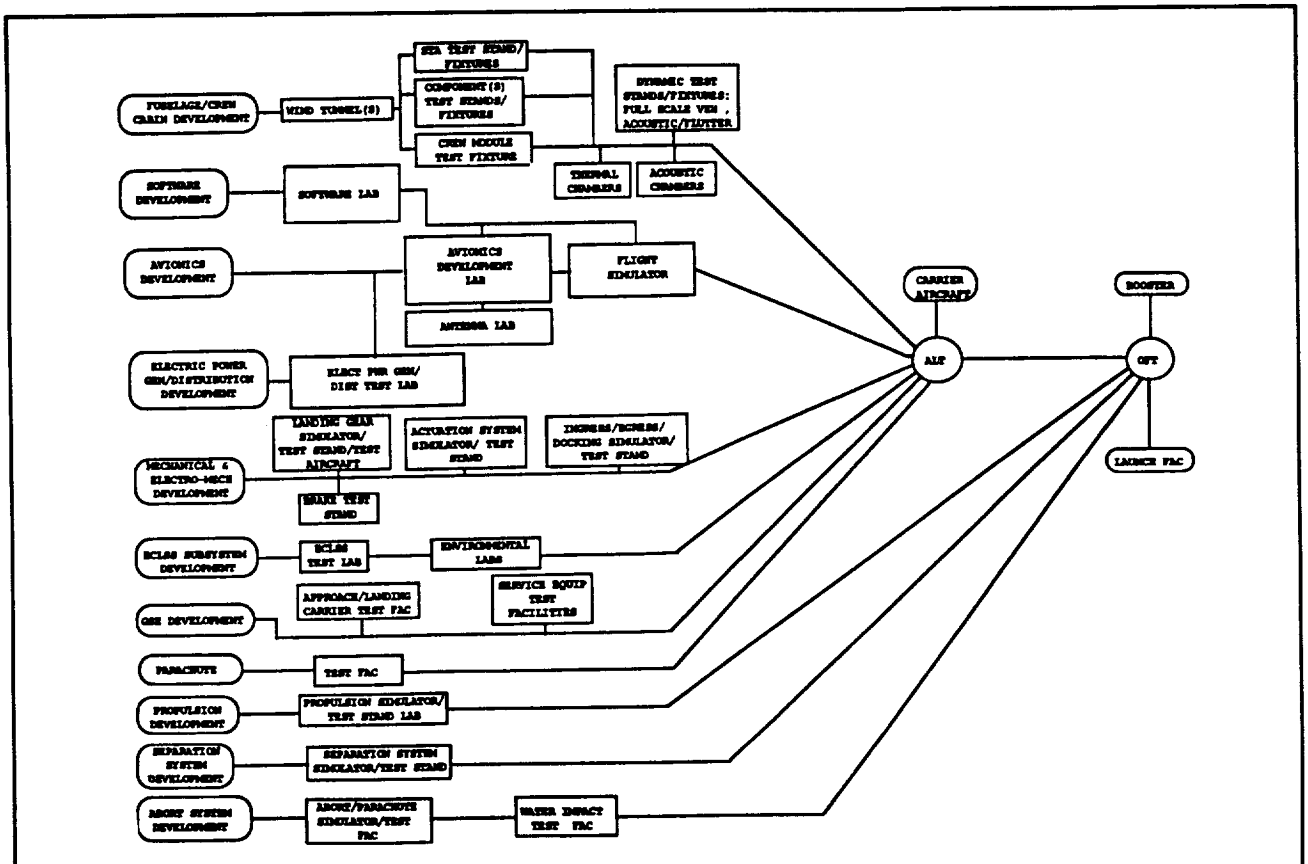


Figure 6-12. Verification/Validation Facilities Logic

Manned Spacecraft Comparison. Lessons learned from other space programs were reviewed, see Table 6-7. This review dealt with the status of major vehicle test articles and selected system capabilities. The Apollo Program dedicated test articles to the major test programs. The Shuttle ALT vehicle was eventually relegated to non-flight status. The preferred PLS intends to dedicate a modified vehicle to this test. Static load tests, if taken to the ultimate, render the article unfit for flight, and as such most static test articles are designed for test only. That was the case for Apollo CSM/LM and 1/4-scale and full-scale Shuttle models. PLS plans to use a non-flight test article.

The initial Verification/Validation testing facilities have been identified in Figure 6-12 for the PLS preferred concept. The majority of these facilities already exist, as indicated in Table 6-4, and will be updated to support PLS requirements.

The Apollo flight system provided a manual override as a backup, which is not practical for the Shuttle or PLS as neither can be fully flown manually. An independent system avoids the potential generic problems of the primary system. The first Apollo flights were unmanned and controlled by a mechanical system. The Shuttle did not incorporate unmanned capabilities, nor will the PLS.

Table 6-7. Test Article and System Capability Comparison

FLIGHT CONF TEST ARTICLE AND SYSTEM CAPABILITIES	APOLLO	SHUTTLE	PLS
APPROACH AND LANDING	YES—DEDICATED	YES—BUT NOT DEDICATED OV-101	PLANNED
WATER IMPACT	YES—BOILERPLATE	NA	PLANNED - BOILERPLATE
STATIC TEST	YES—DEDICATED	YES—BUT NOT DEDICATED OV099	PLANNED - DEDICATED TEST ARTICLE
DYNAMIC TEST	YES—DEDICATED	ONLY MAJOR SEGMENTS	PLANNED - DEDICATED TEST ARTICLE
PATHFINDER	YES—BOILERPLATE	YES—OV101	PLANNED - EXISTING NON-FLIGHT ARTICLE
MODAL TEST	YES—CSM/LM CONFIG	YES—1/4 SCALE AND SHUTTLE WITH OV101	PLANNED - EXISTING NON-FLIGHT ARTICLE
BACKUP FLIGHT SYS	MANUAL OVERRIDE	YES—INDEPENDENT	YES—INDEPENDENT
UNMANNED FLIGHT CAPABILITY FOR ALT/OFT	YES—MECHANICAL BOY (ILO PILOT)	NO	(UNDER REVIEW)

Life Cycle Development. Development test and evaluation (T&E) serves a number of useful functions. It will provide information to PLS decision makers responsible for making cost and risk decisions which impact life cycle cost and reliability over the life of the system. T&E will be conducted to demonstrate the feasibility of conceptual approaches, to minimize risk, to identify design alternatives, to compare and analyze tradeoffs and to estimate operational effectiveness and suitability. As the PLS undergoes design and development, the emphasis in testing will move gradually from development to operational T&E. The later phase will focus on questions of operational effectiveness, suitability and supportability. As noted T&E is a process that will be continuous through the development and operational phases, A,B and C/D, Figure 6-13.

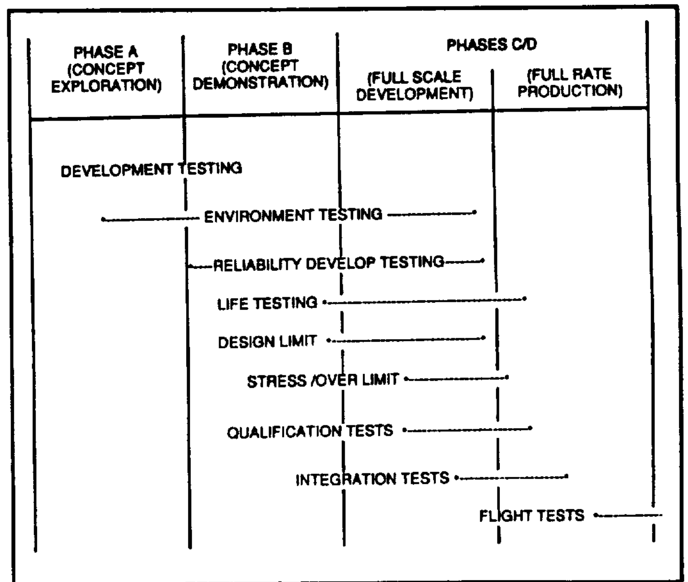


Figure 6-13. Test and Evaluation Phases

Phase A, or Conceptual Exploration, is the time frame when the T&E Master Plan (TEMP) is conceived. The TEMP is the basic planning document for all T&E functions and is the guiding manual for planning, reviewing and approving T&E programs and provides the basis and authority for all other detailed test related documents. The TEMP will identify all critical technical characteristics, operational issues and T&E schedules. The TEMP will be reviewed and updated as the program matures. Key topics contained in the TEMP are shown in Figure 6-14. In addition to development of the TEMP, development testing will begin during Phase A and continue into Phase B.

During phase B, also known as Concept Demonstration Validation, environmental testing will commence and continue into Phase C or Full Scale Development phase. Life, limit and qualification testing will be performed on prototype and production articles during Phases B and C.

Integration and flight testing will occur during the C/D phase. Testing during phase C/D will be handed off to the NASA for operational testing with Rockwell providing support.

6.1.4 Operations Planning

The first production vehicle (PLV-02) build is delayed so that the first operational launch would occur three months after the last OFT flight. A successful OFT program could result in a nine month interval between the last OFT flight and the first operational flight due to the two contingency OFT flights built into the schedule. The delay in the production of the first operational vehicle has a minor impact on the production staffing. The second shift operation will have to be reduced for about one year. These people, who have intimate knowledge of the PLS vehicle could provide a great service at Palmdale, for instance, to process the ALT vehicle and at KSC to help validate the PLS processing facilities.

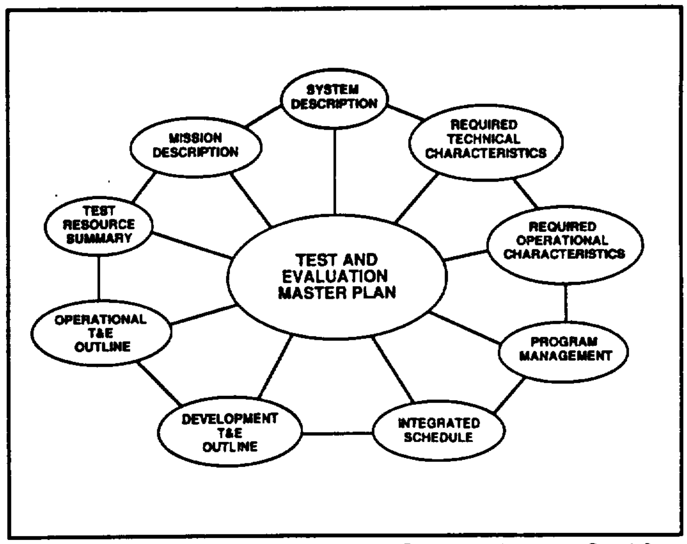


Figure 6-14. Key Elements of the Test and Evaluation Master Plan

The mission planning effort for the first DRM flight in August, 2002, is presented on Figure 6-15. Comparing this figure with the related data for the first OFT (Figure 6-6) shows a reduction of lead time from one year to eight months for the operational system. Further mission planning lead time will materialize with additional operational maturity.

The operations study results established that the optimum turn around time for the PLS program to support eight flights a year was forty three calendar days. Figure 4-14 presented this 43-day turnaround flow.

6.1.5 Safety and Reliability

The aspects of safety and reliability were addressed during the study, but no formal safety plan or hazard analysis was performed. The reliability requirements for logistics and operations were defined in Reference 6-1).

Safety. The PLS design requirements have been developed with full understanding that the number of potential hazards are influenced by the design itself. The PLS preferred concept has many design requirements characteristics that will reduce the number of hazards requiring control. For example:

H_2O_2 and JP_4 Propellant - elimination of hydrazine, which was the propellant in the reference concept.

Electromagnetic Actuators (EMA's) - Hydraulics and auxiliary power units (APU's) are not necessary to support the aerosurfaces, landing gear and brakes.

Robust Design Structure - The launch escape loads are a design driver and assures large ascent load margins.

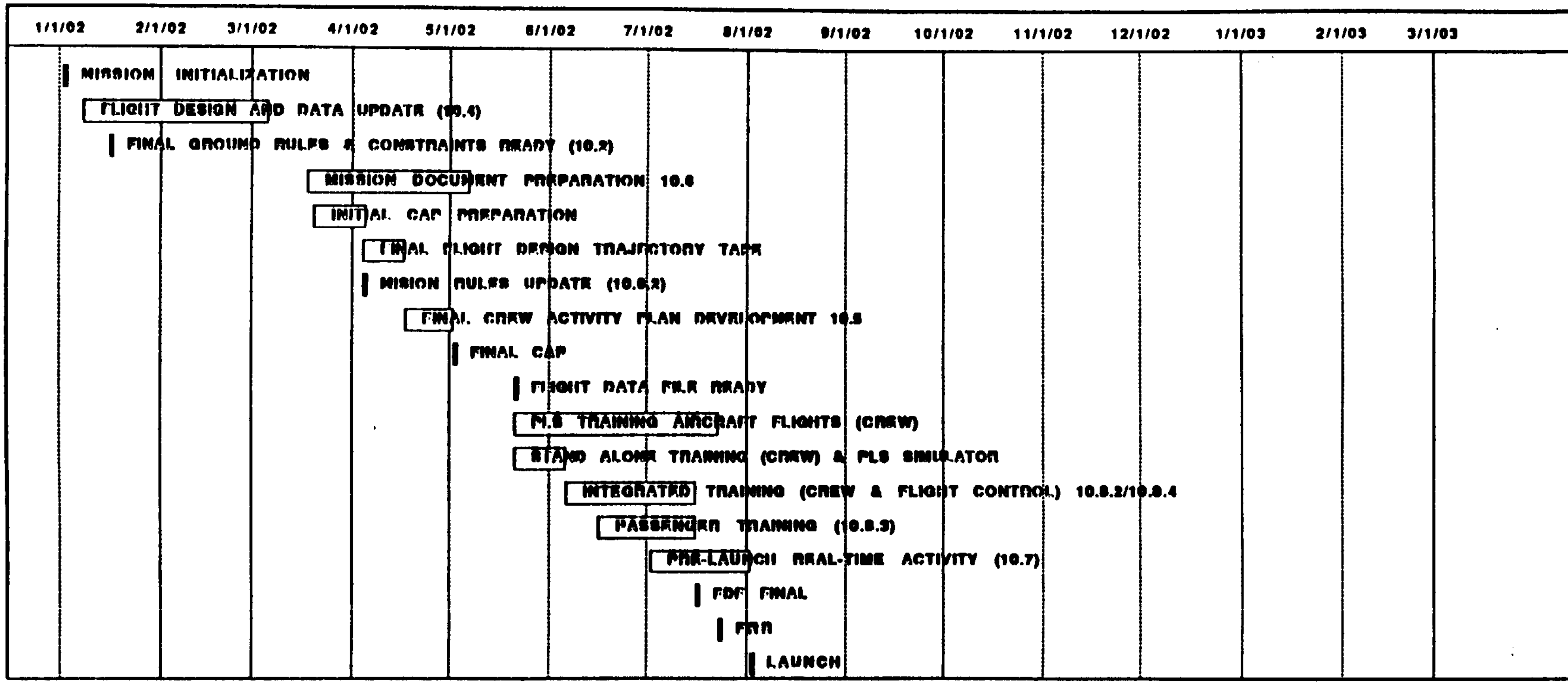


Figure 6-15. Mission Planning Preparation for DRM-1 First Flight

Landing Gear - Design to accommodate maximum weight and landing cross winds.

Avionics - Failure tolerant design to allow mission continuation after failure.

A detailed hazards analysis for the PLS will be performed during a later development phase. A number of general hazard subjects are known now and should be addressed during the initial design phase. A summary of the data for future reference is presented in Table 6-8.

The intent of any design and specifically the PLS is to reduce or control the number of hazards the final design presents to the operational phase of the program. The other two hazard levels can not be eliminated, but all initial and final design efforts should try to control them. Catastrophic hazard, like the loss of the booster due to time critical failure, or critical hazards requiring an emergency action by the crew or system must be addressed and understood and control or risk understood and approved by the program office.

Reliability. Redundancy increases system reliability, but at the cost of increased complexity in fault detection, isolation and control. A future is needed to establish a system to balance the gains in reliability vs. the impact on operations.

System reliability can be increased with good system design and not increase maintenance requirements, selection of overall system architecture and clever parts selection and reliable parts placement in critical areas.

The probability of mission success is the number that should determine redundancy levels. It can easily be shown that a system that has 3 or 4 strings or a fail operational/fail operational/fail safe (FO/FO/FS) using poor quality parts and a poor system design could have a probability of mission success less than a FS system with good parts and a clever system design.

Even when parts and architecture are optimized, redundant systems often add complexity to other systems. For example, one must incorporate more sensors, and MDM's, etc., to be able to detect a fault in a system. These extra components reduce the reliability in other systems by adding parts and complexity. In addition, the adding of the fault isolation and control facilities also includes the possibilities of errors in fault detection. In other words, false alarms could reduce overall mission success probabilities as well.

A compromise must be reached between reliability, maintainability and redundancy levels. One of the most difficult engineering decisions is what success probability is acceptable.

Table 6-8. General Hazard Areas for the PLS Vehicle

Hazardous Condition	Hazard Cause	Hazard Effect	Safety Requirement	Hazard Elimination/Control Provisions Remarks
LOSS OF HABITABLE ENVIRONMENT	INABILITY TO CLOSE/SEAL HATCHES	PRESSURE LOSS	MAINTAIN TBD MIN. CABIN PRESSURE	PROVIDE TBD MAKE-UP PRESSURIZATION TO ALLOW FOR TBD LEAKAGE
	UNKNOWN GASIEOUS CONTAMINANTS	CREW TOXICITY ASPHYXIATION	ELIMINATE TOXIC SOURCES	PROVIDE REAL TIME CREW ATMOSPHERIC CONTAMINANT MONITORING SYSTEM
	EXCESSIVE CO_2	CREW ASPHYXIATION	CO/CO_2 NOT TO EXCEED TBD	PROVIDE CO/CO_2 REMOVAL CONTROL
	LOSS OF CABIN AIR CIRCULATION COOLING	CREW DISCONFORT HEAT STROKE	TBD SFM VENTILATION WITH COMPLETE CHANGE OF AIR IN TBD MINUTES	PROVIDE FO/FS CABIN VENTILATION/AIR CIRCULATION SYSTEM
	POST LANDING TOXIC FLUID VENTING/INGESTION	CREW TOXICITY ASPHYXIATION	BREATABLE ATMOSHERE	TOGETHER WITH MONITORING CAPABILITY, PROVIDE BACK-UP OXYGEN SUPPLY
INABILITY TO/ INADVERTENT SEPARATION FROM SSF	SEPARATION DUE TO STRUCTRUAL FAILURE OF LATCHING MECHANISM	LOSS OF PLS INTO SAME ORBIT	MECHANISMS DESIGNED TO STRUCTURAL MARGIN OF 1.4	SAME
INABILITY TO INADVERTENT SEPARATION FROM SSF. (CONTINUED)	JAMMING OF INTERFACING/ RELEASE MECHANISM	UNUSABLE PLS - LOSS OF CAPABILITY	FAIL - SAFE RELEASE MECHANISM	SAME
	HUMAN ERROR	LOSS OF VEHICLE AND/OR LIFE	BUILT-IN INHIBITS IN RELEASE SEQUENCE	ENSURE THAT NO SINGLE ACTION/ COMMAND CAN JETTISON PLS, ENSURE PRESSURE INTEGRITY, COMPLETED TEST AND CHECKOUT SEQUENCE AND CREW POSITIONING PRIOR TO JETTISON.
PLS/SSF IMPACT	CONTRACT DURING DOCKING	POSSIBLE LOSS OF CAPABILITY	CRITICAL FUNCTION SAFETY ASSESSMENT	ENSURE FO/FS DOCKING TO SSF PROCEDURE
	MECHANICAL RECONTRACT AT RELEASE	DAMAGE TO PLS/SSF	ENSURE SYMMETRICAL RELEASE OF PLS	ENSURE FO/FS DOCKING TO SSF PROCEDURE
	INADVERENT RCS FIRING WHILE DOCKED	DAMAGE TO SSF AND LOSS OF PLS CONSUMABLE	CRITICAL FUNCTIONS INHIBITED	PROVIDE INHIBIT TO PREVENT PREMATURE OPERATIONS

Table 6-8. General Hazard Areas for the PLS Vehicle (Concluded)

Hazardous Condition	Hazard Cause	Hazard Effect	Safety Requirement	Hazard Elimination/Control Provisions Remarks
FIRE/EXPLOSION CONDITIONS	O_2 PARTIAL PRESSURE IN EXCESS OF 30%	IGNITION OF PLS MOUNTED DEVICES	O_2 PP NOT TO EXCEED TBD %	MONITOR PLS ATMOSPHERE TO ENSURE SAFE LEVEL, ANALYZE/ TEST MATERIAL COMPATIBILITY IN TBD % O_2 PP
	HOT SURFACES IN PLS	IGNITION SOURCE	IGNITION POINTS/ HOT SURFACES NOT TO EXCEED TBD °F	DEVELOP COMPARTMENTATION, MAXIMUM AUTO IGNITION LEVELS FOR PLS
	OVERPRESSURE OF PLS CABIN VOLUME	LOSS OF VEHICLE CREW	PLS CABIN PRESSURE NOT TO EXCEED TBD	FO/FS PRESSURE CONTROL SYSTEM
	MATERIALS FLAMMABILITY/ INCOMPATIBILTY	FIRE	MATERIAL FLAMMA-BILITY ASSESSMENT PER MATCO	FO/FS PRESSURE CONTROL SYSTEM
LOSS OF ENTRY CAPABILTY	LOSS OF PAINS	PLS THERMAL DAMAGE/ VEHICLE CREW LOSS	MAINTAIN HEAT SHIELD INTEGRITY	PROVIDE FO/FS TRANSPARENCIES
	RCS ANOMALIES	TUMBLING	FO/FS SYSTEMS FOR CRITICAL FUNCTIONS	DEGREE OF ANOMALY IMPACT COULD RANGE FROM DISPERSION OF LANDING THROUGH LOSS/BREAK UP OF VEHICLE
	UNDETECTED DAMAGE TO HEAT SHIELD	PLS THERMAL DAMAGE	MAINTAIN HEAT SHIELD INTEGRITY	PROVIDE HEAT SHIELD INSPECTION OPTION AT SSF
LOSS OF ENTRY CAPABILITY (CONTINUED)	LOSS OF ELECTRICAL POWER	LOSS OF CRITICAL CONTROL FUNCTIONS	FO/FS SYSTEMS FOR CRITICAL FUNCTIONS	SAME
	ASYMMETRICAL RETRO FIRING	TUMBLING	FO/FS SYSTEMS FOR CRITICAL FUNCTIONS	SAME

6.1.6 Quality Assurance

This subject will undergo perhaps the single most significant series of changes during the PLS program life cycle. The quality program specifications developed during the 1960's and 1970's will undergo a fundamental refocus of requirements, which incorporates the precepts of TQM initiative of the 1990's into the objectives of the quality specifications. This is not to imply that these will or should be discarded, for one significant factor to success in manned spaceflight has been that of strict and disciplined attention to details - a well-known characteristic of quality programs.

The quality programs of the 1970's/1980's were also generally not addressed with any significance until the Phase C/D arrived, the premise being that emphasis on compliance, controls, procedures, and verification did not occur until then. The PLS quality program will have its formal beginning during Phase A. It will be a dedicated emphasis. It is particularly important because of the transitional nature of TQM expansion across industry-/government during the same time frame as the PLS program. Elements of this TQM emphasis are already beginning to be reflected in this DRD. Operations/Maintenance emphasis is, we believe, a strong "Customer Want" for improving turnaround efficiency and lowering the life cycle cost. The MFBP concept is the very beginning of development of detailed process flows and process capability assessments, providing efficient blending with the Government's IQue oversight initiative.

Phases A and B. During Phase A, it is extremely important that the top-level quality functional deployment (QFD) matrices be addressed. The basic premise of "Total Quality" starts with true understanding of "what the Customer really wants" and approaches to help provide it to him. Since approaches can impact basic architectures (e.g. the maintainability approach), they need to be addressed as early as possible to avoid costly engineering changes later in the program. This is a characteristic of our Japanese competitors. They "drive out" changes before commitment to manufacture.

Two of the highest correlation "wants" (QFD terminology) to implement, for example, have been those of: (1) systems simplicity and (2) that of defining the operational fault tolerance needed and the redundancy management schemes to support it. This correlation is increased when moderate-to-extreme weight/volume and resultant performance sensitivity exists from the very beginning. One example, performance margins which can tolerate a major failure condition right after launch commitment. The impact of this example to the Phase C/D quality program to be implemented is profound! Understanding of processes, their variability and reduction of their variability are main themes. As fault tolerance declines, permissible variation rapidly declines. A program intolerant to variation can emerge. Conversely, with a fault tolerant design approach, variability reduction can be safely implemented (as well as better understanding obtained over those areas still remaining which remain intolerant).

Phase B should then form the next set of quality program foundations. This phase must see convergence of a number of efforts, (for example: QFD/sub-tier matrices, development convergence with MFBP, design system organization with MFBP (part, sub-assembly, assembly number "trees", critical process definition and flow down into process requirements/capabilities trades (starting here also to bring the knowledge of critical subcontractors onto the team). The initial formation of formal simultaneous engineering teams takes place. The quality program for Phase C/D needs to be specifically and fully planned here, also particularly the specific MIL-specification/TQM transition timing with PLS as discussed earlier. Advanced technology integration, in the process of historical hard interface control tooling vs. electronic interface control, and related process control parameters, needs to be specifically defined.

As the MFBP continues development, definition of process parameters also will continue. This includes an objective of measurements made of products conformance in as real-time as possible (such as, weld ultrasonic head mounted right behind weld head). Use of statistical process control also needs to be structurally organized during this phase, so that SPC is not just a "randomly-applied" tool.

<u>Phase C/D</u>. The quality program needs to focus on as many of the "fundamentals", now integrated into TQM initiatives. The focus is initially on prevention of defects, development of process flows and capabilities determinations, application/definition of "work teams" (cross-function), training of teams, readiness to perform tasks and increased individual involvement in doing the right things right, the first time. These "work teams" are expansions of the "design teams" from the prior phases.

The quality program must here maintain fundamental assurances of stability and control to the Customer. Therefore, calibration validity of measurements, engineering change control (which should be minimized with the up-front Phases A and B emphasis), nonconforming parts/materials control, test and records integrity, etc. The objective of achieving program success, by providing outstanding value to the Customer with outstanding first-time thru quality at reasonable cost and a dependable schedule, is fully definitized and implemented during this phase.

The quality program requirements specification [MIL-Q-9858 A] tailoring activity was deferred until hardware procurement in Phase B and/or C/D. Tailoring of the specification will be an on-going process as part of refining the acquisition plan.

6.2 MANAGEMENT AND CONTROL

The purpose of this section is to present an overall management and control approach to the PLS Acquisition Phases, A, B and C/D. This section identifies and explains the key aspects of the approach and explains how the PLS Team should be organized for the total program.

6.2.1 Management Team

A key feature of our approach is the organizational structure, Figure 6-16, and its mode of operation. Engineering, with its design responsibility, is balanced by Operations and Support and Manufacturing producibility responsibilities. In an overview roll are the functions of Technology Development, Risk Management, Systems Safety, and Cost Control. These functions, together with the support organizations shown, will operate as a consolidated team for effective communications. An integrated system review board (ISRB) will be the forum for all system and trade study reviews and requirements baselining. This provides board members a voice in all issues and then requires the chairman (the Program Manager) to identify direction for expeditious resolution. All functions participate in the ISRB and have direct access to the program manager relative to any concerns in their areas at any time. The ISRB actions are formally documented. It is through this formal documentation that the PLS program would be directed, tracked, and controlled. The program manager has final signature authority for all board actions.

New technologies that provide cost effective payoff and reasonable risk will be identified and planned for utilization. Labor savings devices such as expert systems, CAD/CAM, robotics, automatic checkout, and paperless record systems will be assessed for ways to achieve cost minimization. Monitoring the effectiveness of our low-cost-driven efficient operations approach will be a key feature of our continuing program risk assessment activity. Risk management is the responsibility of System Engineering but will receive top management attention.

To ensure that the program focuses on life cycle cost, specific assignments will be given to the cost project and risk management functions. The Cost Project manager will assess design-level trades and analyses and develop definitive costs as the design matures to ensure low-cost optimization. The Risk Assessment manager will review cost allocations and study results at the programmatic level to assist the program director in meeting the total system's cost goals. Since the industrial infrastructure model is important to the early part of the acquisition phase, and will aid in leading to low-cost approaches, Project Engineering is given the assignment for its completion as a primary responsibility.

Each management position is held by a technically qualified expert. They attend weekly program reviews conducted by the program manager. These individuals are charged with maintaining close coordination with his customer counterpart to ensure that program objectives are being met.

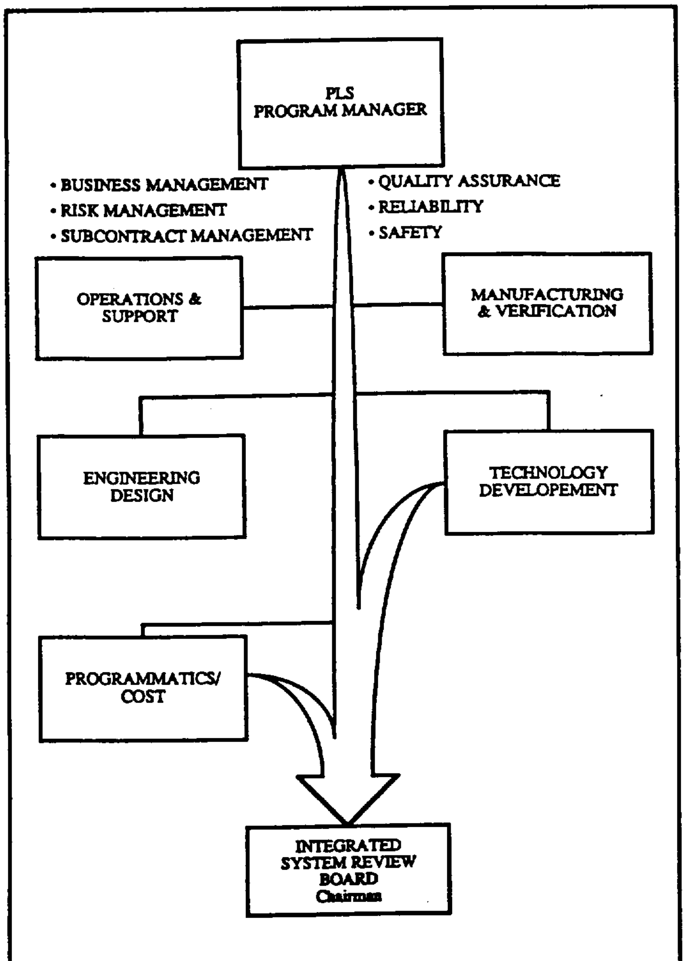


Figure 6-16. PLS Program - A Close-Coupled and Effective Organization

To achieve the desired cost an optimum reliability goals in an integrated end-to-end operations, manufacture and design concept, key subcontractors have been selected for their expertise to ensure that techniques not common to the aerospace industry are considered. NASA's extensive supplier network will be solicited for new ideas that can contribute significantly to the primary goals of the PLS.

6.2.2 Program Control

Management information and control systems will provide visibility and maintain cost and schedule control for all aspects of the PLS Acquisition Phases. Table 6-9 lists the major systems and tools to be used to manage the PLS contract along with the responsible functional organization and their application. Each system's primary use has been identified to show how it provides the program manger with necessary visibility in order to control cost, schedule, and technical performance.

Upon contract award, for each phase, functional budgets will be issued for the contact period. The Program Manager provides master change record authorization, which defines the total detail contract baseline and allocates contract budgets to the various organizations. Program changes requiring budget reallocation are accomplished by the ISRB/MCR process.

A performance management system provides cost and schedule visibility to the program manager. It includes closed-loop planning; work authorization (MCR); budget and schedule analysis, reporting and visibility and baseline change control associated with specific work change control. Figure 6-17 shows the internal process of one proven performance management system.

The performance control system must be an automated system for collection and processing of all cost, budget and schedule information associated with specific work authorizations. The system is utilized for maintaining internal program management

Table 6-9. Proven Management Tools are Required for Program Control

Functional Organization Responsiblity	Management Tool and Systems	Performance Applicability			Data Frequency*
		Cost	Scheduling	Technical	
Program Management	Work breakdown structure	X	X	X	A/R
	Responsibility assignment matrix	X	X	X	A/R
	Program milestones schedule		X		M
	Detailed Task Plan	X	X	X	A/R
	General Order	X			A/R
	Performance Control System	X	X		W
	Study Plan	X	X	X	A/R
	Forecasting funding	X	X		Q
	Contract budget allocation plan	X			A/R
	Planning and scheduling		X		W
	Technical control management			X	W
Subcontracts/ Material	Purchase Order	X	X	X	A/R
	Change Notices	X	X	X	A/R
	Interdivisional orders	X	X	X	A/R
	Technical direction notices			X	A/R
All functions	Functional schedules		X		W
	Cost account plans	X	X		A/R
	Estimates at completion	X	X	X	Q
	Variances assessment report	X			M

* Legend: W - Weekly; M - Monthly; Q - Quarterly; A/R - As required

control. PCS reports are automatically generated to provide timely expenditure and schedule data required for effective monitoring and control. This system would have the features and operational procedures of a validated cost/schedule control system criteria system such as used on GPS and B-1 programs.

At the weekly review's, conducted by the program manager, summary cost and schedule data will be reported by Business Management, to ensure that technical, cost and schedule baselines are synchronized.

The work breakdown structures (WBS's) serve as the baseline documents for program control and are used to guide development of the program schedule. They also support integration of all task assignments and are the framework for the program manpower planning and cost estimates. Management will use the WBS's for program control accounting, visibility, and reporting. The Program Master Schedule (Figure 6-1) indicates milestones reflecting the major contractual and program milestones for management planning and control. A Responsibility Assignment Matrix (RAM), developed early in the Acquisition Phase, will relate the program tasks defined by the WBS to the responsible individuals. In addition, a time phased man-loading summary will be developed that contains a schedule of man-hours per contract WBS subtask per month. The areas of emphasis and peak activity will be displayed, supporting the program objectives.

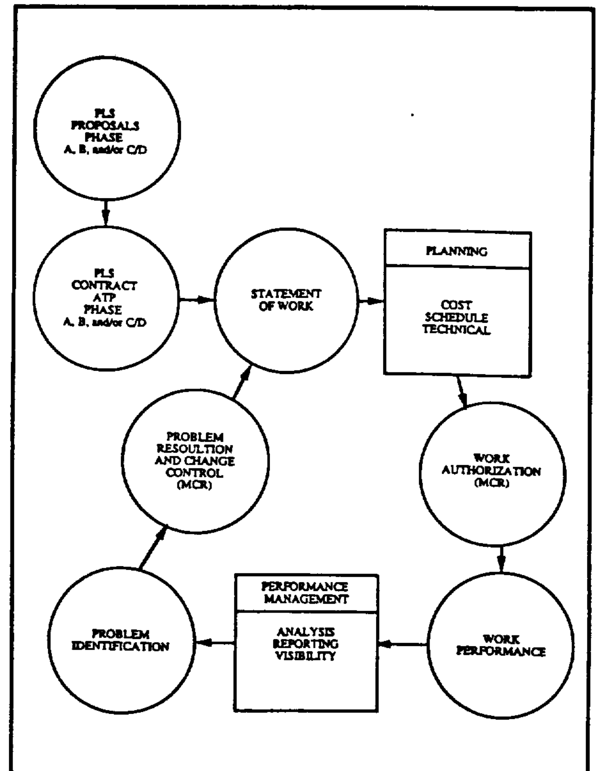


Figure 6-17. Closed-Loop Performance Management System

An automated responsibility assignment list will provide specific visibility into contract data requirement schedules and status. Monthly forecasts of data submittals will be disseminated to all concerned and monitored by the business manager for the program manager. Data Management will provide single point control for the administration, accountability, and coordination of all contract data requirements lists (CDRL's).

The CDRL/DID reference documents and related contract provisions will be analyzed by Data Management personnel experienced in implementing NASA, U.S. Air Force and DoD Requirements. Detailed responsibilities for preparation, inputs, review, approval, and support will be assigned.

Data Management personnel serve as a point of contact for customer functions regarding distribution statements, distribution changes, CDRL data item description (DID) and reference document changes, submittal schedules, submittal/approval status, and detailed records. Distribution of transmittal letters and data product copies will be as specified in the CDRL, other contract provisions, and PCO direction.

6.3 TRADE STUDIES

Two major manufacturing trade studies: (1) Fabrication of the heat shield back face structure, (2) Test program fidelity trade, and one supporting analysis: Manufacturing facilities/resources, have been performed. These studies are fully documented in Reference 5-1.

6.3.1 Single Piece vs. Multiple Section Fabrication of Heat Shield Back-Face Structure (T-5) Trade Study

The use of advanced composite materials offers the potential for greater performance, lower overall costs, and improved design flexibility over more traditional metallic structures. In the case of the PLS heat shield back-face structure, the use of high temperature graphite/polyimide advanced composite materials offers the potential for improved structural and thermal performance over traditional designs such as aluminum sheet metal skin and stringer. In order to realize the inherent performance advantages of the graphite/polyimide, however, numerous aspects concerning structural design, fabrication methods, and repairability of the structure must be addressed.

The principle objectives of this trade study were to address the structural design, fabrication methods, and the manufacturing and operational repair approaches for the PLS heat shield back-face structure. The heat shield back-face structure consisted of a one piece sandwich design comprised of graphite/polyimide facesheets and polyimide honeycomb core. This task identified the factors involved in producing a successful part and included a compilation of PLS composite heat shield considerations which would have influence over whether a strict adherence to the reference configuration is to be used or an alternate or hybrid configuration would be preferred. The concept selected as the preferred back-face heat shield structure, Figure 6-18, consists of a two piece structure, sandwich design, consisting of graphite/polyimide facesheets, honeycomb core, integral stiffeners and hardpoints.

6.3.2 Test Program Fidelity Trade (T-15) Trade Study

The goal of the test program fidelity trade was to establish concepts for the PLS test program that are capable of achieving low life cycle costs. This study addressed this subject and the concepts and criteria that surfaced during the investigation with potential attributes of attributing to low life cycle costs through testing or related functions. These results are summarized below. Details of this study may be found by referencing the test program analyses data documented in Reference 5-1.

- 40% of the orbiter anomalies were attributed to design flaws that escaped the test program.

- Design flaw escapes were attributed to lack of realistic environmental and/or stress/limit testing.

- A high fidelity PLS test program is achievable without perturbation of normal costs/schedules by subjecting products to selected tests aimed at early flaw identification.

- Mature 1992 state-of-the-art components/producers will bring to the program a level of reliability never realized in manned space travel before

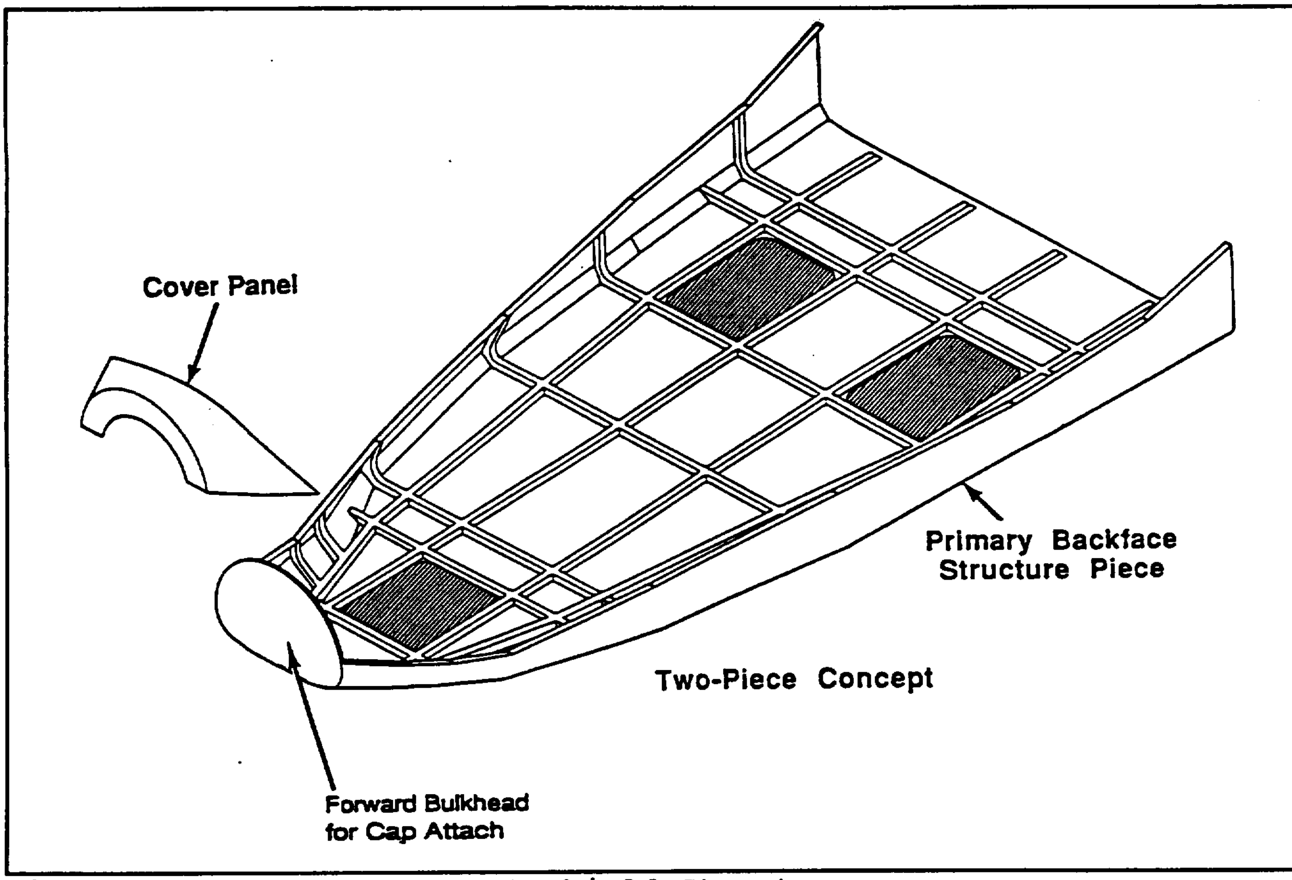


Figure 6-18. Backface Heat Shield Structure

- The PLS test program, to be effective, must include realistic (dirt/debris laden) environmental testing.

- PLS design should be driven by operations verification requirements, robustness, specification constraints, performance, over/under design and the capability for rapid product removal/replacement.

- Time allotted for product removal/replacement should be specified and verified.

- Complete/minimize margin assessments early in the flight program.

- Implement system engineering department/TQM to avoid/minimize cost growth and schedule delays.

6.3.3 Manufacturing Facilities/Resources (S-7) Supporting Analysis

The goals of minimizing the facilities costs for fabrication, assembly, installations and testing of the PLS will best be achieved by utilizing existing available facilities. This would reduce the requirement for new facilities, thus reducing costs. The detail type and size of facilities required will be determined by the technology requirements, schedules, and transportation requirements. Initial fabrication requirements have been identified and existing facilities meet these requirements with modifications. Facilities required for sub-assemblies, assemblies, systems installations and test have been identified and sites available to perform these functions have been identified, including new facilities required. Existing government facilities would also be used. Reference facilities requirements support analyses are documented in Reference 5-1 and identified in the facilities Tables 6-4, 6-5 and 6-6.

7.0 TECHNOLOGY DEVELOPMENT PLAN

This section documents the technological sufficiency of most system and subsystem choices in a review organized by the Work Breakdown Structure's (WBS) Configuration Axis; this organization corresponds most closely with the traditional hardware-oriented Work Breakdown Structure. The Technology Development Plans for two topics needing investment to provide accelerated development are presented in this section. These are the solid hydrogen storage method for the fuel cell alternative and the Booster Warning System.

When missions other than the reference DRM-1 are considered, most of the new mission requirements can also be met with current technology or with very minor development effort. The major exception to this assertion of technological adequacy lies in the electrical energy supply (power supply) for the PLS. The electrical energy supply needed for the PLS to satisfy DRM-1 is about 168 kwh and is supplied by batteries. These are silver-zinc (Ag-Zn) batteries with a limited recharge capability--about 20 charge-discharge cycles. Rechargeable silver-zinc batteries have been selected over one-shot Lithium Thionyl Chloride batteries because the reusable silver-zinc batteries are projected to have a much lower cost.

The Booster for the PLS will require a Warning System (BWS) to alert the crew to impending catastrophic events and to activate the Launch Escape System (LES) rockets. Much of the background knowledge in sensors and signatures to detect deteriorating conditions has already been acquired, but a complete system architecture is not yet available.

Finally, relatively small investments in the areas of hydrogen storage/regeneration and warning systems should be made to provide confidence that these collateral technologies can be available to the PLS program when needed.

7.1 ORGANIZATION

The PLS uses a 3-dimensional Work Breakdown Structure (WBS); the three axes in this approach are:

1) The Configuration Axis (1.X) describes the systems and subsystems including many of the hardware systems. It corresponds most closely to previous hardware oriented work breakdown structures.

2) The Functional Axis (2.X) describes the program functions from technology development to program phase out. Second level of the Functional Axis contained the following elements.

3) The Cost Axis (3.X) describes the categories of purchases.

Since most technology concerns are directed toward hardware systems, the order of presentation is modified to highlight these concerns. Accordingly, the review begins with the hardware oriented Flight Configuration System (WBS 1.6) and then considers the support elements, WBS 1.1 through 1.5.

7.2 REVIEW OF TECHNOLOGY SELECTIONS AND OPTIONS

Technology selections for the Personnel Launch System (PLS) and its Design Reference Mission-1 (DRM-1) are conservative and require only design development efforts as contrasted to technology development efforts. All design selections meet or exceed the PLS goal of achieving technology status of NASA Level 6 (prototype tested in a relevant environment) by 1992. This goal is an upward revision of the original goal of NASA Technology Level 5 (brassboard tested in a relevant environment) by 1992. Also, the technology selections meet other PLS objectives and groundrules in the areas of manned safety and minimum life cycle costs through designs and operating plans for maintainability, reliability and simplicity.

Technology alternatives for the PLS are summarized in Table 7-1. Reasoning behind the choices is presented in the text of this section. For most systems, existing technologies are adequate and the more advanced capabilities of technologies requiring development are not needed to achieve acceptable performance. Also, development costs and/or the costs of the new materials would not be recovered in reduced operating expenditures.

Manufacturing technologies supporting all of the selected systems have at most investment issues, and not technology development issues. Computer-Assisted Design/Computer Assisted Manufacturing (CAD/CAM) is the state of the art, and any issues in this area are investment issues and not technology development issues. Given the projected low numbers of PLSs which are to be built, more advanced topics such as Computer Integrated Manufacturing have questionable relevance--the production rate does not support a major investment in advanced tooling.

Table 7-1. Technology Alternatives

WBS Element & Alternative	Technology Issues & Status
Wing Group - Structure	
Composites - *Graphite/Polyimide [Gr/Pi has highest temperature capability of well characterized materials]	Manufacturing Experience Limited. Early quality problems under control. Structural Adhesive has composites been reformulated to eliminate asbestos. TPS is required.
High Temperature Metal Alloy	Heavy - Limited Space Experience. TPS may be required for multiple reuse.
Tail Group - Structure	
*Graphite/Polyimide	High temperature capability desired.
Aluminum	Aluminum alloys are marginal, Gr/Pi selected for other uses.
Body Group	
Basic Thrust Structure	
*Aluminum alloy 2024	Substantial mfg experience - lowest cost.
Lithium-Aluminum alloy	Li-Al: Limited experience; Li-Al &
Graphite / Polyimide	Gr/Pi: Lower weight & higher costs are not needed or justified.
Titanium Aluminide	Developmental & very expensive.
Primary Structure	
Crew Cabin	
Access Tunnel - Aluminum 2219	Substantial mfg experience - lowest cost.
Upper Access Panels	
*Aluminum alloy honeycomb	Al appears thermally adequate. TPS rq'd.
LiAL, TiAL	Not needed and expensive.
Graphite/Polyimide	Alternative if Al is not adequate.
Heatshield Structure	
*Graphite/Polyimide	Using Al would require too thick TPS. Gr/Pi structural repair method, structural adhesive both require certification.
Thermoplastics	Not as mature as Gr/Pi.
Titanium Aluminide	Not needed, too expensive.

* = Selected for PLS

Table 7-1. Technology Alternatives (Continued)

WBS Element & Alternative	Technology Issues & Status
Thermal Protection System (TPS)	
Body Protection	
Bottom	
*HTP-6 (FRCI) Tile (Direct Bond)	Existing Orbiter tiles suffice.
Stratified density tile " "	Lower maturity, costs similar to FRCI.
Carbon/Carbon Tiles " "	Lower maturity in application.
Carbon/Silicon Carbide " "	Much lower maturity.
Upper Surface	
*FRSI Blanket (Direct Bond)	Existing Orbiter blanket suffices.
Other blankets, tiles available as alternatives	
Leading Edge Protection	
*Carbon/Carbon (ACC) (Fasteners)	Upgraded Orbiter technology suffices.
Carbon/Silicon Carbide (C/SiC) (Fasteners)	Being developed for Hermes - a high probability candidate for later selection.
Attachment Methods	
Metallic fasteners for leading edge	No technology issues
*Direct bond adhesive(s)	Improved adhesive(s) required and available; will need certification.
Landing Gear (Nose Gear, Main Gear)	
*All Electric	Adapting fighter gear is expected to be design challenge only
Hydraulic	Desire avoiding hydraulics
Propulsion - Reaction Control System	
*Hydrogen Peroxide (Mono)	Low toxicity, technology ready.
Monopropellant Hydrazine	Toxicity complicates ground ops.
Bipropellants	Add'l performance not needed, complex system, toxic.
Cryogenic Bipropellants	Cost, technology and volume issues
Propulsion - Orbit Maneuvering System	
*Hydrogen Peroxide/JP-4	Low toxicity, little experience compared to NTO/MMH.
Nitrogen Tetroxide/ Monomethyl Hydrazine	Toxicity complicates ground ops, large experience base.
Monopropellant Hydrazine	Toxic, heavy, bulky, little experience at PLS thrust level.
Cryogenic Bipropellants	Cost, technology and volume issues.

* = Selected For PLS

Table 7-1. Technology Alternatives (Continued)

WBS Element & Alternative	Technology Issues & Status
Prime Power	All power sources have safety issues.
One-time Batteries (Lithium Thionylchloride)	Low weight, very high cost.
*Rechargeable Batteries (Ag-Zn)	High Weight,`20 cycles, mod cost.
Fuel Cell (Modified Orbiter)	To o complex, costly for DRM-1.
Low Weight Rechargeables	Development with time risk.
Low Weight Fuel Cells	Development with time risk.
Electric Conversion & Distribution	
*DC	No technology issues.
AC	No requirements identified.
Mixed	If needed, converted in/near use.
Actuators	
*Electromechanical	Design challenges, but no technology issues.
*Electrohydraulic	
Hydraulic	Desire to avoid complex generation and distribution system
Avionics	
GN&C - *Autonomous with Pilot Backup	Architectural & design issues, but not technology issues.
Communications & Tracking	Adapt SOA designs.
Data Processing	Adapt SOA designs.
Vehicle Health Monitoring - *BITE	Arch & design issues only.
Displays & Controls - * HUD	Adapt SOA designs.
Software: High Order Languages- *ADA	Adapt SOA designs.
Antennas * Under TPS	EM transparency questionable.
Antennas Deployable	Deployable is backup design.
Cooling * Passive	May be marginal.
Cooling Active	By environmental control system, if needed.
Environmental Control	
* 1 Loop (Water)(Redundant)	No technology issues.
2 Loops	
Personal Accommodations	No technology issues.
* Apollo-type waste management	
* New seats	
Recovery and Auxiliary Systems	No technology issues
*Solid rocket motors for abort propulsion	
* Parachutes	

* = Selected for PLS

Table 7-1. Technology Alternatives (Concluded)

WBS Element & Alternative	Technology Issues & Status
Operations -	
Fleet (Ground) Processing - A large number of interdependent topics including: * Automated checkout systems * Auto ground processing expert systems * Auto logistics planning expert * Improved weather protection on ground	At or very near the State-of-the-Art. Architectural and design issues, but not technology development issues.
Mission (Flight) Processing A large number of interdependent topics including: * Automated mission control expert systems * Auto launch control expert systems * Advanced lightning protection on ground and in flight.	At or very near the SOA. Architectural and design issues, but not technology development issues.

* = Selected for PLS

7.2.1 Designs and Technologies Selected for the PLS Spacecraft.

Those designs and technologies selected for the PLS are described with the presentation organized by the Configuration Axis of the WBS for the Flight Configuration System and related hardware (WBS 1.6) and specifically the Manned Spacecraft (WBS 1.6.7):

Wing Group (WBS 1.6.7.1)	
Exposed Wing	Graphite/Polyimide or High Temperature Metal Alloy
Carry Through	Graphite/Polyimide (Gr/Pi)
Tail Group (WBS 1.6.7.2)	Graphite/Polyimide
Body Group (WBS 1.6.7.3)	
Basic Structure	
Thrust Structure	Aluminum Alloy 2024
Secondary Structure	
Crew Cabin	Aluminum Alloy 2219
Access Tunnel	Aluminum Alloy 2219
Tunnel Fairing	Aluminum Alloy 2219
Upper Access Panels	Aluminum Alloy or Gr/Pi
Heat Shield Base	Graphite/Polyimide (Honeycomb)

All these materials have acceptable technology status; only the Graphite/Polyimide (Gr/Pi) composite material has not been used extensively in space.

Quality control problems with the manufacture of Graphite/Polyimide materials and production of parts have been resolved as the technology has matured. The only remaining challenge with the Graphite/Polyimide material in these applications is the development and qualification of field repair techniques for the structures.

The thermal adequacy of the Graphite/Polyimide (covered by TPS materials) for the wing and of the Al 2219 for the upper access panels is based on preliminary analyses and must be confirmed by later detailed analysis.

An additional option in structural materials is Titanium Aluminide. TiAl is being developed for the National Aerospace Plane (NASP) program. The additional thermal/structural capabilities (and higher costs) are not believed to be needed for the PLS.

Thermal Protection System (WBS 1.6.7.4)

The Thermal Protection System (TPS) consists of a variety of materials and designs to keep internal PLS temperatures at safe (and comfortable) levels during reentry.

The Thermal Protection Systems selections all have acceptable development status. There are also a substantial number of alternatives with higher capabilities with acceptable developmental status, to provide backup in case initial choices are inadequate.

The TPS consists of the following major components:

- Leading Edge protection
- Body - high temperature protection
  - moderate temperature protection
- Attachment Methods - Adhesives
  - Fasteners
- Insulation
- Seals

The selected leading edge protection method is Advanced Carbon fiber/Carbon matrix (ACC), a well studied derivative of the Reinforced Carbon/Carbon (RCC) material used on the NSTS Orbiter. A prominent option is Carbon fiber/Silicon Carbide (C/SiC) matrix composite fabrications similar to those being developed for Hermes. C/SiC fabrications have higher strength at service temperature than do the Carbon/Carbon fabrications used on the Shuttle. While full details are not yet available, it appears that C/SiC does not have as stringent requirements for anti-oxidation coatings as does ACC. It is expected that the ACC and/or C/SiC fabrications will be attached to with metallic fasteners because the fabrication's back surface temperatures will probably be higher than can be tolerated by direct-bond adhesives.

The body TPS is divided into two major components - the bottom heat shield and the upper (lee) side surfaces. The bottom (high temperature) heatshield structure will be covered with HTP-6 (FRCI) tile attached by direct-bond adhesive. The thickness of the tile will be chosen to provide acceptable back surface temperature for the adhesive. It is expected that the relatively rigid heatshield will permit large tiles to be employed so that the need to control surface smoothness with gap-fillers can be mitigated. The upper surfaces of will be exposed to considerably lower temperatures than the bottom and an adhesive-attached blanket is adequate; FRSI material has been selected.

Alternative materials exist for most of these choices:

- A stratified density tile developed by NASA-Ames Research Center is a good alternative for the STS HTP-6 tile.

- Carbon/Carbon tiles directly bonded to the lower heat-shield structure are considered to be an alternative to HTP-6, but this has not been investigated thoroughly.

- A thicker layer of the STS FRSI blanket appears to be an appropriate response to providing any needed higher temperature capability for the PLS topside. Most of the other options can also be applied to the top.

- The wings use a Graphite/Polyimide structure with an ACC leading edge and HTP-6 tiles/FRSI blankets on the cooler surfaces. An alternative is to use a high temperature metal alloy for both structure and thermal protection. ACC or C/SiC would still be required for the leading edge. In the event that additional protection against oxidation were needed for the metallic surfaces, Nextel 440 fabric could be attached to the metal with a ceramic adhesive. Nextel is a modified quartz (SiO_2) fiber which can be woven into a moderately flexible fabric.

Potential attachment methods include:

Fasteners: Fasteners for the C/C or C/SiC leading edge fabrications are expected to be made from metal alloys and represent a design challenge but have no technology/-developmental problems.

Adhesives: The principal adhesive used to attach the Tiles to the SIP and the SIP to the Orbiter is RTV-560. RTV-560 (Room Temperature Vulcanizing-560) has a thermal limit of about 550 F. Since the Graphite/Polyimide TPS support structure has a thermal capability of about 600 F, and it is desired to design the PLS to use this capability for a safety margin, RTV-560 is barely adequate for this application. Fortunately a modified RTV has been investigated and found to perform satisfactorily at 625 F. Direct Bond Adhesive RA59P consists of 3 parts RTV-560 plus 0.5% of the usual catalyst Dibutyl Tin Dilaurate mixed with 1 part GR908 plus 0.2% of another catalyst Piperidine.

To manage the heat flux during reentry, the compartments between the heat shield and the crew compartment will require an internal insulation. Fiberglass insulation sewn into panels is used on the STS Orbiter and is adequate here.

A variety of reusable seals will be needed to prevent the extremely hot gas/plasma from infiltrating compartments during reentry via the removable access panels. Specific designs will be needed for the windshields, access compartment doors, landing gear doors, and the gaps made by the aerodynamic control surfaces. The National Aerospace Plane (NASP) also has this problem and has extensively investigated reusable seal designs. Reusable seals are believed to represent a significant design and validation challenge, but a variety of materials and designs are available. Accordingly, seals are not viewed as presenting a technology problem.

Landing Gear (Nose Gear, Main Gear) (WBS 1.6.7.5)

Adaptations of existing fighter aircraft designs have been selected. These represent existing technology. The elimination of hydraulics by using electrical motors for all functions is expected to present a moderate design challenge. See the discussion under Actuators (WBS 1.6.7.12)

Propulsion - Reaction Control System (RCS) (WBS 1.6.7.7)
Propulsion - Orbit Maneuvering System (OMS) (WBS 1.6.7.8)

The propellant combination of Jet Propellant-4 (JP4) and Hydrogen Peroxide (H2/O2) has been selected for the spacecraft Orbit Maneuvering System (OMS) with H2O2 serving as a monopropellant for the Reaction Control System (RCS).

Other propellant combinations considered were:

- Monomethyl Hydrazine (MMH) and Nitrogen Tetroxide (NTO)
- Liquid Methane (LMe) and Liquid Oxygen (LOX)
- Monopropellant Hydrazine for both OMS and RCS.

The JP4/H2O2 propellants were selected over the MMH/NTO combination because the lower toxicity of JP4/H2O2 will facilitate ground servicing operations. Cryogenics, such as LMe/LOX, were not selected because of the substantial additional effort in design, manufacture, and operations required to accommodate vehicles with cryogenic propellants. Since batteries have been selected for primary power, there is currently no potential for the LMe/LOX propellants to used in a fuel cell to meet the primary power requirement and thereby provide sufficient advantage to overcome the additional effort and costs associated with cryogenic propellants.

Hydrogen peroxide as a monopropellant has been used in the X-15 and Centaur vehicle programs, although the JP4/H2O2 bipropellant combination does not appear to have been used in space. The substantial experience with bipropellants accumulated to date suggests that no technology problems exist.

Prime Power (WBS 1.6.7.9)

The electrical energy supply needed for the PLS to satisfy DRM-1 is about 168 kwh and is supplied by batteries with a weight of about 2700 pounds. These are silver-zinc (Ag-Zn) batteries with a limited recharge capability--about 20 charge-discharge cycles. Rechargeable silver-zinc batteries have been selected over one-shot Lithium Thionyl Chloride (LiThCl) batteries with a weight of about 1800 pounds because the reusable silver-zinc batteries are projected to have a much lower cost. Use of Ag-Zn batteries will also result in slightly more complex ground processing operations than for LiThCl batteries because they must be refrigerated between missions to preserve their rechargeability.

The battery packaging design will have to make provisions for a yet-to-be-specified level of redundancy such that the battery pack can fail operational if there is an internal short or other battery problem. All types of batteries present serious safety problems as they can rapidly vent large quantities of toxic gas/liquids or rupture/explode if they are abused, whether intentionally or unintentionally. This design for safety issue applies to all concentrated power sources and represents a significant design challenge, but not a technology problem.

The power system mass tradeoff point for the PLS, that power system mass which would cause batteries to be less desirable than the alternative of fuel cells occurs at about 2,000 pounds. While DRM-1 now appears to require about 2700 pounds of batteries, the potential exists for this power requirement to grow for DRM-1, and other missions may require seven times as much power (and therefore battery mass).

Accordingly, a fuel cell power system must be considered a principal alternative technology for the PLS. The STS Orbiter fuel cells have slightly more capability than is needed for the PLS, but since the cell stacks are designed with 4 kw modules, reducing the orbiter 12 kw fuel cell (with substantial extra surge capacity) to an 8 kw fuel cell is an easy design task. The resultant cell would match the PLS peak power requirement of 7 kw very well.

The major problem with adapting current fuel cells and associated equipment to the PLS is the lack of volume to accommodate the liquid hydrogen storage dewar(s) and related cryogenic plumbing. The PLS does not have the volume to accommodate both non-cryogenic propellants and the cryogenic fuel cell reactants. Also the technology to provide an Integrated liquid Hydrogen and Oxygen Technology (IHOT) propellants/fuel cell reactants will not be available until the mid 1990's, well after the PLS is to be designed. Many of the IHOT problems which need to be resolved concern fluid management techniques and very-long-term reusable insulation for the deep cryogenic liquid hydrogen (boiling point 20.28 degrees Kelvin). These problems with the mid-cryogenic liquid oxygen (boiling point 90.18 degrees Kelvin) are much less severe. Also, with liquid oxygen being denser than water (1.21 versus 0.07 for liquid hydrogen), the volume constraint is not severe.

Consequently, while LOX is both necessary and feasible as a fuel cell reactant, some alternative must be found for liquid hydrogen. The possibility of integrating Liquid Methane/LOX propulsion with LMe/LOX fuel cells was also considered. Unfortunately fuel cells do not use methane directly; it must be reformed into hydrogen and carbon oxides in a separate reactor at high temperatures (about 1,000 degrees F). The hydrogen is then separated and sent to the fuel cell. Such reactors have been built for use in large stationary ground power plants, but no significant design work has been accomplished for space applications.

Another option for a hydrogen source is hydrogen generating solid chemicals, specifically metal hydrides which are known to provide substantial reduction in storage volume with only moderate increases in weight over liquid hydrogen. Calcium hydride is used commercially in ground-based hydrogen generation applications, and Lithium hydride could be used to provide lower reactant weights. The reaction uses the water formed by the fuel cell from an initial supply of gaseous hydrogen. This water is sent to the metal hydride storage/reaction tank where the metal hydride strips the oxygen from the water to form a metal hydroxide. The hydrogen is then recycled to the fuel cell to react with stored oxygen to generate electricity in the process of forming more water. The effect is to make the power generating system operate as if it were an oxygen battery.

The major problem with the adaption of the metal hydride storage technique is that it has never been adapted to space, and a development program will be needed to provide confidence that the reactor will work as expected.

Electric Conversion & Distribution (WBS 1.6.7.10)

The direct current from the batteries will be distributed through a conventional copper wire harness. Voltage adjustments will be made at several different power conversion units. All alternating current devices will perform the DC-to-AC conversion at or near the device.

The PLS spacecraft will use a mixture of copper wire and fiber optic signal distribution harnesses. Substantial weight can be saved and substantial additional monitoring can be accomplished through the use of fiber-optic instrumentation. Unfortunately, optical technology is not yet at a state of maturity to select all-optical avionics, and thus a mix has been selected. A desired but not absolutely necessary part of this mixed wire/optical system is a laser initiated pyrotechnics system. This is believed to offer substantial safety and operational advantages in much lower probabilities of accidental pyrotechnic initiation.. Laser initiated pyrotechnic systems have been demonstrated in brassboard configurations. The major impediment to their adaption appears to be the accomplishment of enough testing to convince safety officers of their reliability.

Actuators (WBS 1.6.7.12)

Direct current electrical motors are replacing hydraulic actuators in aerospace applications. This replacement has been made possible by the development of high-strength permanent magnet materials. The use of electricity permits the elimination of the hydraulic power generation and distribution system with its attendant complexity and cost. Electric actuators are planned for aerodynamic control surfaces, landing gear, the top hatch cover, and to unfold the wings when the PLS is carried in the Shuttle payload bay.

In some applications such as aerodynamic control surfaces hydraulic actuators may be needed because they perform linear motion more effectively than electric motors. In these cases, the electrical motor will drive a local hydraulic pump which will transfer its energy to a hydraulic actuator. All electrical motors will require controllers to respond to the commands from the GN&C computers by sending correct electrical currents to the motors.

These electric motors represent a recent development which has already been adapted to new aircraft, but is only now being applied to space applications. Significant design challenges are anticipated, including the possibility that some form of active cooling may be needed by the motors during reentry. These design challenges, however, do not represent a technology development concern.

Avionics (WBS 1.6.7.13)

The avionics systems represent adaption of existing designs for computers, data buses, navigational instruments, displays, etc. to the requirements of the PLS. While there are significant design challenges, no technology development is foreseen to be needed in either hardware or software,

Ideally, the PLS would like to use phased array antennas and/or simple fixed antennas for all directions and all purposes. Ideally, these would be located under the TPS tiles and blankets. Carbon has three main forms, carbon black, graphite, and diamond; they have relatively high electrical conductivity. This conductivity causes attenuation of electromagnetic signals and is enhanced and applied in radar absorbing materials used in stealth aircraft. Accordingly, the lower heatshield tiles (FRCI) and other alternatives have sufficient carbon content that they would very likely attenuate signal strength too much.

Such antennas can probably be accommodated underneath the upper surface TPS (FRSI), which is made of modified quartz fiber. Such antennas would see only slightly more than a hemisphere, and this should be adequate for most purposes. The fall-back of deployable antennas is always available.

The selected PLS spacecraft design provides for passive cooling of the avionics in order to eliminate the complexity of active cooling plates during manufacturing and processing. This will be accomplished by using metal conduction paths to appropriate heat sinks. If heat buildup problems are anticipated, the avionics cooling can be integrated with the environmental cooling system. In either case, technology problems are not foreseen.

A vehicle health monitoring system (VHM) is planned to exceed that currently being used, but progress is needed primarily in designs and software to handle the data flows appropriately. VHM is expected to benefit from ongoing investigations in this area being conducted at many locations. Topics in this area represent a significant design challenge.

Substantial effort has been made to determine precursor events which signal impending problems in liquid rocket engines and which may be used to trigger the escape warning system. The knowledge base in this area has not yet been integrated with other monitoring information to provide a testable system. Accordingly some investment is needed to provide confidence that the booster alert/warning system architecture will be available on the PLS development schedule.

Environmental Control (WBS 1.6.7.14)

The Environmental Control Systems provides the air and air-conditioning for the crew. It consists of air tanks, make-up oxygen tanks, a solid amine cartridge assembly for carbon dioxide removal, and a single fluid (water) heat rejection/air-conditioning system. Only the solid amine cartridges represent a technology update; the STS orbiter is also planning to use solid amine cartridges in the future.

Personnel Accommodations (WBS 1.6.7.15)

This topic covers all crew accommodations other than the air/-air-conditioning system: seats and other furnishings including storage bins, water supply, and waste disposal. The waste disposal system represents Apollo technology: diapers and fecal bags. For future longer missions, a better space toilet is desired, but the volume to install it is very limited.

Recovery & Auxiliary Systems (WBS 1.6.7.16)

These are the mechanisms to separate the PLS spacecraft from the booster adapter, the docking system, and the parachutes. The landing gear doors and the wing unfolding latches are included here rather than under other titles. No technology issues have been identified for these topics.

Adapter System (PLS-to-Booster Adapter) (WBS 1.6.15)

This cone-shaped structure adapts the PLS spacecraft to the booster and has some subsystems of its own: Structure, Thermal Protection, Propulsion (Solid Rocket Motors), Power (Batteries), Electrical Distribution, Avionics, and Separation Mechanisms. The desire to reduce serial operations at the launch pad encourages the use of laser-initiated pyrotechnics. These represent a developing technology for the Adapter. Laser-initiated pyros have already been demonstrated, and require only sufficient testing to convince safety officers of their acceptability.

<u>7.2.2 Selections for Other WBS Elements</u>

The other elements of the second level of the WBS (as noted above) are:

- WBS 1.1 Fleet Processing System
- 1.2 Mission Processing System
- 1.3 Logistics System
- 1.4 Payload Processing System
- 1.5 Communications System

Major operational cost reductions over current space launch systems are required for a successful PLS program. Therefore, significant improvements over current STS practices will be required for WBS elements 1.1 and 1.2 (Ground and Mission Processing Systems). The goal is to adapt civil and military aircraft technology and practices to the PLS to achieve the lower costs associated with aircraft operations. Some specific examples will be cited later.

Existing technology and practices, together with normal progress and improvements, are expected to suffice for the PLS Logistics System (WBS 1.3) and Communications System (WBS 1.5). From the fact that PLS will be a relatively small program, it needs to adapt proven and low cost designs and practices to minimize costs. The logistics and communications systems are the PLS's links to the external world and thus must be compatible with the rest of the world. Accordingly, investment in significantly different methods would tend to be counter-productive. Therefore no technology development requirements have been identified during the brief examination of these WBS elements.

The Payload Processing System (WBS 1.4) for Design Reference Mission-1 (DRM-1) consists of crew training and related launch preparation. When other DRMs are examined, additional payload processing needs will be identified. Because payloads must fit through the access hatches, it is expected that most payload processing requirements can be met using the approaches and methods being developed for Ground Processing. Also, most internal payload preparation steps would normally be accomplished at the manufacturing site or in a separate launch site building. Only last-minute checking and initialization of payloads should be considered for the PLS processing facility.

Ground and Mission Processing (WBS 1.1 and 1.2) goals require approaching airline levels of staffing for space vehicles. This has never been accomplished before, and will represent a major advance in spaceflight capability.

A substantial investment will be required in design to minimize the need for operating personnel and in testing the design results as well as preparing the ground crew to operate with the vehicle. Some of this investment has already been made through this study's designs: the access panels and design for access and maintainability as well as the baselining of Built-In Test Equipment (BITE) to provide health monitoring of all systems.

Since the launch and space environment are more benign than many aspects of supersonic/hypersonic flight in the atmosphere, designing/testing/training for low operational costs is feasible. Among the factors permitting this judgement are:

- The PLS relies upon its booster to provide most of the Earth-to-Orbit propulsion so that the PLS is not burdened with complex, high performance engines.

- The PLS uses batteries rather than fuel cells, and electromagnetic actuators rather than hydraulics.

- The previously mentioned access panels and BITE facilitate appropriate maintenance and provide evidence that healthy systems do not need maintenance. Essentially, elements requiring substantial maintenance have been designed out, and the remaining elements have been selected and designed to permit aircraft maintenance philosophies to be implemented.

What remains to be designed into the PLS is the knowledge acquired from aircraft programs that most properly designed, manufactured, and tested systems can be reused with very quick, designed-in checks to affirm that they are still healthy. This advancement seems obvious, but will require the most difficult change of all -- a cultural change. The PLS will be the first vehicle to adapt this approach and will be surrounded by, and possibly use a booster which follows the older operating approaches.

From the standpoint of technology, no major advances are needed -- only a commitment to apply the existing and evolving designs and operating techniques which keep civil and military aircraft operating every day.

These activities represent the technological base from which the PLS designers can select to provide capabilities to meet their goals. No need for funding for technology development specifically for PLS is foreseen at this time.

7.3 TECHNOLOGY DEVELOPMENT PLANS

The following pages, Table 7-2 and 7-3, present technology development plans in two areas. Each plan has one sheet presenting a technology definition and assessment and another sheet presenting the components of the plan and the associated schedule and funding.

- Hydrogen Storage for Fuel Cell
- Booster Warning & Launch Escape Activation System

The cost numbers for the development programs were provided by cognizant experts working in their respective fields.

Table 7-2. Technology Plan - H2 Storage for Fuel Cells

Technology Category: Electrical Power

Major Technology: Hydrogen Storage for Fuel Cell

Description	Figures of Merit	Current Level	Required
Current: Cryogenic Storage of Liquid Hydrogen	Storage Volume	96.3 lbs in 21.4 cu.ft. Spherical Tanks	< 10 cu ft
Proposed: Calcium Hydride Reactor		1066 lbs in 9 cu ft Non-cryogenic tanks are easier & less expensive to design in all shapes	

Applications	Need Date	Criticality
For PLS: Reduced Storage Volume	1995	Contingent - Needed only if batteries do not suffice
Other Applications: Long-term Storage of Hydrogen	?	?

Operational Benefits

1. Reduced Storage Volume
2. Indefinite Storage Time
3. Avoid Deep Cryogen Storage/Use Hazards

Alternative Technologies

1. Lithium Hydride for weight reduction
2. Mixed Lithium & Calcium Hydrides for weight reduction and appropriate hydration properties
3. Highly non-spherical cryogenic storage tanks
4. Metal Hydride / Chemical storage of oxygen -- e.g. hydrogen peroxide-- Avoids all cryogens, but has substantial mass penalty.

Life Cycle Costs & Benefits

1. Life Cycle Costs should be slightly lower for hydride systems over liquid hydrogen; liquid oxygen will still be needed.
2. The major benefit is reduction in hydrogen storage volume. For PLS this is a contingent enabling technology.

Risks: Low

Development and schedule risks are low for all alternatives; phenomena are well understood.

Cost risk is slightly higher; unforeseen problems may require rework.

Table 7-2. Technology Plan - H2 Storage for Fuel Cells (Cont'd)

Technology Category:
Electrical Power

Major Technology:
Hydrogen Storage for
Fuel Cells

Agency / Contractor:
International Fuel Cells or
Other Fuel Cell Manufacturer

Description:
Adapt alkaline / alkali metal hydride generation of hydrogen gas to space use as an alternative to liquid hydrogen storage.

• Design container / reactor and associated plumbing and controls.

• Conduct Vibration tests to simulate launch / flight environment.

Objectives:
Verify that vibration does not impair ability of reactor to be operated and controlled in a flight environment.

Notes:

1. Zero gravity tests are not considered to be needed.
2. Special facility probably not needed.
3. Test of Lithium Hydride & /or mixtures of Lithium and Calcium Hydrides would range from $1 M to $10 M additional depending upon early test results.

Technology/Activity	1991	1992	1993	1994
1. Design container / reactor and associated plumbing & controls.				
2. Test container / reactor in vibration environment simulating launch and flight.				
Resources, $,M		3	4	3

Table 7-3. Technology Plan - Booster Warning & LES System

Technology Category: Avionics

Major Technology: Booster Warning & Launch Escape Activation System (Booster Warning System) (BWS)

Description	Figures of Merit	Current Level	Required
Current: Incomplete/partial systems	Demonstrated System Effectiveness	Partial Systems and Tests	Ground Test

Proposed: Establish criteria, architecture, simulate, reduce to ground hardware and test. Flight hardware is not desired.

Applications	Need Date	Criticality
For PLS: BWS design readiness demonstration	1994	Moderate -- Early tests reduce cost and schedule risk
Other Applications: All manned launch vehicles	?	STS does not have LES, but could use early warning

Operational Benefits

1. Improved Safety

Alternative Technologies

None - There are many alternatives for the design and components, but no alternative technologies.

Life Cycle Costs & Benefits

1. The cost for the demonstration will be 3 to 5 man-year equivalents.
2. The development cost for the PLS booster is low, probably less than 50 man-year equivalents through IOC and 3 to 5 man-years per year during the operations phase.
3. The benefit of developing a BWS/LES Activation System is the reduction of crew fatalities.

Based on Apollo / STS experience, the benefit of avoiding one crew loss is estimated as 2 years of the PLS operating budget.

Risks: Low

Development and schedule risks are low ; phenomena are well understood.

The proposed simulations and tests will eliminate bad architectures and confirm at least one good architecture.

Cost risk is slightly higher; unforeseen problems may require rework.

Table 7-3. Technology Plan - Booster Warning & LES System (Cont'd)

Technology Category:
Avionics

Major Technology:
Booster Warning and Launch Escape Activation System (BWS)

Agency / Contractor:
Rockwell International or Other Launch Vehicle Mfr. with Avionics Capability

Description:
Ground test an integrated Booster Warning System using standard electronic modules via a staged program:

- Establish Criteria - determine physical parameters to measure and how many sensors must agree.
- Decision Architecture Selection - Determine system organization and redundancy; e.g.: Triple redundancy vs Dual-Dual redundancy.
- Simulate sensor network / computer with software.
- Construct Brassboard computer with standard cards (e.g. JIAWG modules) and simulated sensor network.
- Test Brassboard system to confirm correct operation and rejection of false alarms.

Objectives:
Verify BWS design readiness. Flight hardware is not desired.

Notes:

1. Special facility not needed.

Technology/Activity	1991	1992	1993	1994
1. Establish Criteria		—		
2. Architecture Selection		—		
3. Simulate Architecture and Sensor Network		—		
4. Construct Brassboard with Standard Modules			—	
5. Test Brassboard			—	
Resources, $,M		0.2	0.3	

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ACRONYMS

AFDFP	Ascent Flight Design Freeze Point
AFSIF	Ascent Flight Systems Integration Group
APF	Adapter processing facility
AIRINC	Aeronautical Radio Incorporated
AMLS	Advanced Manned Launch System
ATA	Air Transportation Association
CAD	Computer-aided design
CAE	Computer-aided Engineering
CAM	Computer-aided Manufacturing
CDR	Critical Design Review
CIRD	Cargo Integration Review Dry Run
CIR	Cargo Integration Review
CITE	Cargo Integration Test Equipment
D&PSF	De-service and pyro safing facility
DCR	Data Change Request
DRD	Data requirements description
DRM	Design reference mission
Engr CCR	Engineering Cargo Compatibility Review
FDRD	Flight Definition Requirements Directive
FFBD	Functional flow block diagram
FH	Flight Hour
FIAB	Flight Integration Assessment Baseline
FOR	Flight Operations Review
FRR	Flight Readiness Review
FPSR	Flight Planning and Storage Review
FRD B/L	Flight Requirements Document Baseline
GFE	Government furnished equipment
GN&C	Guidance navigation and control
GR&C	Groundrules and constraints
Hdwr	Hardware
HPF	Horizontal processing facility
Integ	Integration
ISDS	Inadvertent separation self-destruct system
IUS	Inertial upper stage
LCC	Life cycle cost
LPS	Launch Processing System
LRU	Line replaceable unit
LSEAT	Launch Systems Evaluation and Assessment Team
LSFR	Launch Site Flow Review
M/P	Mass properties
MH/MA	Manhours per maintenance action
MMU	Mass Memory Unit
MTBA	Mean-time-between-abort
MTBM	Mean-time-between-maintenance
MTBR	Mean-time-between-removal
NDI	Non-destructive inspection
O&M	Operations and maintenance
OCD	Operational capabilities development
OFT	Operational flight test
OI-CL	Operational Increment Configuration Inspection
OMRS	Operations and Maintenance Requirements Specification
OTA	Office of Technology Assessment

PCAP	Preliminary Crew Activity Plan
PLS	Personnel Launch System
Prel FRD	Preliminary Flight Requirements Document
PTR	Performance Test Review
R/M	Reliability and maintainability
Recon	Reconfiguration
RFIT	Ready for integrated Training
RFST	Ready for stand alone training
SAIL	Shuttle Avionics and Instrumentation Laboratory
SAR	Search and Rescue
SDT	Shuttle Data Tapa
SIR	Systems Integration Review
SPC	Shuttle Processing Contractor
SRM	Solid rocket motor
TCTD	Terminal Count Down Demonstration Test
TDDP AFP	Trajectory Design Data Package Assessment Flight Profile
TDDP CFP	Trajectory Design Data Package Conceptual Flight Profile
UES	Universal Environmental Shelter
UMA/FA	Unscheduled maintenance actions per flying hour
UMPTS	Up Mission Processing Start

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Final report

16. Abstract

To assure national leadership in space operations and exploration in the future, NASA must be able to provide cost effective and operationally efficient space transportation. Several NASA studies and the joint NASA//DoD Space Transportation Architecture Studies (STAS) have shown the need for a multi-vehicle space transportation system with designs driven by enhanced operations and low costs. The NASA is currently studying a personnel launch system (PLS) approach to help satisfy the crew rotation requirements for the Space Station Freedom. Several concepts from low L/D capsules to lifting body vehicles are being examined in a series of studies as a potential augmentation to the Space Shuttle launch system. Rockwell International Corporation, under contract to the NASA Langley Research Center, has analyzed a lifting body concept to determine whether the lifting body class of vehicles is appropriate for the PLS function. This report discusses the results of the pre-phase A study.

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