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AD B076276

SPACEPLANE CONCEPT FEASIBILITY EXAMINATION

Final Technical Report
and Supporting Documentation

February 1983

Prepared for:

Air Force Space Division
Los Angeles Air Force Station
Los Angeles, California 90009

Sponsored by:

Defense Advanced Research Projects Agency
Strategic Technology Office
Advanced Concepts Division

Contract F04701-81-K-0001

SRI Project 3449

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The Spaceplane concept is a man-rated transportation vehicle that is compatible with either Shuttle, ELV (MX,TITAN) or air-launch capabilities. It is a near-term (late 80's) system utilizing low risk development technologies. It is capable of accomplishing short duration (24 hours or less with internal life support) military missions in or from space and capable of earth return and parachute recovery. This concept will provide autonomy, flexibility, maneuverability, responsiveness, survivability, and cost-effectiveness required of military operations.

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20. The concept involves transforming a reentry vehicle, designed by Sandia, into a manned spaceplane, capable of performing aerodynamic plane changes and accomplishing low earth as well as possible geosynchronous, orbit missions.

This report has been reviewed and is approved for publication.

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Approved:

Earle D. Jones, Vice President
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Research and Analysis Division

PREFACE

SRI International is pleased to submit this Final Report of the Spaceplane Examination contract to the Headquarters Space Division (AFSC) in accordance wih the Contract Data Requirements List (CDRL) Item No. A001 of the Short Form Research Contract F04701-81-K-0001.

The High-Performance Spaceplane (HPSP) or simply the "Spaceplane" system concept was conceived by Mr. Fred W. Redding, Jr., under contract DNA 001-79-C-0419 titled Strategic Concepts Analysis Support and contract DNA 001-81-C-0217 titled Strategic and Space Systems Concepts Analysis Support. The contracts were established to provide professional conceptual support on a timely basis to the Office of the Deputy Director, Defense Research and Engineering, Strategic and Space Systems (now named Strategic and Tactical Nuclear Forces). The contracts were administered by the Defense Nuclear Agency (DNA). The general tasks were to conceive and analyze strategic offensive, defensive and space systems as assigned and to analyze and evaluate other concepts designated by the Strategic and Space Systems Office. Assigned under the general tasks to provide a review and critique of Space Shuttle payload plans, options and alternatives from a military conceptual viewpoint with emphasis upon payloads with man in the loop, the idea of the Spaceplane was generated and approved. Two Spaceplane-specific tasks were then stated in the Work Statement to (1) prove the need and value of the high performance manned military spaceplane operating from the Space Shuttle and (2) prove the need and value of the high performance manned military spaceplane operating independent of the Space Shuttle. As a result of the work performed, the Defense Advanced Research Projects Agency (DARPA) began its sponsorship by funding a modification of the DNA contract as a task to analyze mission and payload requirements with emphasis on performing a broad range of missions. The result is evidenced by the multi-mission nature of the configuration. The DARPA add-on work was monitored by its Strategic Technology Office under DARPA Order No. 4097. In response to an unsolicited SRI proposal titled Spaceplane Research and Technology Analysis DARPA continued its sponsorship by funding the work reported herein under DARPA Order No. 4229 monitored by the Strategic Technology Office.

The work was administered and managed by the Department of the Air Force, Headquarters Space Division, Air Force Systems Command. Technical management was directed by Lieutenant Colonel Darryl W. Smith, Chief of the Advanced Concepts Division (YLXC) under the Deputy for Technology (YL).

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1.0 INTRODUCTION

Headquarters Space Division of the Air Force Systems Command contracted with SRI International (SRI) to define and evaluate a small man-rated space transportation vehicle or high-performance "spaceplane" for military space operations in cislunar space. The purpose of the contract is two-fold:

(1) To define and evaluate a small man-rated space transportation vehicle for military space operations which is compatible with the Shuttle, expendable launch vehicles or air launching and is capable of earth return and parachute recovery

(2) To investigate configuration changes necessary to accomplish selected "off-design" missions

The work was done by SRI supported by three subcontractors: Aerojet Liquid Rocket Company (ALRC); Hamilton-Standard Division of United Technologies; and the Government Products Division, Pratt and Whitney Aircraft Group (P&W) of United Technologies. The principal division of technical responsibility was:

ALRC	Storable propellant propulsion system
Hamilton Standard	Environmental control and life support system (EC/LSS)
Pratt & Whitney	Cryogenic propellant propulsion system
SRI	Spaceplane conceptual design and integration and program manager

Unfunded support by Honeywell Avionics Division of St. Petersburg, Florida, has been provided for the Spaceplane avionics definition and integration.

Unfunded support by General Electric has been provided to SRI and Hamilton Standard. GE's new spacecraft recovery system should result in a nearly ideal recovery system from supersonic initiation through soft landing at sites insufficient for non-controlled parachute landing and without weight or volume penalty relative to the non-controlled parachute system.

Rendezvous, station-keeping and docking are expected to be required routinely with the Spaceplane or "Space Cruiser". The integration of these capabilities with the pilot, the guidance, navigation and control subsystems, propulsion and vehicle airframe subsystems is best done from the outset. In this belief, SRI has obtained unfunded support from the LinCom Corporation of Houston, Texas, and Los Angeles.

The Spaceplane airframe subsystem and aerodynamics are based upon the IRBM "SWERVE" reentry vehicle geometry being developed by Sandia National Laboratories. Sandia was contracted by the Department of Energy under DARPA sponsorship for the associated work and support to this Spaceplane Examination.

General Research Corporation - Los Angeles Operations (GRC) performed mission analysis and payload definition tasks under the associated Spaceplane Context Examination Contract also sponsored by DARPA.

Lockheed Missiles and Space Company also provided unfunded consultation on overall vehicle integration, thermal protection systems, and airframe.

1.1 THE PROBLEM

The problem recognized in the previous contracted analyses* is the non-military characteristic and severely limited military capability of past, current, and proposed spacecraft at the time when the military need is substantial and increasing rapidly. Manned spacecraft programs and concepts have been and are continuing to be characterized by all or most of the following non-military characteristic examples:

- Space maneuverability which is limited severely
- Payload-maneuverability in space which is limited severely
- Inability to perform synergetic and other maneuvers in and out of the atmosphere
- Substantially constrained mission profiles
- Weather dependency of launch and recovery
- Launch schedule inflexibility
- Vulnerability of the launch facilities and the global ground support to direct attack
- Dependence throughout their mission on extensive ground support monitoring, tracking, control and communications.
- Little or no space rescue capability
- Dependence of orbital transfer vehicles on the Orbiter or future space station

These characteristics and capability limitations contrast sharply with the autonomy, flexibility, maneuverability, responsiveness, survivability and cost-effectiveness required of military aerospace operations. Furthermore, other manned space vehicle programs and concepts have reinforced the commonly held perception that the economics, technology, and safety of man in space inherently force the continuation of these non-military characteristics into the future.

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*DNA001-81-C-0217

The National Command Authority and the Department of Defense rely heavily on unmanned satellites as vital elements in command, control, communications, intelligence, surveillance, reconnaissance, and warning. Unmanned satellites have additional problems relative to manned vehicles such as inherent vulnerability to anti-satellites, single-mission utility and inability to adapt or to think. Balance and mutual support must be achieved between the manned and unmanned military space systems. The military manned vehicle must be capable of going where the action is, including to where the satellites are and can be in peace-time, conflict, and war.

1.2 THE NEED

The need then is to provide the military man in space a highly cost-effective, near-term vehicle system with the required military characteristics and capabilities that will 1) protect the United States resources from threats in and from space; 2) conduct needed aerospace offensive and defensive operations to use and protect the use of space by the United States and its allies; 3) enhance the land, sea and air forces; 4) serve as a practical utility vehicle in the support of space assets and in the exploitation of space; and 5) support as many aspects of U.S. national policy as possible, including arms control.

The specific vehicle need is for a truly military, multi-mission vehicle that integrates well with the Shuttle and other launch vehicles where required and that eliminates or minimizes the need for other vehicles or upper stages. The key performance requirement in space is payload-maneuverability or equivalently, payload-velocity. Whatever the payload weight and dimensions the maximization of achievable velocity is the "name of the game". Under the limitation of chemical-propellant rocketry and with the high velocities required for orbital maneuvers, clearly the maximization of the mass ratio in the rocket equation is required. This transforms into the need for the smallest practicable manned vehicle with the largest practicable amount of total impulse (propellant).

1.3 THE SOLUTION

The Spaceplane transforms the Space Shuttle Orbiter to an aircraft carrier in space and extends its military capabilities to higher energy orbits. The Spaceplane is a new generic type of piloted, rocket-powered airplane. It is configured for exoatmospheric, transatmospheric, and endoatmospheric flight and the maximum payload-maneuverability in space. The limitations exemplified in the problem statement of Section 1.1 are not inherent in the Spaceplane system concept. Compatible with ground and air launch the Spaceplane can also operate completely independent of ground operations and the Shuttle system. It can be cached on orbit, returned to the Orbiter or piloted to a landing at airfields or unprepared sites.

It differs considerably from the other manned and unmanned space vehicles that have been studied or proposed. It differs in configuration, cost, performance, ease, and speed of development.

Figures 1-1 and 1-2 preview the Spaceplane configuration that resulted from this Spaceplane Examination and reported in this Final Report. Study of the figures will assist in understanding the mate-ial presented in the report. External configuration detail of the Spaceplane has been omitted for security purposes.

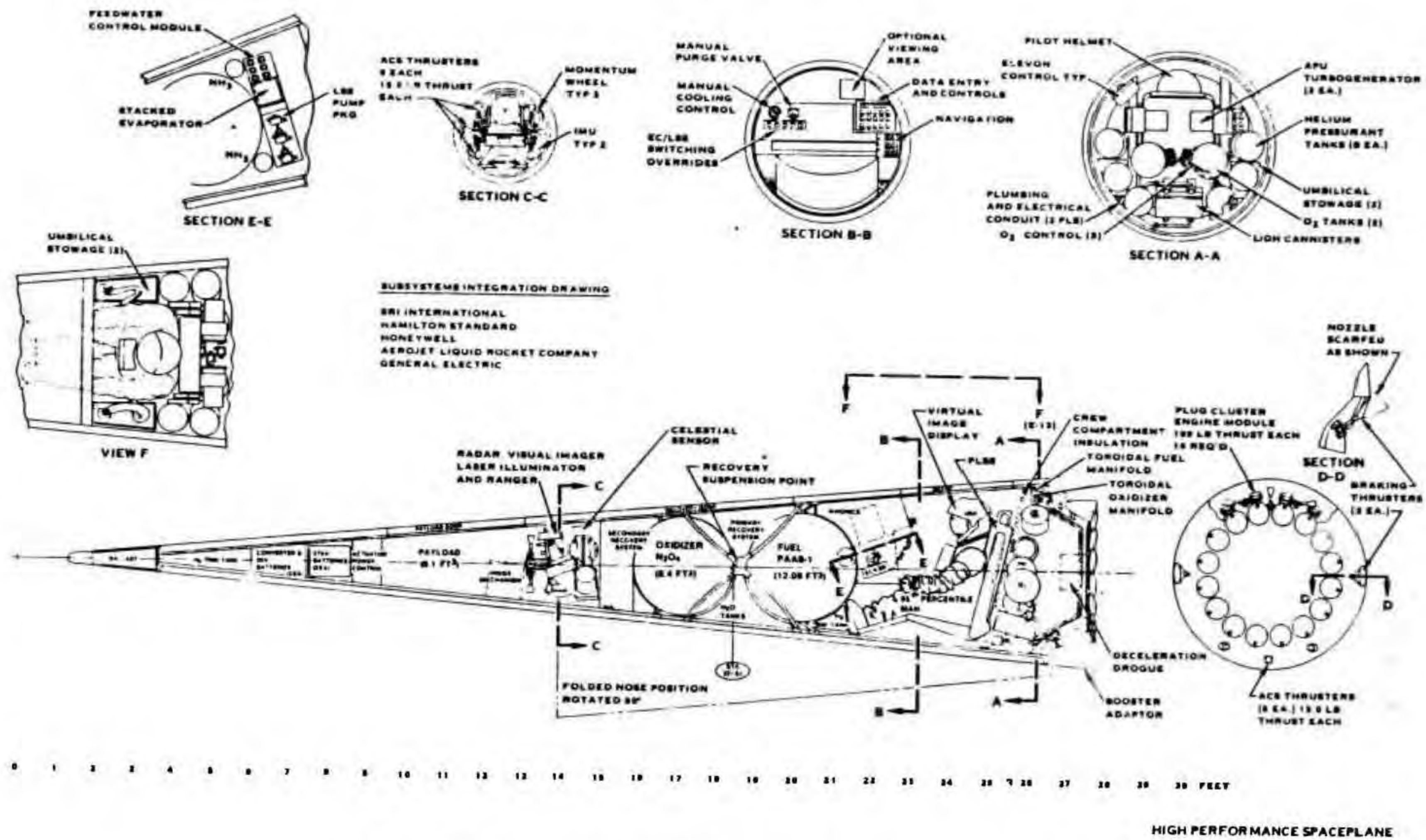


Figure 1-1 Spaceplane Internal Layout Preview

Figure 1-2 Centaur Spaceplane

2.0 CONCEPTUAL DESIGN

This section presents principal elements of the Spaceplane design rationale and introduces the configuration decisions that were made during the course of the Spaceplane Examination. The objective is to provide context and insight to the analysis, configuration decisions and details presented in the balance of the report.

2.1 SPACEPLANE DESIGN GOAL

The overall dèsign goal for the Spaceplane vehicle and its total system configuration is to obtain a near-term, truly military, multi-mission space vehicle. As a corollary, it must go where the satellites are, where the action is, where the need is.

To enable the successful attainment of the overall design goal the following set of criteria or design features were determined and used:

- Maximum delta-velocity... What the military needs in space is the capability to obtain as much velocity change as possible.
- Maximum payload-maneuverability... Whatever the payload weight may be, the design need is to maximize the velocity that can be imparted by the vehicle. Large payloads can be carried externally to minimize the volume constraints.
- Cislunar operations... Go where the satellites are or can go.
- Synergetically-maneuverable... The high delta velocity required to perform a substantial plane change in low earth orbit can be greatly reduced by using aerodynamic lift to perform the plane change. Propellant is only required to make up the velocity required to deorbit and that lost due to drag.
- Minimum weight and volume... Optimizes the Spaceplane's payload and velocity to orbit during the launch phase. Maximizes the available payload-velocity and permits reduction in transit time during maneuvers.
- Modular system... External mounting of large payloads, propellant, stages, life support consumables, support equipment, and sidecars.
- Launch options... Shuttle, air, ground-launched, expendable-launch vehicles. Reusable launch vehicle concepts such as AMSC type vehicles.
- Unmanned mode... May be advantageous for some missions.

- o Austere-site landing... Capability to land at unprepared sites, helicopter-suitable areas, etc.
- o Launch and forget/listen... Autonomous option with respect to ground operations, a military requirement.
- o State-of-the-Art... Accomplish the above within the state-of-the-art, using developed technology.
- o Minimize cost... Small, low-cost vehicle, reusable, rapid turn-around, maximum payload per flight, maximum maneuverability, minimum launch cost. Austere control and recovery support.

2.2 DISTINGUISHING CONFIGURATION DECISIONS

During the examination, numerous configuration decisions were made as the work progressed. The following list of decision results identifies and characterizes the Spaceplane configuration beyond that shown in Figures 1-1 and 1-2. Explanation and justification for the decisions are omitted here and found in the pertinent sections of the report.

The geometrical shape of the airframe internal moldline is conical, reflecting the basic conical shape of the reentry body. The reentry body airframe studied and tested by Sandia National Laboratories has small, extremely swept wings or "strakes" with elevons. The nose section containing the forward payload bay, ballast and power batteries extends in space to expose the forward reaction control nozzles for firing. No nozzles are located in the thermal protection structure (TPS) with this approach. The nose can be removed and replaced while in its extended position. The nose after full extension folds aft alongside and is snubbed near the nosetip while in the folded position. After the nose is folded, light-weight structure can be attached to the forward bulkhead or ring to attach the external payloads. In this way the payload is pushed by the Spaceplane. The pilot is seated at the aft end in a seat or couch that can be raised until the pilot's head is outboard, similar to an open cockpit aircraft. In the raised position the pilot can view the external payload. Also, the pilot can view the forward payload bay contents when the top panel or door is open. The airframe is advanced thermal tile over a composite non-metallic substructure such as graphite polyimide. Connections for refueling are located in the aft end with the PCE, rather than penetrate the TPS. There are two payload bays, one in the nose section and the other in the aft end among the PCE nozzles. Landing is by controllable lifting parachute or "Parafoil." The parachute is deployed from near the vehicle's center of gravity between the propellant tanks after deployment of a deceleration drogue from the PCE plug volume. A redundant, identical chute is located forward of the oxidizer tank. After deployment and disreefing of the lifting parachute the Spaceplane assumes a horizontal attitude for flight to the ground. The lifting aerobrake is reusable. A 195 lb, six-foot one-inch pilot or 95th percentile man is assumed. An 8 PSI EMU or spacesuit, under development, is planned. This suit eliminates the requirement for prebreathing before flight. The portable life support back pack is detachable before launch and after landing. Fail operational/fail safe design criteria are used for environmental and life support. Pumped fluid coolants are used with coldplates for heat transfer from the heat source hardware such as avionics. An expendable water heat sink is used for heat rejection. A helmet-mounted, internal, virtual-image display is provided. Voice control of the vehicle may be advantageous. An autonomous optical navigator with accuracy similar to the GPS is planned. Ring laser gyro inertial platforms are used in the guidance and navigation system. Monopropellant-driven auxiliary power units (APU's) are provided and integrated with the rechargeable power battery. The aircraft is all-electric. No hydraulics are used on board. The plug-cluster engine has 16 nozzles with independent on-off control for thrust vector and thrust magnitude control, thus eliminating actuators. The propellant tanks are spherical for light weight and are centered about the vehicle's center of gravity.

The propellants selected are nitrogen tetroxide as the oxidizer and a proprietary amine blend for fuel. The fuel is also used as a monopropellant in the APU's. The PCE nozzles are film-cooled. Elastomeric bladders are used in the pressurized propellant tanks. The attitude control system has nozzles mounted at the nose fold and with the PCE to provide six-degree-of-freedom attitude and translation control. Momentum wheels are provided for fine attitude control. A mercury trim control system is included for real-time, on-orbit CG trim for reentry stability. It is expected that outboard propellant tanks will be saddle-mounted to protect the TPS. As shown in Figure 1-2, the Wide Body Centaur upper stage was used as the reference for the external cryogenic stage. The Centaur would be modified by replacing the two RL-10 engines with a single RL-10 Derivative IIB engine. For overspeed reentry with the Centaur or entry of the Centaur alone, a lifting aerobrake could be attached to the aft end of the Centaur.

2.3 MULTI-MISSION CAPABILITY

The Spaceplane provides the opportunity for a high degree of mission flexibility because of:

Its configuration and performance that exploits both the space and atmospheric environments

Its maximum payload-maneuverability design

Man's unique on-site capabilities

Man-machine unification in the vehicle

Unmanned operation mode

Mission categories are exemplified by reconnaissance and surveillance; inspection and verification; anti-satellite and anti-anti-satellite; placement, supplementing and standing in for unmanned satellites; on-orbit service, repair and update of satellites; and missions requiring multiple atmospheric entry and exit. The Spaceplane transforms the Space Shuttle Orbiter into a multiple-aircraft carrier in space and extends its military operations throughout cislunar space. Compatible with ground and air launch, the Spaceplane can also operate completely independent of ground operations and the Shuttle system.

For very high delta velocity missions, such as rendezvous with a sequence of satellites or the placement or payloads in geostationary orbit at an altitude of approximately 20,000 nautical miles, the external cryogenic propellant propulsion module or stage is attached. Figure 1-2 depicts the Spaceplane with such a configuration. The module contains an RL-10 Derivative engine and cryogenic (liquid hydrogen and liquid oxygen) propellant tanks.

2.4 CONFIGURATION REASONING

2.4.1 Why Configured for Entry?

Entry capability is required for autonomous operation, proper energy management, and safety. Autonomous entry and recovery enables the Spaceplane to operate independent of recovery by the Orbiter. Proper energy management is vital to mission performance. Safety is vital to mission success, the pilot, and the avoidance of rescue costs.

In terms of energy management, the capability to enter and maneuver in the atmosphere enables important capabilities such as:

1. Maximizing the propulsive velocity available to do mission tasks when less velocity is required to reach the atmosphere and land than to return to the Orbiter or other rendezvous point

2. Aerobraking at perigee in the atmosphere rather than requiring retro-propulsion with its resultant weight penalty and loss of subsequent maneuver velocity

3. Use of aerodynamic lift to change the direction of flight (orbital plane change) and then to return to space flight. This energy-efficient maneuver is called the synergistic plane change and is efficient for a vehicle with the lift-to-drag ratio and low drag of the high slenderness ratio configuration of the Spaceplane

4. Use of aerodynamics to maneuver to a landing point on earth and to minimize pre-entry propulsive maneuvers.

In safety terms, the entry capability provides a recovery return choice between the earth and a space station such as the Orbiter as a function of the time available to reach sanctuary; the specific failure, problem or damage that forced the premature recovery or abort, medical needs, or subsequent docking risks to the Orbiter. The Spaceplane can serve as a rescue vehicle for the Orbiter or a space station.

2.4.2 Why The Generic Conical Shape?

The cone is the most understood and tested shape for reentry. It is the shape of the ballistic missile reentry body for reasons in concert with the Spaceplane, particulary the need for low drag. Alternatively, the conical ellipse is well known for its high lift-to-drag ratio while retaining the low drag of the cone. These result in the minimum loss of velocity during the endoatmospheric maneuvers. Therefore the least amount of propellant is consumed in returning to orbit. The cone and conical ellipse present the smallest surface area consistent with high aerodynamic performance and especially if ablative material is used as the outer surface, surface area means weight in the thermally protected reentry body.

Minimization of vehicle weight is vital to maximizing propulsive maneuverability in space, the maximizing of vehicle payload capability, and to the performance and size of any Spaceplane launch vehicle, whether it is the Orbiter, an expendable rocket, or a launch aircraft.

The conical or low eccentricity elliptical cross-section conical shape is correct for the generic highly maneuverable spaceplane. Orbiter-like vehicles designed to meet a large, internal payload volume specification for returning payloads to earth require the large winged, non-axisymmetric shapes exemplified by the Orbiter but are penalized greatly in weight and performance. Because there is no drag in the vacuum of space, payloads can be carried externally in space and under a launch vehicle shroud or in the payload bay of the Orbiter during launch. The size and weight of the Spaceplane is therefore minimized, resulting in the best payload-maneuverability performance.

We may conclude that the conical reentry body is best used for vehicles with specifications for maximum payload-maneuverability, maneuverability with small internal payloads, synergistic plane changes, lightest weight, compatibility with launch by the Orbiter or the MX ICBM booster, the Titan booster, near-term availability and lowest cost. Other shapes can be best where substantial internal payload volume from space to the ground is the driving requirement.

The latter type of vehicle is exemplified by the Orbiter. Vehicles of this type may be characterized as logistic vehicles. That is, they are principally payload launch and recovery vehicles for payload operations with respect to low orbit. Their use in higher orbits or for high velocity-change maneuvers is not cost-effective or generally practical.

2.4.3 Aircraft Launch

Preliminary analysis of aircraft launch for the Spaceplane showed that the combination of the Boeing 747-200F freighter aircraft and a rocket launch vehicle or booster could place greater than 15,000 lbm into low orbit. Therefore the Spaceplane under a shroud with a large payload and/or propellant in Spaceplane-external tanks could be orbited. Alternatively, two Spaceplanes could be orbited by one launch vehicle. Aircraft launch would be a flexible means for delivering large payloads to orbit without the Spaceplane where they could then be transferred to orbits by an already orbiting Spaceplane.

The launch vehicle is of conventional design. The Titan LR-87 type engine would power each of two strap-on boosters attached to a two-stage core rocket powered by a single chamber LR-87 engine on the first stage and one Pratt and Whitney RL-10 Derivative IIB engine on the second (final) core stage. Liquid oxygen and RP-4 or commercial liquid propane would be the propellants for the Titan engines. Rapid development, military characteristics, ease of handling, high performance and low cost to orbit could be obtained with this mission-flexible launch vehicle and aircraft combination.

2.5 PERFORMANCE GOALS

Velocity with internal propellants	4000 fps
Velocity with cryogenic propulsion	>25000 fps
Total velocity without payload	>29000 fps
Payload to geosynchronous orbit (includesSpaceplane	10000-12000 lbm
Velocity to payload of 160,000 lbm	3700 fps
Endurance with internal consumables	24 hr
Endurance with external consumables	days to weeks
Number of aircraft per Orbiter bay	
with internal propellant	8
with Centaur cryogenic propulsion module	1

Launch options
- Shuttle
- Titan
- MX booster
- Aircraft launch

Recovery
- Flying parachute
- Unprepared site
- Helicopter-compatible site or pad

Turnaround Time — Similar to High Performance Aircraft

Crew

Pilot

Multiple-passenger sidecars in space

Weight

Dry	4000 lbm
Wet	5600 lbm
Wet with auxiliary fuel in bays	6300 lbm

3.0 SPACEPLANE INTEGRATION

Introduction

The integration of Spaceplane subsystems reflects a team effort on the part of the following organizations:

- Sandia National Laboratories
- General Electric
- Aerojet Liquid Rocket Company
- Hamilton Standard
- Honeywell Avionics Division
- Pratt & Whitney Aircraft - Government Products Division
- SRI-International
- Wright Patterson AFB (AMRL)
- Space Division (USAF)
- Aerospace Corporation
- School of Aerospace Medicine (USAF)
- NASA/Johnson Space Center

The airframe is based on the Sandia Winged Energetic Reentry Vehicle (SWERVE), under development by Sandia National Laboratories. This conical maneuvering re-entry body vehicle, characterized in Figure 3-1, presents a significant challenge in terms of integrating the numerous systems and subsystems into a small volume, while maintaining the specified CG limits required for aerodynamic stability. Spaceplane, as an operable vehicle, requires integration of the following subsystems and equipment.

- Thermal Protective System (TPS)
- Lift Control Surfaces
- Environmental Control/Life Support System (EC/LSS)
- Substructure
- Ballast
- Electric Power Subsystem
- Life Support
- Avionics and Communications
- Controls and Displays
- Recovery System
- Propulsion/Attitude Control
- Pilot/Couch
- Internal Payload/External Payload

3.1 VEHICLE LAYOUT

Looking at the vehicle defined in Figure 3-1, it is clear that a judicious use of volume inboard of the structure must be made to accommodate packaging of all subsystems. In addition, subsystems must be packaged to be capable of withstanding acceleration loads during launch, re-entry, and synergistic plane changes. The challenge in subsystem equipment integration is to achieve a high packaging density, thereby keeping the vehicle small. Packing density P_D is defined as:

$$P_D = \frac{\text{Total subsystems volume}}{\text{Total vehicle packaging volume}} \times 100$$

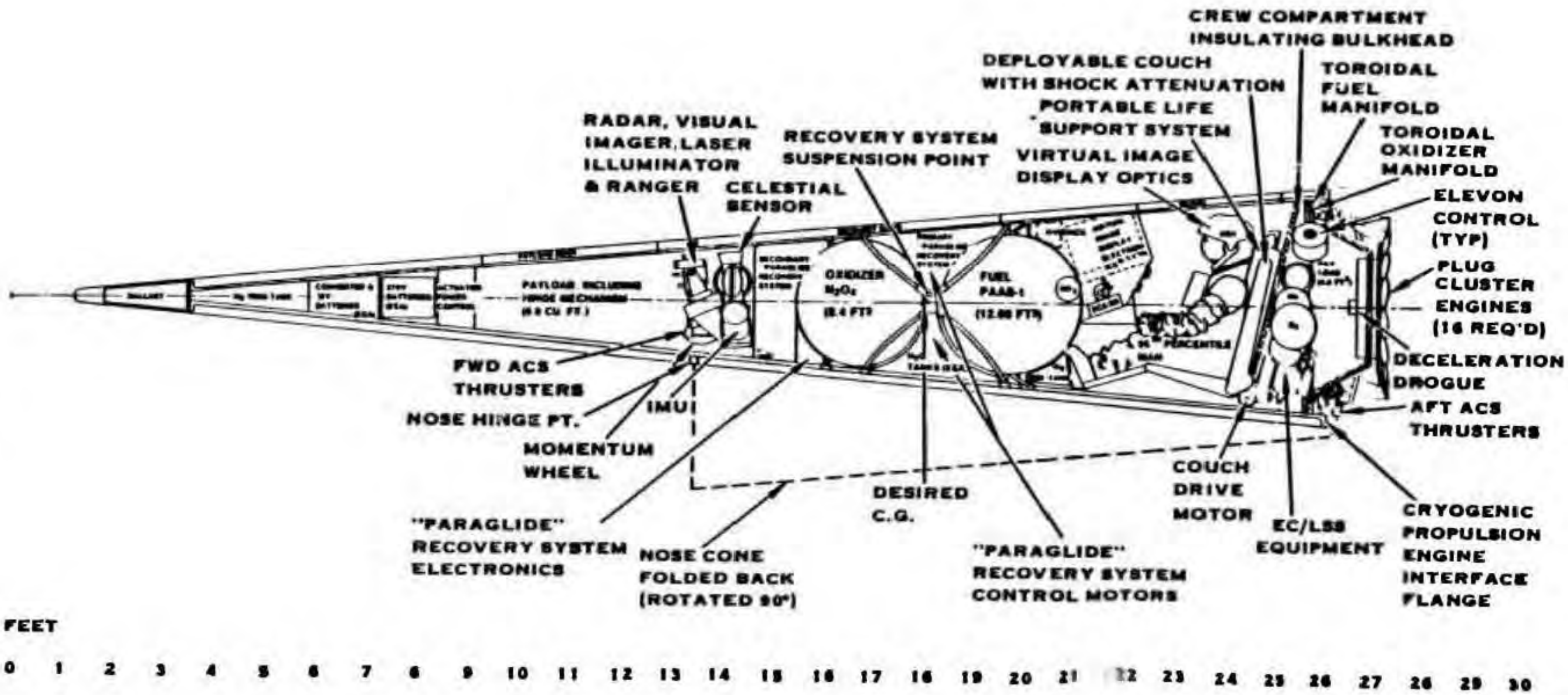


FIGURE 3-1 VEHICLE LAYOUT

For example, the Apollo portable life support system in the space suit system achieved a $P_D = 45\%$. In the Space Shuttle program, the new Portable Life Support System (PLSS) in the EMU has a $P_D = 80\%$. Results of Spaceplane integration studies indicate a $P_D = 62\%$, excluding volume allocated for payloads. Key considerations in the work of increasing Spaceplane packing density are:

- Optimize subsystem functional schematic
- Minimize component volume
- Conformal component packaging
- Combine components
- Optimize fluid line sizes
- Custom designs

Therefore, selection of equipment has involved technology that will result in low volume and weight. In addition, control of the vehicle center of gravity in both radial and axial directions is important to vehicle dynamic performance. Therefore, judicious placement of equipment and consumables, to minimize CG shifts, is required. Another issue considered was positioning of the pilot to provide adequate mobility to accomplish required tasks and withstand acceleration loads without physiological impairment. These issues and others were taken into account in the integrating of equipment and crew in Spaceplane.

The basic geometry of the Spaceplane concept is shown in Figure 3-1. A detailed vehicle integration layout, with various cross-sectional views, is included in this report as Appendix D. The conical vehicle is approximately 26.5 ft long, with an included cone angle of 10.5 and a dry weight in the range of 4,000-5,000 lbs. The nose cone of the vehicle is hinged to fold back on-orbit to provide payload access and forward viewing. The various subsystem equipment to be integrated is listed in Figure 3-2, including weights and volumes of the various subsystems.

Major constraints in configuring the vehicle layout were: minimum desired vehicle weight and length, specified center-of-gravity, folding nose, payload requirement, vehicle conical geometry, minimum specified propulsion system V and need to protect the pilot from g-loads. The folding nose feature influenced hardware packaging to the extent that equipment involving fluid coolant or propellant was packaged away from the nose section to avoid requirement for high-pressure, flexible, fluid lines. Ballast required for CG trim and electrical system equipment, including batteries, were located in the nose because of their high density. To minimize ballast required to trim for the desired CG, the conical shape of the vehicle dictated that, wherever possible, heavy components be packaged forward, with light components toward the rear of the vhicle. Major consumables, such as evaporator cooling water and propellant, were balanced about the CG in such a way that their consumption during the mission would result in minimal shift in overall vehicle CG. Volume for ballast was located at the vehicle nose tip, where it would have the longest lever arm to the vehicle CG.

SUBSYSTEM	WEIGHT (LBS)	VOLUME (FT^3)	COMMENTS
THERMAL PROTECTION SYSTEM (TPS)	600	29.0	LIGHTWEIGHT TILES
LIFT SURFACES	360	—	CARBON-CARBON, OUTSIDE OF CONE
SUBSTRUCTURE	550	20.0	GRAPHITE-POLYIMID
BALLAST	TBD	1.0	VOLUME AVAILABLE FOR 500 LB TUNGSTEN
APU'S, BATTERIES, CONDITIONING	215	3.9	RECHARGEABLE BATTERIES
LIFE SUPPORT (WET)	389	7.8	FAIL OP/FAIL SAFE WITH EMU, 100% CONSUMABLES REDUNDANCY
AVIONICS AND COMMUNICATIONS	274	5.3	
ELEVON CONTROL SYSTEM	80	0.5	ELECTRIC SYSTEM (MOTORS, LINKAGES, ETC.)
RECOVERY SYSTEM	342	9.0	GE RECOVERY SYSTEM, INCLUDES DECELERATION DROGUE AND BACKUP
PROPULSION SYSTEM	83	6.0	PLUG CLUSTER ENGINE (16)
ATTITUDE CONTROL SYSTEM	45	0.5	
PROPELLANT, PRESSURANT AND TANKAGE	1691	20.5	2500 FPS VEHICLE ΔV MINIMUM, APU PROPELLANT
PILOT, EMU AND COUCH	480	17.0	95TH PERCENTILE MAN
PAYLOAD	250	12.0	TO BE DEFINED
TOTAL *	5359	132.4	

FIGURE 3-2 OPERATIONAL SPACEPLANE SUBSYSTEM CHARACTERISTICS

*total does not include ballast (see Figure 3-3)

A ballast trim capability has been included, using mercury that can be pumped between tanks fore and aft. The volume allocated for payload was located near the hinge point so that, on-orbit with the nose folded back, the payload would be accessible and have forward viewing capability. Folding the nose back on-orbit also provides required viewing for rendezvous and navigational electronic equipment, which has also been packaged at the nose hinge point. Both the primary and secondary recovery systems were located near the vehicle CG to minimize shroud-line runs to the vehicle load-bearing suspension ring and permit the vehicle to descend under the shroud in a controllable fashion. A description of the recovery system is included in this report as Appendix E. The pilot was located toward the rear of the vehicle to provide adequate volume for task mobility and positioning in an upright position to offer protection against the physiological effects due to axial-directed g-loads during Spaceplane flight maneuvers.

A hatch is provided above the pilot to provide vehicle ingress/egress and to permit open-hatch viewing in orbit. The couch is deployable so that, on-orbit, the pilot can open the hatch, remaining supported by the deployed couch with the body extended through the hatch. This capability provides the pilot with direct viewing of such space operations as rendezous and docking. As indicated, the Spaceplane cockpit and couch have been sized to accommodate a 95th-percentile male pilot.

Most EC/LSS equipment and APUs have been packaged behind the pilot. This position places components requiring venting of water, ammonia vapor, and APU gas exhaust through lines to the low-pressure base region toward the rear of the vehicle.

A second payload compartment is provided in the Spaceplane layout of Figure 3-1. An aft compartment is provided with access through a door located on the base enclosure.

The calculated CG, based on vehicle structure, skin, pilot, all on-board equipment and consumables, and a 250-lb payload in the forward compartment, was found to be behind the desired CG location.

This results in a requirement for 388 lbs of ballast at the vehicle nose to move the calculated CG forward to coincide with the desired CG. Figure 3-3 shows the ballast required for various combinations of payload weights in the aft and forward compartments, together with overall vehicle weight. payload weights heavier than those of Figure 3-3 are possible, although, in general, fixed compartment volumes will be the limiting factor in payload capacity.

3.2 OFF-DESIGN MISSIONS

Several Spaceplane off-design missions are of interest. However, these missions would have an impact on baseline EC/LSS choices as defined to satisfy the 24-hour DRM.

PAYLOAD AND BALLAST WEIGHT (LBS)			VEHICLE WEIGHT (LBS)	
P_{FWD}	P_{AFT}	B	W_{WET}	W_{DRY}
314	250	500	6173	4673
250	201	500	6060	4560
250	0	388	5747	4247
0	15	500	5624	4124
0	0	492	5601	4101

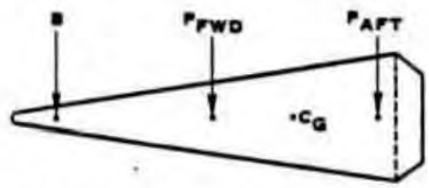


FIGURE 3-3 VEHICLE WEIGHT SUMMARY

Unmanned Vehicle

Certain missions might be accomplished with an unmanned spaceplane. For these missions, there would be no requirement for life support equipment and consumables to sustain the pilot. Analysis shows that approximately 31 ft^3 of volume and 985 lbs could be freed up by removing the pilot and equipment and consumables required to support the pilot, as defined in the baseline EC/LSS. That portion of the EC/LSS required for thermal control of avionics and payload equipment would, however, be required. If the freed-up volume of 31 ft^3 were replaced with propellant, then Spaceplane would have approximately the same total wet weight as the manned version of the vehicle, but would more than double the DRM baseline ΔV capability of 2500 fps.

Extended Duration

The practical time limit on the Spaceplane pilot being space-suited is estimated to be somewhere between 1 and 3 days. This limit is based primarily on considerations of pilot comfort and hygiene. Therefore, if Spaceplane missions of extended duration are to be considered, habitat provision will be required for the pilot to doff the EMU periodically to attend to personal hygiene. One approach would be for Spaceplane to carry along a sidecar capsule that would have an airlock and EC/LSS provision. The Sidecar EC/LSS provision might be obtained through direct interface with Spaceplane systems. Extra Spaceplane consumables for extended missions could be carried in external tanks mounted on the sidecar.

A cylindrical aluminum sidecar, with airlock, having hemispherical ends and dimensions of 4' dia. x 12' long x 0.25" thick would weigh approximately 1,000 lbs. If the mission carried Spaceplane through radiation-intense portions of the Van Allen belts, then an increase in the sidecar thickness (e.g., 0.5") might be required to provide the pilot added radiation protection. Depending on the orbit, the Spaceplane pilot might be required to spend a portion of each day in the sidecar capsule on extended-duration misisons to maintain safe radiation-dosage margin. Also, to minimize expendable water for Spaceplane cooling, it is envisioned that a radiator would be used for heat rejection.

The additional weight for the extended-duration mission, beyond that for the 24-hour DRM, is estimated to be approximately 200 lb/day, as indicated:

Spaceplane Consumables for Extended-Duration Missions

- H_2O Cooling - approximately 90 lb/day
- APU Propellant - 100 lb/day
- LiOH - 13 lb/day
- Waste Management - 8 oz/day (wipes, bags)
- Food - 16 oz/day
- Drinking Water - 36 oz/day
- Oxygen - 5 lb/day

Thus, for a 30-day Spaceplane mission, the total approximate Spaceplane wet weight (lbs) would be:

Spaceplane	6,000
Sidecar	2,000 (0.5" thick)
Consumables	6,000 (200 x 30)
	14,000 lbs

Note that if a deployable Solar array electrical system were used, an additional weight saving might be realized, because only a small portion of the propellant (100 lb/day) would be required. A deployable space radiator heat sink would save an additional 90 lb/day.

4.0 ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

Introduction

In support of the SRI International High-Performance Spaceplane Examination Study for the USAF Space division, Hamilton Standard has served SRI as the Environmental Control & Life Support System (EC/LSS) subcontractor to assist in the total specification, preliminary design, and/or selection of subsystems required to transform strategic maneuvering reentry body technology into a piloted, military, high-performance Spaceplane. The Spaceplane shall be capable of accomplishing a diverse set of missions throughout cislunar space and in the upper atmosphere. The objectives of Hamilton Standard's EC/LSS studies were:

1. To provide a thorough definition of the EC/LSS requirements for the Spaceplane.

2. To define an EC/LSS that provides required performance parameters utilizing existing technology/hardware wherever possible. Consideration was given to both the Design Reference Mission (DMR) and several alternate missions to ensure EC/LSS compatibility with overall mission objectives.

3. To identify EC/LSS technological issues and prepare a plan for completing subsequent program phases in an efficient and economical manner.

4. To support SRI in the overall integration of Spaceplane subsystems.

5. To present findings in a final technical report and presentation.

6. To support SRI as required to ensure program success.

This section of the report documents results of EC/LSS and vehicle subsystem integration studies.

Many sources of information were utilized during the data-gathering phase of the study. An attempt was made at each facility contributing data to coordinate Hamilton Standard's request for data through a single individual. The facilities contributing data and the individuals who provided the coordinating function are as follows:

Mr. Fred Redding	SRI International	Arlington, VA
Maj. Darryl Smith	USAF-Space Division	Los Angeles, CA
Mr. George Wright	Sandia National Laboratories	Albuquerque, NM
Mr. Roland Mayer	General Electric	Philadelphia, PA
Mr. Kirk Christensen	Aerojet Liquid Rocket Company	Sacramento, CA
Mr. William Isely	Honeywell Avionics Division	St. Petersburg, FL
Mr. James Brown	Pratt & Whitney - Aircraft Government Products Division	West Palm Beach, FL
Mr. Robert Van Patten	Wright Patterson AFG (AMRL)	Dayton, OH
Mr. William Haynes	Aerospace Corporation	Los Angeles, CA
Mr. Wil Ellis	NASA-Johnson Space Center	Houston, TX
Mr. Clark Neily	LINCOM Corporation	Houston, TX

Hamilton Standard personnel who actively participated in this study program were:

Mr. Donald Stein	Study Manager
Dr. Harrison Griswold	Engineering Manager
Mr. Jeffrey Wehner	Analytical Engineer
Mr. Philip Heimlich	Senior Design Engineer
Mr. David Faye	Design Engineer
Mr. Paul Tremblay	Reliability Engineer
Mr. John Nason	Analytical Engineer

SUMMARY

The integration of man into a maneuvering reentry vehicle has been accomplished with an Environmental Control and Life Support System design that combines existing shuttle Extravehicular Mobility Unit (EMU) hardware with components that are derivatives of other space life support equipment. Standard design practice, coupled with knowledge of manned systems, has made possible a vehicle integration that provides a quick-response, military spacecraft. This capability will be required in the future as the United States increases its utilization of space.

The Design Reference Mission (DRM) used in this study as the basis for the EC/LSS design is characterized as follows:

- Modified SWERVE vehicle
- One-man crew
- Shuttle launch - Extravehicular Activity (EVA) to Spaceplane vehicle
- Compatible with ground launch
- 24-hour mission duration
- GEO excursion possible
- EVA capability

The environmental control and life support system proposed for Spaceplane incorporates:

- 8 psi EMU and oxygen vent loop
- Fail-operational/fail-safe reliability
- Pumped water coolant loop
- Evaporative cooling heat rejection
- Unpressurized vehicle cabin

The EC/LSS, which includes expendables necessary for life support for a 24-hour mission that includes an excursion to geostationary orbit (GEO), provides EVA capability through the EMU.

Consumables provided in the EC/LSS include food and water, LiOH for CO_2 removal from the vent loop, oxygen for breathing and water, and ammonia for evaporative cooling. The pilot is suited in the EMU throughout the mission and is connected to the vehicle EC/LSS through umbilicals while seated in the couch. The EMU interface with the vehicle EC/LSS is shown at the left of the schematic. When conducting EVA, the pilot disconnects umbilicals at the suit and is supported by the self-contained EMU life support system. Redundant loops and components are used to satisfy a fail operational/fail-safe reliability criteria. Waste management systems for urine, vomitus, and feces are integrated within the EMU. The pilot couch, controls and displays, and other crew/hardware interfaces have been designed to interface with the EMU and be compatible with dexterity capabilities of a pilot's pressurized glove.

4.1 REQUIREMENTS DEFINITION

In this section, those aspects of the Design Reference Mission and the Spaceplane vehicle that impact the EC/LSS design are identified. The EC/LSS is then defined with respect to the functions that it must provide to adequately support the vehicle and pilot. Finally, a design specification is given for each EC/LSS subsystem required to satisfy the vehicle EC/LSS requirements. Additional EC/LSS performance, design, development, and verification requirements are presented in this report as Appendix A.

MISSION DEFINITION

Future military missions in space will require a manned vehicle with a high degree of autonomy, flexibility, maneuverability, responsiveness, survivability, and cost effectiveness. Such a vehicle must possess the following characteristics:

- Independent of extensive ground control, monitoring, tracking, and communications
- Capable of space rescue operations
- High degree of maneuverability in space
- Large external payload provision
- Flexibility in mission profiles, launch schedules, reentry conditions, and recovery
- Launch mode flexibility and survivability
- Support of astronaut extravehicular activity.

The Spaceplane defined in this study is a straked-wing, conical reentry vehicle, approximately 26 feet long, with a base diameter of 56 inches. For cislunar and other high-velocity change missions, an external propulsion module with cryogenic propellant tanks is required. An internally stored propellant propulsion system is standard in the basic vehicle to provide propulsion for many types of missions in low-to-medium altitude orbits. External tankage can be added for increased velocity change beyond internal propulsion system capability. The Spaceplane is compatible with launches using the Space shuttle, aircraft, and a family of expendable launch vehicles. Land recovery of Spaceplane by a controllable parachute is baselined.

The Spaceplane provides the potential for performing space missions with truly military characteristics. Spaceplane could be used to defend satellites, gather reconnaissance information, and conduct communications and intelligence missions. The vehicle could also fill an important role through the service of geosynchronous-orbit communications satellites, because extravehicular activity (EVA) by the pilot is included in the baseline concept.

The EC/LSS must support a 95th-percentile pilot on a mission of 24-hour duration, including an excursion to GEO, under accelerations in excess of 3- g's. The EC/LSS must integrate into the Spaceplane vehicle defined by the geometry of the SWERVE, where minimum length will be selected to accommodate a single seated pilot and required systems.

Sections 4.1.1 to 4.1.6 below define those aspects of the Design Reference Mission (DRM) that impact the design of EC/LSS. Section 2.1.7 includes discussion of the impact on EC/LSS of proposed alternate missions.

4.1.1 Environment

The Design Reference Mission (DRM) calls for a Spaceplane launch into low earth oribt (LEO), using the Space shuttle at optional inclination and altitude. This launch mode will require the pilot to perform extravehicular activity (EVA) from the shuttle cabin to enter the Spaceplane carried aloft in the shuttle payload bay. Following deployment from the Space Shuttle, the Spaceplane will operate in space from LEO up to GEO altitudes. Depending on the mission, Spaceplane will pass through portions of the Van Allen Belts and will experience darkness for approximately 38 minutes of every orbit when on the dark side of the earth. The level and total dosage of radiation is dependent upon the orbit track, as indicated in Figure 4-1. As shown, for low earth 28-1/2 and 55 orbits, radiation exposure is limited to short intervals during each orbit when the vehicle passes through the South Atlantic Anomaly (SAA) or Polar Horns. Higher exposures to radiation will occur on higher attitude orbits where the spacecraft spends a significant portion of each orbit within the Van Allen Belts.

4.1.2 Duration

The duration of the Spaceplane DRM will not exceed 24 hours. The 24-hour duration is defined as elapsed time from Spaceplane deployment from the Space Shuttle to Spaceplane terrestrial touchdown following reentry and recovery system deployment. Certain intelligence gathering and surveillance missions could be of much shorter duration (on the order of 4 to 5 hours).

4.1.3 Launch

The DRM baselines a Space shuttle launch of Spaceplane at an optional altitude and inclination. The Space Shuttle Cabin EC/LSS will support the pilot prior to entering the Spaceplane in the Shuttle payload bay. Once Shuttle achieves orbit, the pilot will don the EMU spacesuit in the Shuttle airlock, for egress from the airlock into the payload bay to enter Spaceplane. Following checkout of systems, the pilot will switch to Spaceplane EC/LSS for the duration of the Spaceplane mission. The probable Spaceplane deployment technique would utilize the available shuttle Remote Manipulator System (RMS) to transport the Spaceplane out of the payload bay.

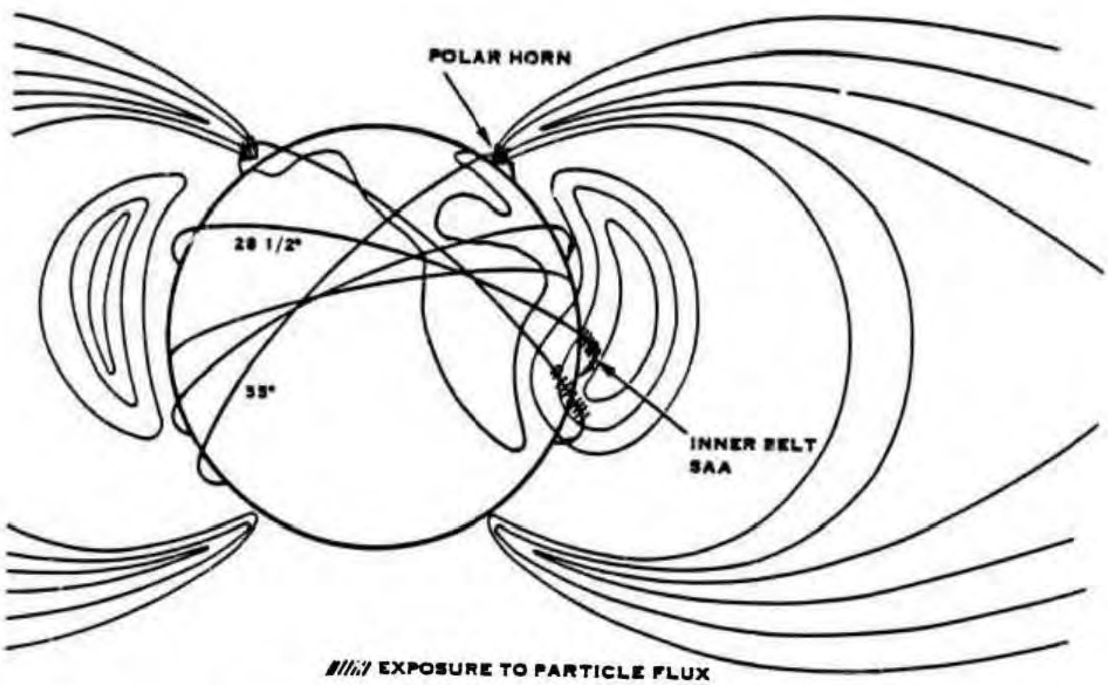


FIGURE 4-1 RADIATION DEPENDS ON ORBIT TRACK

4.1.4 Space Operations

The Spaceplane will include on-board propulsion and attitude control systems for use during satellite rendezvous and docking, as well as for other orbital maneuvers.

The propulsion system thrust will produce accelerations up to 1-G. During rendezvous and docking flight operations, the pilot will require visual verification of relative attitude of Spaceplane to the satellite. Some missions may require an orbital plane change that will be accomplished by reentering the earth's atmosphere, where the Spaceplane control surfaces can be used to effect the plane change. The maximum g-loads during these maneuvers will be 3-g's sustained for not more than 60 seconds.

Some missions may require extravehicular activity (EVA). During EVA, the pilot will be supported by the EMU. The maximum EVA duration during any given mission will be 7 hours, with EVA scheduled to safely limit pilot exposure to Van Allen Belt radiation.

4.1.5 Reentry/Landing

The Spaceplane will reenter the earth's atmosphere and maneuver to dissipate momentum. The maximum deceleration expected will be 3-g's sustained for not more than 60 seconds. Following this period of initial reentry deceleration, a drogue chute will deploy, followed by deployment of the main parachute(s). The maximum deceleration during this phase of reentry will be 3-g's sustained over a period not to exceed 60 seconds. The impulsive impact shock of a Spaceplane hard landing will not exceed (TBD) lb_f.

4.1.6 Detectability and Survivability

To protect against detection, the leakage and/or venting of expendables from Spaceplane shall be limited to not more than (TBD) cc per hour. In addition, the Spaceplane wall construction shall offer adequate crew protection from high temperatures due to solar and aerodynamic heating, as well as radiation from the Van Allen Belt. Vehicle hardening against radiation from a nuclear detonation or laser weapons is not a specific requirement in this study.

4.1.7 Alternate Missions

The first alternate mission to the DRM is a 24-hour mission with an excursion to Geosynchronous Orbit (GEO). EVA may be a requirement on this alternate mission. Other alternate missions to be considered are missions of up to 7 days duration in both LEO and GEO with EVA performed. The EC/LSS for these extended-duration missions may be different from that defined for the DRM.

4.2 VEHICLE REQUIREMENTS

Introduction

This section defines Spaceplane EC/LSS requirements in support of the Spaceplane DRM. The various interfaces between the Spaceplane vehicle and the pilot are shown in general form in figure 4-2. A useful way to view Figure 4-2 is in terms of inputs to and outputs from the crewman. There are two modes of operation to consider: Intravehicular Activity (IVA) and Extravehicular Activity (EVA). In either mode of operation, the crewman requires a pressurized environment and basic consumable inputs of oxygen for breathing and water and food to meet metabolic requirements. The pilot generates metabolic by-product outputs in the form of moisture, carbon dioxide, heat, and various other liquid, gas, and solid waste products that must b accomodated by collector and storage systems. The moisture, carbon dioxide, gas contaminants (e.g., odors), and heat must be continuously removed from the pilot's atmosphere, with make-up oxygen provided to replace that which is consumed. Metabolic heat, plus that generated from electrical and propulsion systems, as well as solar and aerodynamic wall heating that would raise the pilot's environmental temperature, need to be controlled through heat transport and rejection.

A couch/restraint system which interfaces with the space-suited pilot is required for comfort and for protection during launch and flight maneuvers. Spaceplane controls and displays, designed to reflect human factors, are required to provide the pilot with Spaceplane subsystem and mission operational status and the capability to effect control of systems in response to displayed data.

4.2.1 Thermal Control

Thermal protection shall be provided to the pilot to insulate against the temperature extremes of reentry and solar heating and subcritical cooling of space. Also, provision must be provided to remove metabolic and electronics heat loads to maintain pilot comfort and allow electronics to operate within design limits.

4.2.2 Atmosphere Revitalization

Spaceplane systems shall provide the pilot environmental pressurization and an oxygen-enriched atmosphere for breathing. Provision shall be included for atmosphere revitalization, including CO_2 removal, oxygen make up, trace contaminant gas removal, and humidity control.

4.2.3 Radiation Protection

Radiation protection shall be provided for the pilot so that safe limits shall not be exceeded. This protection must be adequate for the DMR as well as alternate missions, including those to GEO. No specific requirement is imposed for protection from nuclear detonation or from laser weapons.

4.2.4 Accelerations

During synergistic plane changes, reentry, recovery system deployment, and landing, the pilot will be subjected to the effects of accelerations and shock loads and will therefore require protection.

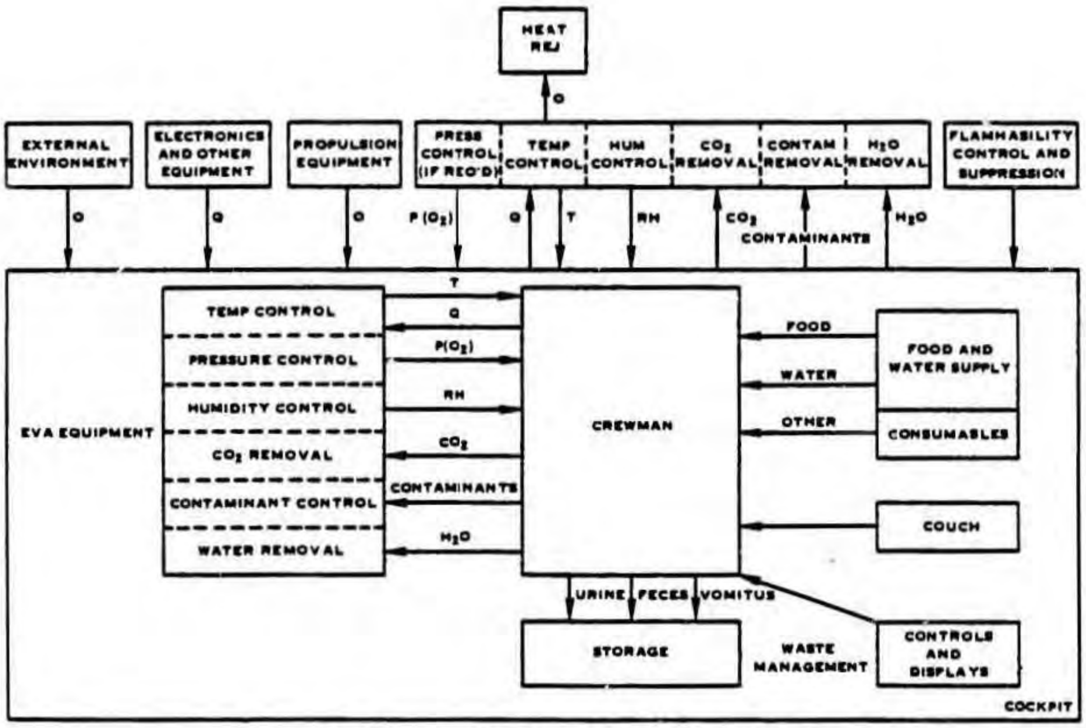


Figure 4-2 **SPACEPLANE EC/LSS RELATIONSHIP**

4.2.5 Flammability Control

Spaceplane systems shall employ fire-resistant materials and include fire suppression equipment, as required, to provide adequate fire safety margin.

4.2.6 Food and Water

Spaceplane systems shall provide metabolic consumables of water and food to the pilot. The total quantity oif consumables will be dependent on mission duration.

4.2.7 Waste Management

Provision shall be included for pilot waste management, including the collection of urine, feces, and vomitus. These systems shall provide for positive containment and limit contamination of the atmosphere and equipment.

4.2.8 Displays and Controls

Provision for display of Spaceplane subsystem and mission operational data will be included. Controls will be provided for the pilot to perform designated functions. Appropriate pilot controls and interfaces with all Spaceplane subsystems will be designed with appropriate human engineering.

4.2.9 Integration

Spaceplane subsystems will be designed to minimize overall Spaceplane weight and volume. This design will require subsystem configuration flexibility to permit optimal integration into Spaceplane.

4.2.10 EVA Provision

Spaceplane shall provide equipment necessary to support EVA. Adequate radiation protection will be provided for EVA to be conducted within the Van Allen Belts.

4.2.11 Illumination

During periods in full sunlight, solar filtering will be required to protect the pilot. Artificial lighting will be required to conduct IVA and EVA on the dark side of orbits.

4.3 SYSTEMS DEFINITION AND DESIGN SPECIFICATION

Spaceplane systems will be required to satisfy the various EC/LSS-related requirements outlined in the previous sections. The following sections define and present design specifications for various systems comprising the spaceplane EC/LSS. Critical systems are to satisfy a fail operational/fail-safe reliability criteria.

4.3.1 Couch

The couch will provide suitable interfaces to accommodate/restrain an EMU-equipped pilot, including the Portable Life Support system (PLSS) backpack. The couch shall be adjustable to permit EVA egress/ingress and viewing during space docking operations. The couch shall incorporate an energy-absorption device to protect the pilot from landing impact shocks (TBD). The couch shall be adjustable for different-size crew members, be lightweight, require minimal volume, and be constructed of fire-resistant materials.

4.3.2 Displays and Controls

The displays and control subsystem consists of the necessary sensors and instrumentation, with appropriate signal conditioning, to provide control and monitor of the EC/LSS system during all modes of operation.

A computer-based controller, in conjunction with a digital display, is envisioned to control and monitor all EC/LSS hardware and instrumentation in the Spaceplane. This type of equipment is compact, efficient, and state-of-the-art.

All Spaceplane displays and controls shall be designed to be operable and clearly visible by an EMU-equipped pilot. The equipment shall feature lightweight and compact designs. Illumination and solar filtering shall be provided in Spaceplane so that displays can be viewed independently of ambient illumination.

4.3.3 Waste Management

Because the pilot is in the EMU throughout all mission phases, waste management systems must be integrated into the EMU. A urine collection device shall be provided to collect and store 2,000 cc of urine. Collection devices shall be included for collecting (TBD) gm of solid fecal waste and (TBD) gm of vomitus matter. These devices shall be designed so that contamination of the atmosphere shall be limited to gases and water vapor not exceeding (TBD) cc total in a 24-hour period.

4.3.4 Support Equipment

Provision shall be included in the design of EC/LSS equipment for post-mission servicing and maintenance. The Spaceplane shall be designed so that total EC/LSS service time shall not exceed (TBD) hours. Repair or replacement of any major EC/LSS sybsystem/component shall take no longer than (TBD) hours.

Support equipment for post-mission checkout, recharging, and cleaning of on-board subsystems will be necessary to assure rapid vehicle turnaround and proper functioning of all on-board EC/LSS equipment. This equipment will provide the capability to:

- Recharge the oxygen, water, and ammonia tanks
- Replace LiOH canisters
- Check out electronic circuits and components
- Replace/remove waste collection devices
- Clean the cabin and equipment
- Check and recharge coolant loops
- Replenish food and drinking water

The equipment includes both ground-based and launch-vehicle support equipment as may be required.

4.3.5 Flammability Control

Proper fire control will be achieved through selection of proper materials and hardware designs within the vehicle to preclude the possibility of fire. Positioning of subsystem equipment involving high-pressure oxygen and propellants will be given special attention.

4.3.6 Food and Water Supply

Spaceplane shall provide for storing and dispensing food and water to the pilot for a 24-hour mission. Provision for positive food and water containment, eliminating contamination of the environment, shall be made.

A diet adequate to sustain the pilot throughout the Spaceplane mission should provide approximately 3,000 to 4,000 kilo-calories of energy per day, with adequate quantities of proteins, fats, carbohydrates, minerals, and vitamins. Exact caloric requirements for the pilot should be based on the actual body weight, age, height, and level of activity for the specific mission.

The diet should provide constituents as follows:

	Kilocalories	Calories %
Protein	420	12
Fat	1,050	30
Carbohydrates	2,030	58
	3,500	100

Vitamins and minerals will be supplied by food and dietary supplements.

In addition to the physiological requirements, the diet must be acceptable to the pilot and easy to consume.

Because the pilot is suited throughout the mission, the basic diet types will consist of dried or semi-dried solids and liquid items that are easily stored within the EMU.

4.3.7 Air Revitalization

A Spaceplane system shall be provided for maintaining an oxygen vent loop at a pressure of 8 $\pm$ 0.025 psia. The recirculated ventilating oxygen entering the crewmember environment shall be within the limits of 40F to 90F, except there shall not be condensation of free water or helmet fogging, and the dewpoint shall not exceed 65F maximum at the suit inlet. Absorbed moisture shall be removed from the ventilation flow by condensation.

Control of the level of CO_2 and particulate and trace contaminants shall be provided. The CO_2 partial pressure shall be maintained at less than 7.6 mm Hg for metabolic rates up to and inlcuding 1,600 Btu/Hr for one hour and less than 15 mm Hg for metabolic rates up to 2,000 Btu/Hr for 15 minutes. Particulate matter in the ventilating gas shall be limited to 0.1 milligrams per cubic meter. The system shall be designed to remove the trace contaminants generated by the crewmember at a rate that will maintain the concentration of the contaminants below the maximum allowable concentration presented in Table 4-I.

4.3.8 Thermal Control

A thermal control system shall be provided for rejecting the heat loads specified in Table 4-II. The thermal control system shall use a coolant that is compatible with EMU materials and is non-toxic. The system shall be gravity independent and provide adequate cooling to the pilot and Spaceplane equipment throughout all mission phases.

4.3.9 EVA Equipment

Equipment shall be provided to support EVA. This equipment shall consist of an EMU that includes a spacesuit and self-contained, portable life support system (PLSS) providing for a 7-hour nominal EVA. Visoring shall be provided for protection from high-intensity solar illumination, with provision for lights for dark-side activity. The portable life support system backpack shall be detachable for 1-G operations.

The PLSS shall satisfy the same requirements of the vehicle EC/LSS, with the crewmember working at an average metabolic rate of 1,000 Btu per hour for a maximum duration of seven (7) hours and six (6) hours minimum under the worst-case solar exposure.

The PLSS shall store and deliver oxygen for metabolic consumption, system leakage, and water tank pressurization.

The PLSS shall provide atmosphere revitalization and maintain the proper spacesuit pressure level during EVA. The EMU shall integrate food, drinking water, and waste management systems.

TABLE 4-I
TRACE CONTAMINANTS PRODUCED BY MAN

Compound	Maximum Allowable Concentration PPM	Biological gm/man day
Ammonia	25	0.25
Methane	1,000	0.047
Acetaldehyde	10	0.000083
Acetone	100	0.00013
Ethyl Alcohol	17	0.004
Methyl Alcohol	13	0.0014
n-Butyl Alcohol	3	0.0013
Methyl Mercaptan	1	0.00083
Hydrogen Sulfide	1	0.000075

TABLE 4-II
SPACEPLANE HEAT LOADS

Source	Average Heat Load Btu/Hr	Peak Heat Load Btu/Hr
Avionics & Communications	1,100	1,100
Life Support Equipment	600	600
APUs and Electrical Power	1,000	1,000
Payload	300	300
Environmental Heating	300	>300
Metabolic	700	1,000
TOTAL	4,000	11,300

4.4 TRADE STUDIES

Trade studies were conducted to determine appropriate EC/LSS subsystem choices for Spaceplane. The results of these trade studies, presented in the following sections, define for Spaceplane:

- heat transport
- heat rejection
- cabin pressure
- extravehicular mobility unit (EMU).

4.4.1 Heat Transport

Two options considered viable for heat transport in the Spaceplane vehicle for equipment cooling and removal of metabolic and environmental loads were use of a circulating coolant or heat pipes. Trade-off criteria used included cooling capability, volume and weight impact, and performance limitations. The advantages and disadvantages of each of these two options are presented in Figures 4-3 and 4-4. Heat pipes were rejected from further consideration, primarily because of performance degradation when subjected to varying g-loads, the requirement for circulation of coolant through the spacesuit liquid cooling garment (LCG), and difficulties in interfacing heat pipes with the cold plates that are required for equipment cooling. Several coolants were considered for use in the circulating coolant option, including water, water/propylene/glycol, and Freons. Water was selected as the coolant for the circulating coolant heat transport loop because of its compatibility with existing EMU components, its excellent thermal characteristics, its weight advantage, and its lack of toxicity. Characteristics of the fluids screened are shown in Table 4-III. The choice of water, however, will necessitate protecting coolant loop components from freezing.

TABLE 4-III
FLUID SCREENING

Fluids are in order of decreasing desirability at 60F

No.	Fluid	Freeze Temp F	Lb/Ft^3	Btu/Lb-F	C_p 3
1	Water	32	62.4	1	3894
2	65% Propylene Glycol - 35% H_2O	-65	76.5	0.735	1809
3	F-21	-211	86.7	0.26	132.1
4	F-114	-137	92.7	0.24	118.7
5	F-12	-252	83.9	0.24	97.3
6	F-11	-168	93.5	0.213	84.5

The recommended fluid of choice is water for the following reasons:

Recommendation:

Circulating Water

ADVANTAGES (ASSUMING WATER)

- COMPATIBLE WITH EXISTING SUIT COMPONENTS
- GOOD THERMAL CHARACTERISTICS
- NO TOXICITY
- NO PERFORMANCE LIMITATIONS
- GRAVITY INDEPENDENT

DISADVANTAGES

- REQUIRES REDUNDANT CIRCUIT (ADDITIONAL WEIGHT)
- REQUIRES GOOD CONTACT CONDUCTANCE BETWEEN COLD PLATE AND BLACK BOXES

Figure 4-3 CIRCULATING COOLANT

ADVANTAGES

- NEAR ISOTHERMAL OPERATION/SELF REGULATING
- GOOD "O-G" OPERATION
- REDUCED WEIGHT/VOLUME (< 10 LBS) DUE TO REDUCED TUBING RUNS
- REDUCED PUMP POWER (< 8 WATTS)

DISADVANTAGES

- REQUIRES GOOD CONTACT CONDUCTANCE BETWEEN HEAT PIPES AND BLACK BOXES AND HEAT REJECTION DEVICE
- STILL REQUIRES CIRCULATING COOLANT FOR SUIT LCG AND VENT LOOPS
- PERFORMANCE SIGNIFICANTLY DEGRADED BY HIGH "G" LOADINGS IN VARIOUS PLANES

Figure 4-4 **HEAT PIPES**

Reasons:

- Water offers good thermal properties
- Water is non-toxic and compatible with EMU materials
- Not gravity dependent
- Small weight/power penalty
- EMU requires coolant circulation

4.4.2 Heat Rejection

Figure 4-5 summarizes the trade study performed to select the heat-rejection device to be utilized by Spaceplane. An estimated average heat-rejection rate of 4,000 Btu/hr for Spaceplane was assumed generated from the following sources:

Source	Heat Load (Btu/Hr)
Metabolic	700
Life Support Equipment	600
Avionics and Communications	1,100
APU and Electrical Power Equipment	1,000
Payload	300
Environmental Heating	300
TOTAL	4,000

The advantages and disadvantages of each of the heat rejection options considered, indicated in Figure 4-5, are summarized in Figures 4-6 through 4-10. Following is a discussion of each of the heat rejection options.

Use of a radiator was eliminated from further consideration for the 24-hour duration Spaceplane because of the volume and weight impact (7.5 ft^3 with 89 ft^2 surface area and over 100 lbs). Also, the radiator could not operate in the atmosphere during a synergistic plane change and would impact mission flexibility by forcing certain orientations in space for rejection of heat. Also of importance is the complexity added by a radiator deployment/retraction mechanism and the increased detectability posed by the radiator's highly reflective surface. As mission length increases, however, the advantage of a radiator in reducing expendables could outweigh the disadvantes (i.e., 41bs/hr of water evaporated for heat rejection).

KEY CONSIDERATIONS

- SIZE AND WEIGHT
- PERFORMANCE LIMITATIONS
- MISSION IMPACT
- CONTROL REQUIREMENTS

OPTIONS

- RADIATOR
- THERMAL CAPACITOR
- EVAPORATION:
  - SPRAY FLASH EVAPORATION
  - SPRAY BOILER
  - STACKED EVAPORATOR

ASSUMPTIONS

- REQUIRED HEAT REJECTION OF 4000 BTU/HR FOR 24 HOUR MISSION
- STEAM EXHAUST ACCEPTABLE FOR DETECTABILITY

Figure 4-5 THERMAL CONTROL SYSTEM HEAT REJECTION DEVICE

ADVANTAGES

- UTILIZE DEEP SPACE SINK TEMPERATURES FOR HEAT REJECTION
- INDEPENDENT OF MISSION LENGTH
- REDUCED AMOUNT OF CONSUMABLES

DISADVANTAGES

- LARGE VOLUME (APPROX. 7.5 FT^3 WITH 89 FT^2 SURFACE AREA) AND WEIGHT (APPROX. 133 LBS)
- WILL NOT OPERATE AT LOW ALTITUDES
- INSUFFICIENT VEHICLE SURFACE AREA
- IMPACTS MISSION PROFILE FOR PROPER HEAT REJECTION ORIENTATION
- REQUIRES DEPLOYMENT/RETRACTION MECHANISM
- HIGHLY REFLECTIVE SURFACE PROVIDES MEANS OF TRACKING
- REQUIRES ADDITIONAL COOLING DEVICE DURING REENTRY/SYNERGISTIC PLANE CHANGE

Figure 4-6 **RADIATOR**

ADVANTAGES

- COOLING CAPACITY LIMITED TO CAPACITY OF STORAGE DEVICE
- DEVICE IS SELF-CONTROLLING
- DOES NOT REQUIRE ADDITIONAL COOLING DEVICES
- SOME HEAT SINKING AVAILABLE TO FUEL TANKS

DISADVANTAGES

- REQUIRES LARGE WEIGHT (595 LBS/DAY) AND VOLUME (9.5 FT^3/DAY) OF HEAT STORAGE

Figure 4-7 THERMAL CAPACITOR

ADVANTAGES

- ALL HEAT REJECTION PROVIDED BY SINGLE DEVICE
- NO MISSION PROFILE EFFECTS
- GOOD CONTROL ON COOLANT BY VARYING SPRAY FREQUENCY

DISADVANTAGES

- WEIGHT (15 LB) AND VOLUME (240 IN^3)
- WASTED VOLUME DUE TO SPRAY NOZZLE PATTERN
- LARGE AMOUNT OF CONSUMABLES (88 LB OF H_2O, 8 LB OF NH_3 PER DAY)
- NEED EXHAUST PRESSURE CONTROL TO PREVENT COOLANT FREEZING WHEN EVAPORATING NH_3 DURING REENTRY

Figure 4-8A. SPRAY FLASH EVAPORATION (ORBITAL WATER EVAPORATION - REENTRY AMMONIA EVAPORATION)

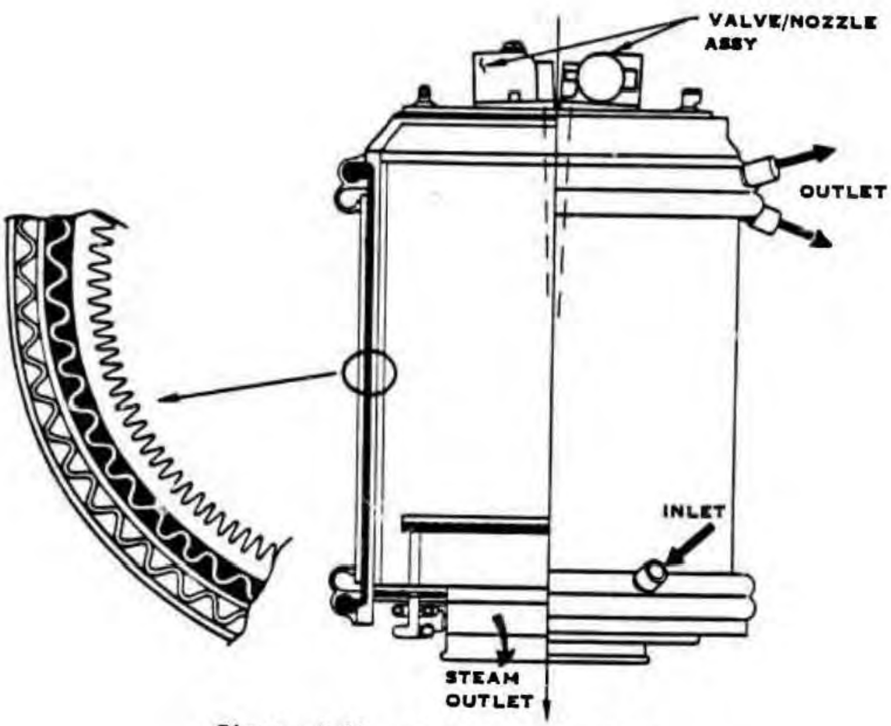


Figure 4-8B **FLASH EVAPORATOR**

ADVANTAGES

- ALL HEAT REJECTION PROVIDED BY SINGLE DEVICE
- NO MISSION PROFILE IMPACT

DISADVANTAGES

- LARGER AMOUNT OF CONSUMABLES THAN FLASH EVAPORATOR DUE TO CARRYOVER IN "O" G ENVIRONMENTS
- NEED EXHAUST PRESSURE CONTROL TO PREVENT CIRCULATING COOLANT FROM FREEZING WHEN EVAPORATING NH_3 DURING REENTRY
- MIXED FLUIDS IN DEVICE WHEN CHANGING TO NH_3 DURING REENTRY

Figure 4-9A. SPRAY BOILER (SPRAY BOILING ON ORBIT, POOL BOILING DURING REENTRY)

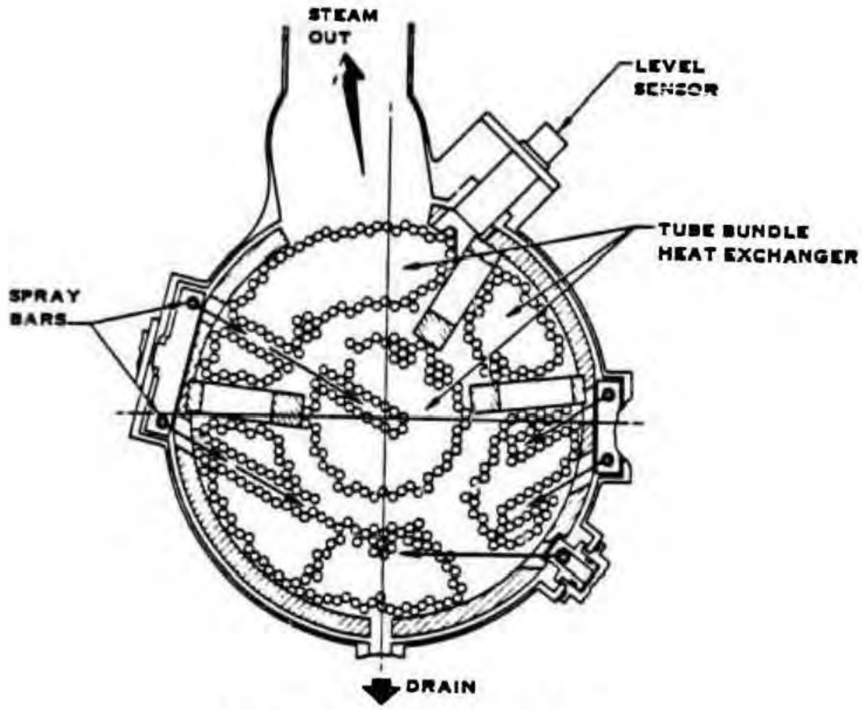


Figure 4-9B. **SPRAY BOILER**

ADVANTAGES

- ALL HEAT REJECTION PROVIDED BY SINGLE DEVICE
- DEAD PASSAGE ELIMINATES EXHAUST PRESSURE CONTROL AND NH_3 LEAKAGE
- LIGHTWEIGHT DEVICE (APPROXIMATELY 10 LBS)
- LITTLE WASTED VOLUME (APPROX. 90 IN^3 VS. 240 IN^3)
- SAME AMOUNT OF CONSUMABLES AS FOR SPRAY FLASH EVAPORATION
- WICK FEED CONTROL SIMILAR TO SPRAY FREQUENCY CONTROL
- NO MISSION PROFILE IMPACT

DISADVANTAGES

- DEVELOPMENT REQUIRED

Figure 4-10A. STACKED EVAPORATOR

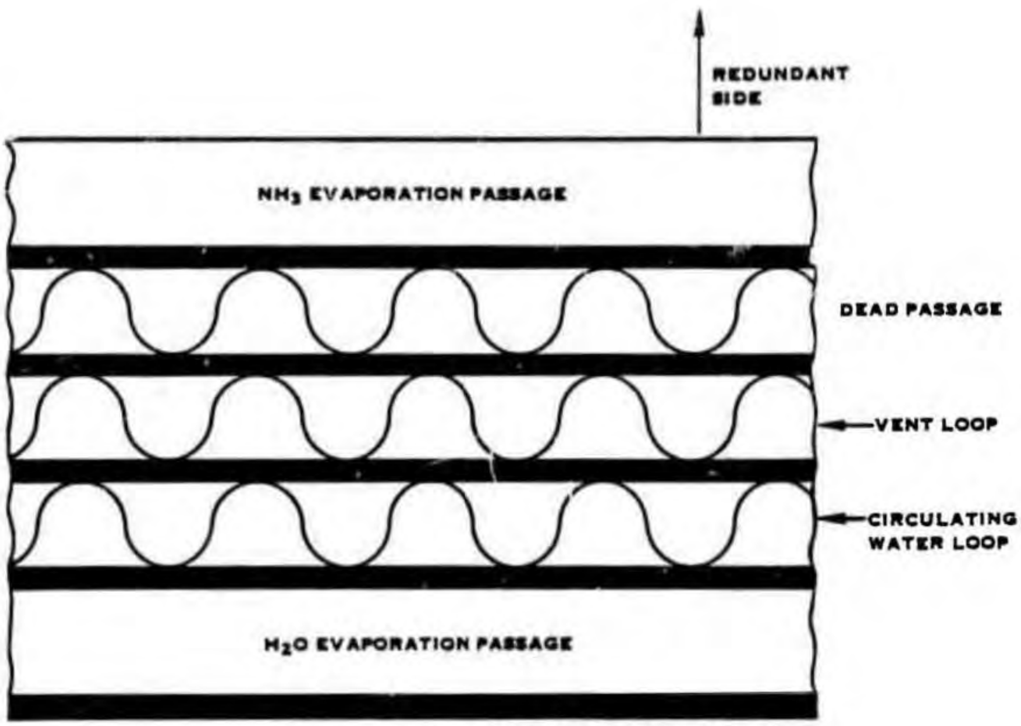


Figure 4-10B. STACKED EVAPORATOR

Similarly, a thermal capacitor phase change device, using heat of fusion instead of heat of vaporization, was rejected from further consideration due to the large weight of phase change material required to reject 96,000 Btu/day. A thermal capacitor in the form of on-board propellant was considered as a possible approach. Although there would be no tank vapor pressure problems with this approach, there would be only 18,000 Btu total cooling available from the propellant because of a coolant-supply temperature requirement of approximately 50F. This approach would save approximately 17 lbs of evaporative cooling water, but the total weight saving would be less than 8 lbs. because of the additional ducting, interface heat exchangers, dual interface loop pumps, and cooling loop bypasses required. This approach is therefore not recommended because of added system complexity and a mission heat rejection dependence based on propellant consumption.

Evaporative cooling was selected as the means for Spaceplane heat rejection. Due to its high heat of vaporization and density, water was selected as the evaporant above 100,000 feet and on-orbit; however, because of its lower saturation pressure, ammonia was selected at altitudes below 100,000 feet. Devices evaluated in this trade study were similar to the Space Shuttle Flash Evaporator Subsystem (FES) and Water Spray Boiler (WSB) in addition to a derivative concept from the FES and WSB called a "stacked evaporator," shown in Figure 4-10B. The stacked evaporator was selected for Spaceplane application for the reasons indicated in Figure 4-10A. The FES was rejected because a cylindrical shape is required to accommodate the required spray nozzle pattern and requires excessive volume as compared with other devices. There would also be a need to change over from H_2O to NH_3 in the same core during descent at about 100,000 ft. This changeover would require exhaust pressure control during descent to prevent NH_3 evaporation temperature from becoming too low, thereby freezing the coolant loop. The WSB was rejected for spaceplane application for the latter two reasons, plus the fact that its higher degree of carryover in a zero-g environment requires additional expendables.

The recommendation is therefore:

Recommendation:

Stacked Evaporator

Reasons:

- Single heat rejection device
- Low device weight and volume
- No carryover
- No loss in mission flexibility
- Built-in coolant freezing protection

4.4.3 Cabin Pressure

A study was conducted, as indicated in Figure 4-11, to evaluate the trade-off of an unpressurized cockpit versus a cockpit pressurized with either oxygen,

KEY CONSIDERATIONS

- VEHICLE SIZE/WEIGHT
- COCKPIT LEAKAGE
- FLAMMABILITY CONTROL
- MISSION PROFILE
- CABIN PRESSURIZATION/VENTING

OPTIONS

- UNPRESSURIZED COCKPIT
- PRESSURIZED COCKPIT
  - O_2 ATMOSPHERE
  - O_2/N_2 ATMOSPHERE
  - INERT ATMOSPHERE

Figure 4-11. PRESSURIZED VS. UNPRESSURIZED COCKPIT an oxygen/nitrogen mixture, or an inert gas. Key issues considered in the study included the impact on volume and weight of such parameters as vehicle wall thickness, seals, and hatch mechanism, stored gas for leakage makeup or cabin repressurization and pilot volume requirements for EMU glove and helmet storage. Also considered was the possibility of fire in an oxygen atmosphere cockpit and the potential impact on the mission caused by leakage leading to crewman incapacitation. The advantages and disadvantages of each option are presented in Figures 4-12 through 4-14. A key issue in this trade study was whether the crewman could function for the entire 24-hour full mission while fully suited. It was concluded that, with improved waste management (i.e., feces, urine, and vomitus collection), a crewman could perform the one-day mission fully suited with helmet and gloves. Although it would be desirable to allow the crewman the additional freedom provided by doffing gloves and helmet, an unpressurized cabin was recommended as the baseline configuration for Spaceplane for the following reasons:

Recommendation:

Unpressurized Cockpit

Reasons:

- Reduced vehicle size/weight
- Less stringent sealing requirements, resulting in reduced fabrication and maintenance cost
- Minimal fire hazard
- No mission impacts due to cabin leak

4.4.4 Spacesuit

The EC/LSS selected for the Spaceplane includes an unpressurized vehicle cabin, with the pilot in a spacesuit throughout all mission phases. The spacesuit and life support system selected is an 8 psi derivative of the Shuttle EMU, with self-contained portable life support system (PLSS). For IVA, the crewman would be supported by vehicle life support systems and consumables via umbilicals. To conduct EVA, the pilot would disconnect the EC/LSS umbilicals at the EMU suit and activate the PLSS. Because previous EVA experience has shown that management of life support umbilicals requires more than one crewman, the EMU with self-contained PLSS for EVA was selected.

The EC/LSS candidate spacesuit options considered viable for Spaceplane, together with principal discriminators considered in the trade study selection process, are indicated in Figure 4-15. It was apparent from considering candidate mission profiles that the existing Shuttle EMU, although satisfying most EC/LSS and EVA requirements, would require modification. A modified 4 psi EMU, which incorporated minimum modifications to the existing Shuttle EMU to meet mission requirements, was then considered. Finally, an 8 psi EMU that incorporates several additional features to provide for mission quick response and flexibility was defined. A comparison of the three EMU configurations studied is made in Figure 4-16. The results of the comparison study were

ADVANTAGES

- REDUCED WEIGHT (~20 LBS FOR THINNER WALLS)
- REDUCED VOLUME (NO HELMET/GLOVE DONNING/STORAGE)
- REDUCED LEAKAGE
- NO FIRE CONTROL IN VACUUM
- NO MISSION IMPACT DUE TO SIGNIFICANT LEAK

DISADVANTAGES

- PILOT SUITED AT ALL TIMES
- REQUIRES IMPROVED SOLID WASTE MANAGEMENT
- REQUIRES VEHICLE REPRESSURIZATION

Figure 4-12 UNPRESSURIZED COCKPIT

ADVANTAGES

- IMPROVED PILOT COMFORT, IF UNSUITED
- LESS STRINGENT SOLID WASTE MANAGEMENT INTERFACE (I.E., HELMET OFF)

DISADVANTAGES

- INCREASED WEIGHT (~ 20 LBS FOR WALLS)
- INCREASED VOLUME (HELMET/GLOVE DONNING/STORAGE)
- REQUIRES VEHICLE SEALING/EMERGENCY BREATHING SUPPLY
- HIGH FIRE HAZARD IN O_2 ENVIRONMENT
- POTENTIAL LEAKAGE IMPACT ON MISSION

Figure 4-13. PRESSURIZED COCKPIT (O_2 OR O_2/N_2 ATMOSPHERE)

ADVANTAGES

- AIDS CREW MOBILITY WHEN SUITED
- LOW FLAMMABILITY
- NO ORBITAL MISSION IMPACT DUE TO LEAKAGE

DISADVANTAGES

- CREW MUST BE SUITED
- REQUIRES IMPROVED SOLID WASTE MANAGEMENT
- INCREASED WEIGHT DUE TO THICKER VEHICLE WALLS
- REQUIRES IMPROVED VEHICLE SEALING

Figure 4-14. PRESSURIZED COCKPIT (INERT GAS ATMOSPHERE)

OPTIONS
- SHUTTLE EMU
- MODIFIED SHUTTLE EMU
- NEW SPACE SUIT SYSTEM

DISCRIMINATORS
- WEIGHT/VOLUME
- MOBILITY
- PREBREATHE REQUIREMENT
- CREW SAFETY
- MISSION IMPACT

ASSUMPTIONS
- UNPRESSURIZED CABIN
- EVA CAPABILITY REQUIRED
- PLSS PROVIDES EC/LSS BACKUP

Figure 4-15. **SPACE SUIT CONSIDERATIONS**

FEATURE	SHUTTLE EMU	MODIFIED 27.6 KPA (4 PSI) EMU	55.2 KPA (8 PSI) EMU
TOTAL PRESSURE	26.3 KPA (4.1 PSI)	28.3 KPA (4.1 PSI)	55.2 KPA (8 PSI)
PREBREATHE	3 HOURS O_2	3 HOURS O_2	NONE
GLOVES AND JOINTS	EXISTING	EXISTING	NEW GLOVES AND JOINTS
UMBILICAL PROVISION	SCU – NOT ADEQUATE	SCU – MODIFY VENT AND WATER LOOPS IN PLSS	YES
EVA CAPABILITY	7 HOURS NOMINAL	7 HOURS NOMINAL	7 HOURS NOMINAL
UCD CAPACITY	950CC	2000CC	2000CC
SOLID WASTE MANAGEMENT	NONE	SOLID COLLECTORS NEW HELMET REQUIRED	YES
WATER AND FOOD	595G (21 OZ) WATER 227G (8 OZ) FOOD STICK	1021G (36 OZ) WATER 454G (16 OZ) FOOD STICK	1021G (36 OZ) WATER 454G (16 OZ) FOOD STICK
1-G MOBILITY	LIMITED	LIMITED	GOOD WITH DETACHABLE PLSS
DRY WEIGHT	109 KG (240 LB)	109 KG (240 LB)	113 KG (250 LB)
CONSUMABLES	0.55 KG (1.22 LB) OXYGEN 4.54 KG (10 LB) WATER 4.21 KG (9.3 LB) BATTERY 1.13 KG (2.5 LB) LIOH	SAME	SAME
COUCH INTERFACE	PLSS ACCOMMODATED BY COUCH	PLSS INTEGRAL PART OF UPPER COUCH	PLSS INTEGRAL PART OF UPPER COUCH AND DETACHABLE
STRUCTURE	NOT DESIGNED FOR 3G'S WITH MAN	PLSS STRUCTURAL MODIFICATIONS	DESIGNED FOR 3G'S WITH MAN
COMFORT	NOT DESIGNED FOR 3G'S WITH MAN	HUT PADDING AND NECK RING CUSHIONING	MODIFIED UPPER TORSO

Figure 4-16. **CANDIDATE EMU COMPARISONS**

that the existing Shuttle EMU would not satisfy Spaceplane mission requirements without modifications. It was concluded further that the EMU for Spaceplane should be the 8 psi EMU, because this choice of spacesuit operating pressure eliminates the requirement for oxygen prebreathe prior to going EVA, which is required for Shuttle launch. For a ground-expendable booster launch, with a pressurized Spaceplane cabin and a 4 psi EMU serving as backup, prebreathing would still be required, because a cabin leak early in the mission could subject the pilot to the "bends." The choice of 8 psi pressure for the EMU is based on results of recent prebreathe studies that indicate that a crewmember acclimated to a 14.7 psi ambient atmosphere, such as the Space shuttle, could immediately go EVA at a suit pressure of 8 psi (pure oxygen) without suffering denitrogenization bend problems. Presently, going EVA in the 4 psi EMU spacesuit system requires approximately a 3-hour, pure oxygen, prebreathe. The elimination of the prebreathing requirement is consistent with the quick response requirement of military missions. The most significant EMU development impact of elevated suit pressure is the need for new gloves and suit joints to maintain crew mobility and hand dexterity for performing both IVA and EVA tasks. NASA is expected to develop a derivative 8 psi version of the Shuttle EMU, and this technology would be available for use in the Spaceplane program.

The recommendation for Spaceplane is therefore:

Recommendation:

8 psi EMU

Reasons:

- Elimination of prebreathe
- Improved 1-G mobility
- Improved waste management
- Improved comfort under G-loads

4.4.1.1 Preliminary Design

The EC/LSS equipment and consummables required to integrate a crewman into Spaceplane and to support the crewman in cislunar space throughout the 24-hour DRM consists of the following major components:

- EC/LSS Subsystem

  Liquid Loop
  Vent Loop
  Consumables

- Extravehicular Mobility Unit (EMU)

  Portable Life Support System (PLSS)
  Spacesuit Assembly
  Food and Water Supply
  Waste Management Collectors
  Consumables

- Couch Restraint System
- Displays and Controls
- Support Equipment (Ground, Flight)

The preliminary design of each of these EC/LSS components is discussed below. A detailed requirements specification for the EC/LSS is presented in Appendix A. Wherever possible, use was made of existing and proven space hardware to reduce Spaceplane development time and overall program cost.

4.5 PRELIMINARY DESIGN

Before proceeding with a discussion of individual EC/LSS components, several basic EC/LSS design choices are discussed. A clarification of these choices provides insight into the overall design of the Spaceplane EC/LSS.

4.5.1 Reliability Criteria

The Spaceplane EC/LSS has been designed to meet a fail operational/fail-safe reliability criteria. Analysis has shown that the difference in total systems weight for this choice over that of a fail-safe system is approximately 35 lbs. This additional weight is the result of adding redundant pumps, fans, valves, and regulators. The added margin of mission and pilot safety is considered to outweigh the disadvantages of a slightly heavier system. Reliability/redundancy issues are presented in Table 4-IV.

Table 4-IV

RELIABILITY/REDUNDANCY ISSUES

RELIABILITY CRITERIA	HARDWARE REQUIREMENT	MISSION IMPACT
FAIL OPERATIONAL/ FAIL SAFE	DUAL REDUNDANCY FOR NON-CRITICAL COMPONENT/FUNCTION TRIPLE REDUNDANCY FOR CRITICAL COMPONENT/ FUNCTION	MISSION ABORT AFTER 2ND FAILURE OF A CRITICAL COMPONENT/FUNCTION
FAIL SAFE	DUAL REDUNDANCY FOR CRITICAL COMPONENT/ FUNCTION	MISSION ABORT AFTER 1ST FAILURE OF A CRITICAL COMPONENT/FUNCTION

4.5.2 Flammability Control

Flammability control for the Spaceplane EC/LSS is achieved through proper selection of materials and hardware designs within the cabin to preclude the possibility of fire. Fire requires three basic ingredients: a combustible fuel, oxygen to combine with the fuel in combustion, and an ignition source. To determine the potential of a fire occurring in Spaceplane, the three requirements were analyzed. The atmosphere contains oxygen; therefore, this fire ingredient cannot be totally eliminated. Ignition sources cannot be totally eliminated because electrical power is used in subsystems, instrumentation, lighting, and displays. The ignition-source hazard can be reduced, however, by the use of proper electrical circuit design, by use of circuit breakers, by selection of nonarcing electrical components, by isolation of as much electrical equipment as possible outside the cabin, and by selection of proper wiring. The approach to flammability control is primarily through the selection of non-flammable subsystem materials, provision for adequate equipment cooling, thermal and electrical insulation, and isolation of fuel/oxidizer bi-propellants. Special care is taken with the cabin vent loop, as well as the EMU itself, because this equipment operates in an essentially pure oxygen atmosphere at 8-22.7 psia.

Standard material selection and design standards will be adhered to for high-pressure oxygen systems. Special care will be taken to avoid introducing contamination particles during manufacture or service and to select oxygen-compatible, nonmetallic materials within the EC/LSS. Nonmetallic materials used in the EC/LSS could include coatings, adhesives, films, elastomers, gaskets and seals, thermal insulation, potting/encapsulations, lubricants, and greases. An activated carbon filter, which is part of the LiOH cartridge, will remove odors and harmful trace contaminants in the oxygen vent loop atmosphere. Finally, the cabin ventilation system could be used to vent any residual contaminants from the cabin ambient atmosphere should this become necessary. The basic approach to contamination control is summarized in Figure 4-17.

4.5.3 Radiation Protection

The EC/LSS for the Spaceplane must provide the crewman adequate radiation protection for a 24-hour mission, including an excursion to GEO. Radiation data have established the following regarding single, short-duration radiation exposures by man, measured in REM:

SHORT EXPOSURE DOSES

Result	Dose
Effects similar to standard X-Ray	1-5 REM
Statistical shortening of life	200 REM
Death within hours	500 REM

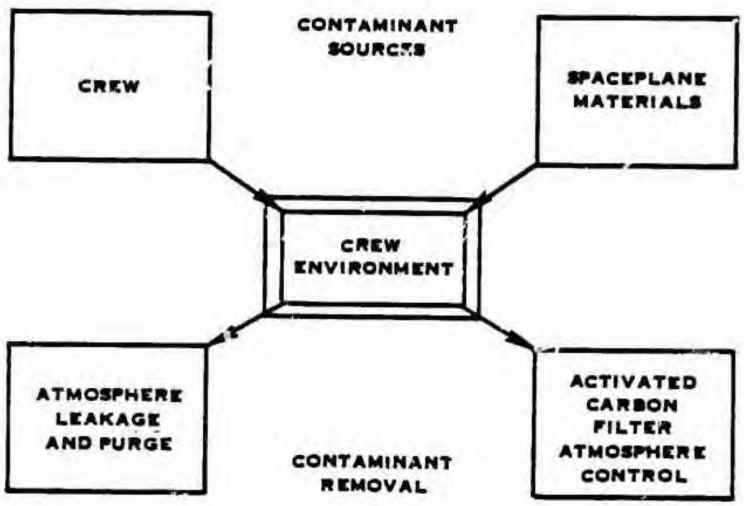


Figure 4-17. CONTAMINANT SOURCE AND CONTROL FOR SPACEPLANE

There are four sources of radiation to which the spaceplane pilot could be exposed:

- Solar Flares
- Galactic Cosmic Particles
- Trapped Solar Electrons & Protons (two Van Allen Belts)
- Unscheduled Radiation Exposures

Of these sources, only the Van Allen Belt radiation depicted in Figure 4-18 is predictable and was considered in designing Spaceplane for short-duration missions in earth orbit.

In the analysis for Spaceplane, it was assumed that the vehicle will have a skin wall providing the equivalent radiation shielding of 0.25 inches of aluminum and that the pilot is suited in the Shuttle EMU. Van Allen Belt radiation data indicate that, for the Spaceplane configuration, the peak, short-term, pilot radiation exposure experienced on a trip to GEO would be less than 5 REM and that the worst total round-trip radiation dosage experienced on a transfer orbit between LEO and GEO would be 10.5 REM. The radiation dosage at GEO altitude in the same vehicle would be approximately 1 REM/day. Therefore, if a 24-hour GEO mission is assumed with the 11-hour Hohman transfer orbit and 13 hours spent in GEO, the total dosage would be 10.5 Hohm + 13 (1/24) GEO = 12.4 REM. The NASA 30-day safe limits set for rediation exposure are:

NASA 30-DAY CUMULATIVE RADIATION EXPOSURE LIMITS

Organ	Dose (REM)
Eyes	37
Skin	65
Blood-forming organs	25

Therefore, for the 24-hour Spaceplane mission scenario considered, the cumulative exposure of 12.4 REM would not jeopardize the pilot. The blood-forming organs would have received approximately 50% of the recommended allowable 30-day dosage of radiation. Using the NASA standard, the pilot could fly one more such mission in the same month. Although further radiation analysis is required, preliminary results indicate that Spaceplane, with a skin thickness equivalent to 0.25 inches of aluminum, would offer the pilot the required 24-hour orbital radiation protection in orbits from LEO to GEO.

4.5.4 Computer Model

A computer model was constructed for thermal and EC/LSS design studies for the Spaceplane. The Spaceplane computer model is designed to compute optimum values of coolant flow and heat sink outlet temperature for an ECLS consisting of one air and one water loop, both cooled by an evaporator using water or ammonia evaporative cooling. The optimum values for flow and heat sink outlet temperature are those that result in the lowest total equivalent weight for the EC/LSS. Total equivalent weight is calculated as the sum of EC/LSS component fixed weights, power equivalent weight, and a system water loss

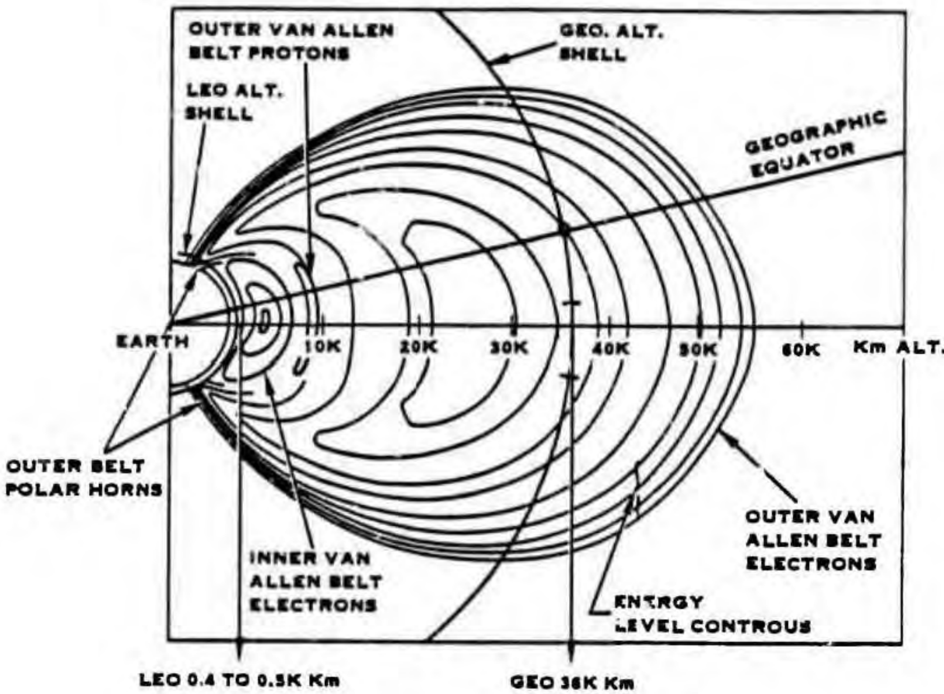


Figure 4-18. **VAN ALLEN BELTS**

penalty. Water loss penalty is reflective of the total system heat load and is the quantity of expendable water that must be carried for evaporative cooling.

The optimization logic begins with the lowest allowable heat sink outlet temperature and the minimum value for coolant flow, as determined by pilot liquid-cooling garment requirements. The program first iterates to determine optimum total equivalent weight at the minimum allowable heat sink outlet temperature. This value is saved, an incremental change is made in heat sink outlet temperature, and the flow iteration performed again at the new heat sink outlet temperature to determine a new value of total equivalent weight. This new value is compared to the saved value from the previous iteration. The program continues to iterate on temperature and flow until the lowest value of total equivalent weight is achieved.

The Spaceplane computer program can accept the heat loads of up to 50 electronics and payload components as inputs to the program. Inputs to the program are heat loads (watts), base areas (ft^2), and component base temperature limits (F). Required cold plate areas are calculated as a fixed factor (packing density) times the summation of base areas for all components. In the subroutine that evaluates cold plate performance, the base temperature of each component is calculated and checked against the input component temperature limit.

Once optimum values of heat sink outlet temperature and flow rate have been determined, a system schematic is printed, showing calculated performance criteria for the optimized system. A summary of the fixed weight calculations and total equivalent weights, along with component pressure drips, is also output by the program. Charts summarizing computer model logic, uses, input requirements, and program output are presented as Appendix B.

4.5.5 EC/LSS Subsystem

All Spaceplane EC/LSS interfaces are shown in Figure 4-19. In this section, the indicated EC/LSS Subsystem and its interfaces with the crewman and vehicle equipment will be discussed. All other equipment shown interfacing with the crewman will be discussed in subsequent sections.

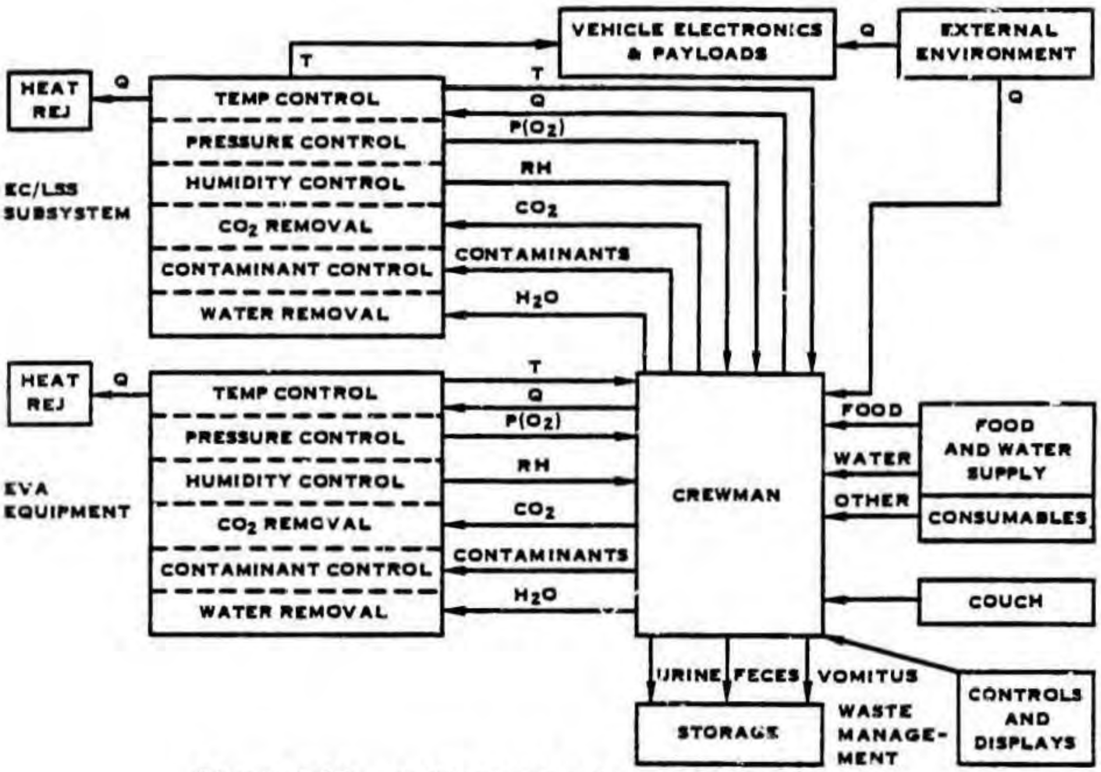


Figure 4-19. **SPACEPLANE EC/LSS INTERFACES**

4.5.5.1 EC/LSS Subsystem Specification

The EC/LSS subsystem for Spaceplane shall provide environmental and thermal control necessary to support a one-man crew and vehicle subsystems throughout all phases of a mission of 24-hour duration. The EC/LSS subsystem shall meet the following specifications:

Atmosphere Control

- Pressure
- Humidity
- Carbon Dioxide Removal
- Oxygen Makeup
- Trace Contaminant Removal
- Temperature

Thermal Control

- Ventilation Circuit Temperature
- Electronic Equipment Cooling
- Pilot Cooling

4.5.5.2 EC/LSS Subsystem Description

The EC/LSS subsystem selected for Spaceplane, shown schematically in Figure 4-20, exhibits the following major characteristics:

- Unpressurized cabin in space
- Cabin pressurization and ventilation provision
- Pilot spacesuited throughout mission
- 8-psi EMU
- EMU provides life support system backup
- Oxygen vent loop
- Water coolant loop
- 7-hour nominal EVA capability
- Evaporative water/ammonia heat rejection
- Fail-operational/fail-safe reliability criteria
- 100 percent consumables redundancy

The EC/LSS subsystem consists of four major components, which are discussed in the following sections.

Vent Loop

The oxygen ventilation circuit forms a closed loop with the pilot's pressure garment assembly and Liquid Cooling and Ventilation Garment (LCVG). Within this closed loop, the oxygen ventilation circuit circulates O_2 for breathing while removing CO_2, excess moisture, and trace contaminants. The oxygen ventilation circuit will operate within a pressure range established by the regulated O_2 control module, shown at the top of Figure 4-20. The fan, which is redundant, circulates O_2 through the entire circuit.

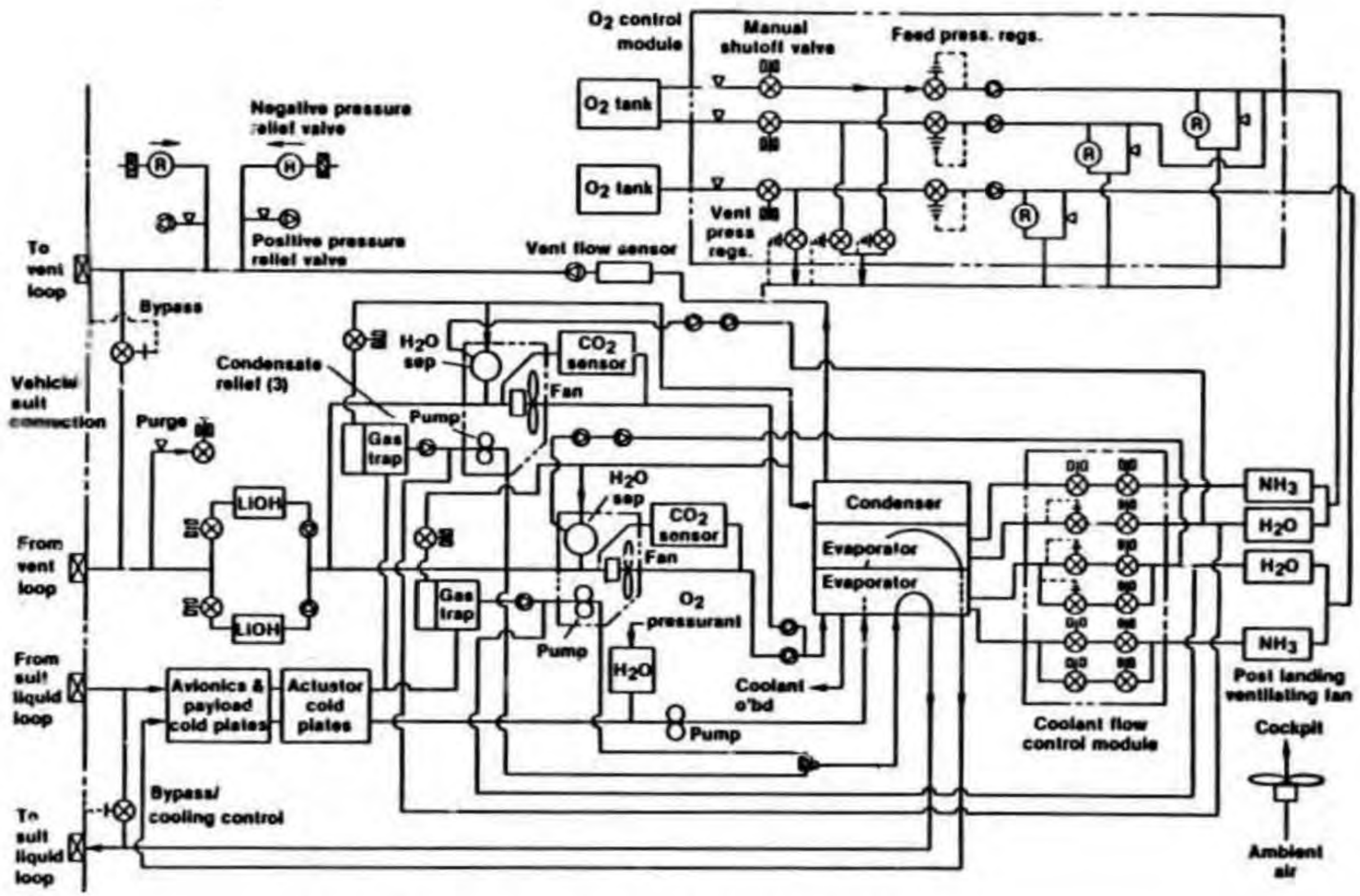


Figure 4-20. **SPACEPLANE EC/LSS SUBSYSTEM SCHEMATIC**

Moisture-laden O_2 is cooled to its dew point in the condenser section of the evaporator. The condensed moisture, along with a small quantity of O_2, is drawn off the condenser at the slurper, which acts as a boundary layer suction bleed, and fed to the separator, which centrifugally separates the condensed moisture. Condensate is then pumped to the feedwater circuit by a pitot tube in the separator. The resulting O_2, free of entrained moisture, is returned to the vent loop at the fan inlet. After leaving the evaporator, the cooled O_2 passes through the Vent Flow Sensor and Check Valve prior to return to the spacesuit.

In the spacesuit, O_2 flows into the helmet through the helmet duct, where it cools the head, defogs the visor, and picks up respired CO_2 and moisture. From there, O_2 flows over the body to the extremities, evaporating perspiration. Vent tubes in the LCVG pick up the O_2 and return it to the vent circuit through the umbilical closing the loop.

The Contaminant Control Cartridge containing LiOH removes CO_2, as well as trace contaminants from the gas stream. Outlet CO_2 concentration is monitored by the CO_2 sensor, which uses fan pressure rise to drive the gas-sampling stream within which a flow orifice limits sample-stream flow.

The Positive Pressure Relief Valve relieves O_2 ventilation circuit pressure should the O_2 pressure regulators fail open. The Negative Relief Valve opens during vehicle pressurization to permit vent circuit internal pressure to track rising vehicle pressure in the event the primary regulator does not maintain suit pressure to within 0.0 to 0.2 psid over vehicle pressure.

The Purge Valve is a manually actuated valve that passes sufficient purge flow to provide emergency cooling, defog, CO_2, and nitrogen washout in the event of a vent loop or cooling system failure. In this mode, the purge oxygen is supplied from the O_2 control module.

A Pressure Transducer in the vent loop provides O_2 ventilation circuit pressure information to a Caution and Warning system, while the suit pressure gauge provides direct, visual readout of suit loop pressure to the crewmember.

When the pilot goes on EVA, the umbilicals connecting the spacesuit to the vehicle EC/LSS subsystem are disconnected at the suit. The umbilical interfaces for both the vent loop and cooling loop are shown at the left of Figure 4-20. The umbilical is configured so that, when disconnected, a bypass valve closes, enabling the vehicle vent loop to continue to circulate flow. This closure is required because the pump gas traps in the liquid coolant loop are tied into the fan water separators.

The vent loop is re-oxygenated with make-up oxygen prior to being returned to the pilot from the high-pressure oxygen supply. In addition, the stored high-pressure oxygen is used to pressurize the water and ammonia tanks, which, in turn, are used to supply fluid to the evaporator for heat rejection. The vent loop equipment in Spaceplane is located behind the pilot's seat, as shown in Figure 4-21.

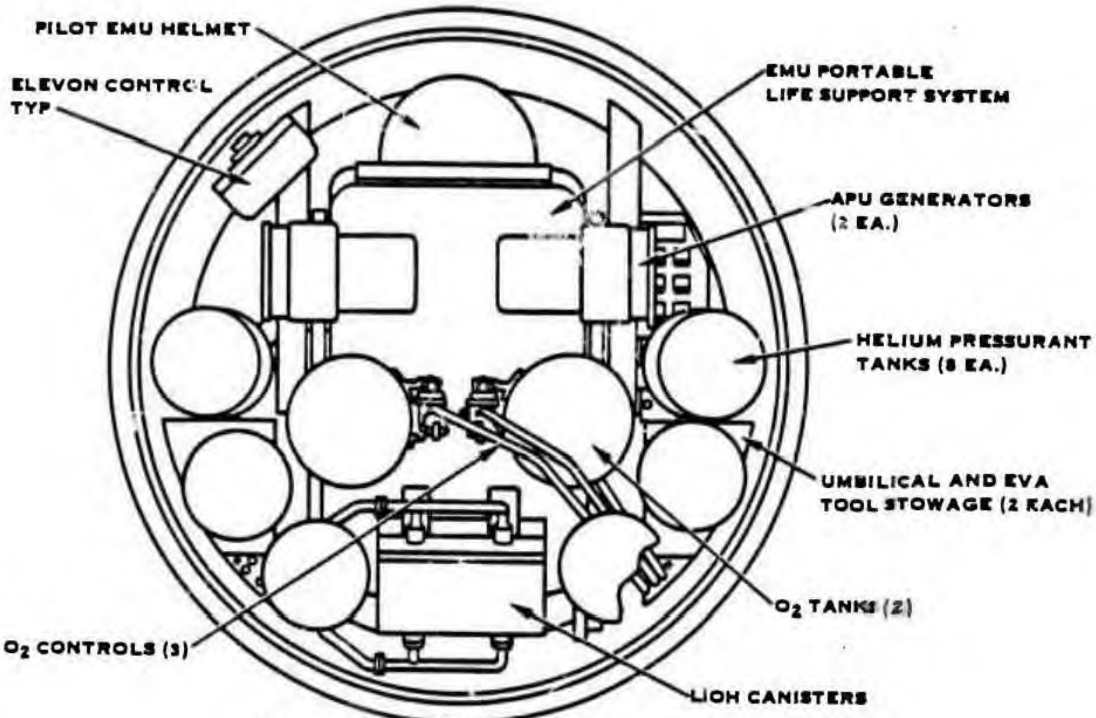


Figure 4-21 EC/LSS HARDWARE INTEGRATION
(VIEW: LOOKING FORWARD FROM BEHIND PILOT)

The oyxgen tank stores a total of 5 lbs minimum of usable oxygen. The tank is isolated from the vent loop by a Manual Shutoff Valve. Two regulators provide oxygen at usable pressures. One regulator provides 15.0 $\pm$ 0.2 psid gas to pressurize the feedwater and ammonia circuits. Check valves in the feed circuits prevent water and ammonia tank depressurization in the event of leakage. The other regulator pressurizes the ventilation circuit. A Flow Limiting Restrictor upstream of both regulators limits O_2 flow in the event either regulator fails open. An orifice, downstream of the Feed Pressure Regulator, permits a trickle flow to the vent loop, preventing the regulator from dead-heading. A Relief Valve, in parallel with the orifice, protects the feed loop from an open failure of the regulator, dumping the flow to the ventilation loop and thence overboard via the suit loop relief valve. The suit loop relief valve will also handle the flow from a failed-open Vent Loop Regulator. A check valve prevents backflow through the regulator when O_2 is shut off.

Liquid Coolant Loop

The redundant liquid transport loop shown in Figure 4-20 circulates cooling water through the LCVG and various Spaceplane cold plates for electrical component cooling. The centrifugal pump shown circulates water through the liquid transport circuit.

The liquid transport loop pressure level is established by feedwater circuit pressure applied at the inlet of the centrifugal pump. Liquid transport circuit pressure is tapped off the feedwater circuit upstream of the feedwater pressure regulator so the pressure level is the same as exists in the water tanks; namely, 15.0 $\pm$ 0.2 psid above O_2 ventilation circuit pressure. The higher pressure in the liquid loop limits diffusion of gas from the oxygen vent loop into the liquid loop, which could cause pump inlet vapor lock.

Water temperature to the pilot is manually adjustable by the cooling control bypass valve located on the EMU. During operation, the coolant flow control module adjusts the feedwater or NH_2 feed rate, thus adjusting the liquid loop temperature.

Purging of LCVG gas that has entered the liquid loop is accomplished by the Gas Trap located immediately upstream of the pump. LCVG flow brings bubbles to the gas trap, which is a hydrophyllic screen that passes water but blocks bubbles. The bubbles are bled off through orifices, along with approximately 1% to 3% of the LCVG water flow, to the fan water separator in the vent loop.

After system shutdown, the pitot-actuated valve in the water separator inlet line automatically closes at 2 psi above suit loop pressure, preventing the LCVG water from leaking through the separator into the Ventilation Circuit. This valve closes in response to low separator pitot tube pressure. If this valve remained open after a water separator malfunction, LCVG water could enter the ventilation circuit. During a dry startup, this valve is manually overridden to an open position after the separator is energized. This condition remains in effect until pitot pressure builds up to 5 psid above suit loop pressure, thereby automatically holding the valve open.

Liquid transport loop pressure tends to force water through the Gas Trap to the water separator. Should the pump become airbound, the Check Valve assures that make-up water flows through the pump, clearing it of bubbles.

A Temperature Sensor at the heat rejection device outlet monitors the cooling water temperature leaving the heat rejection device and provides an analog signal input to the coolant flow control module.

After cooling the crewmember, the circulating water passes through cold plates to cool avionics, electronics, and payload before returning to the evaporator for heat rejection.

The feed circuit shown on the right side of Figure 4-20 contains the equipment, expendable water, and ammonia to reject heat loads imposed on the system by the man, equipment, and environment. The feedwater circuit also collects metabolic condensate to supplement expendable cooling water.

Approximately 150 lbs of feedwater and 10 lbs of feed ammonia are stored in respective reservoirs. The reservoirs are then pressurized to 15 psid above O_2 circuit, assuring that any gas dissolved in the feedwater remains in solution in the reservoirs. Water at this pressure tops off and pressurizes the liquid transport loop. The feedwater (or ammonia) then passes through the respective pressure regulator that reduces pressure to (TBD) psid at a flow rate of (TBD) lbs/hr. The feedwater, or ammonia, boils in the stacked evaporator heat rejection device, which cools the LCVG water and O_2 flowing adjacent to the device. Shutoff valves initiate feed flow to the heat rejection device in response to a crewman command. Feed Pressure Sensors provide continuous monitoring of ambient and feed pressures to a Vehicle Caution and Warning System.

Condensate picked up by the slurper is passed to the water separator, along with some ventilation gas. A gas and water mixture also enters the water separator from the Gas Trap through the Pitot-Actuated Valve. The water separator separates the gas/water mixture, returning the gas to the ventilation circuit and pumping the water through the Condensate Water Relief Valve to the feedwater circuit for use as feedwater.

During reentry operation, no feedwater is required, because ammonia provides the cooling function. Condensate thus generated is either stored or discharged to ambient through the evaporator.

During a dry startup, several minutes are required for sufficient condensate to accumulate in the Water Separator. During this time, the pitot tube may be dry, developing no head. Pressurized feedwater is prevented from flowing back through the separator and flooding the suit loop by means of the Condensate Water Relief Valve in the pitot discharge line. When sufficient condensate has built up in the pitot tube so that 3.1 to 3.7 psid above water tank pressure is generated across the relief valve, the relief valve opens, allowing condensate to leave the separator. If the pitot tube is partially dry in startup, pumping into the feedwater circuit might occur. The Condensate Water Relief Valve prevents this pumping by blocking the flow until full operating pressure is reached.

The Water Separator is initially charged with water by temporarily depressing a manual override button on the Pitot-Actuated Valve to allow some of the water in the water transport circuit to enter the Water Separator. After the separator has built up a 2 to 5 psid head, the Pitot-Actuated Valve opens to allow subsequent water/gas mixture to enter the separator from the gas trap. When 3.1 to 3.7 psid head has been generated by the Water Separator, the Condensate Relief Valve opens, allowing water to leave the separator and enter the feedwater circuit.

If the suit is stored after feedwater recharging and thermal expansion causes the water tank pressure to exceed 19 $\pm$ 1 psid, the Feedwater Relief Valve will open, bleeding the excess water to ambient.

Servicing of the unit is normally performed between missions and involves filling and draining the water and ammonia tanks. Bacteria Filters in the water fill line prevent bacteria migration to the vehicle.

The heat loads from various sources used in the preliminary design of Spaceplane EC/LSS subsystem are shown in Table 2-II. As indicated, the average heat load assumed in space is 4,000 Btu/Hr with higher peak load of 11,300 Btu/Hr sustained during the reentry phase of the mission. The evaporator is sized to reject heat at the highest peak loads to be encountered. Results of the EC/LSS computer model run for the two cases are shown in Figures 4-22 and 4-23.

Negative and positive cabin pressure relief valves are included to maintain cabin pressure equal to that of the ambient environment. A surface pressure profile representative of the stagnation point at the vehicle nose and at a point at the vehicle base is shown in Figure 4-24. The pressure relief valves would be located at the vehicle base to avoid overpressurization of the cabin. As shown in Figure 4-24, the cabin could be safety repressurized to one atmosphere from space vacuum during the 25-minute (1,500 sec) reentry cruise, which starts at 300,000 feet.

After recovery system canopy deployment, the vehicle hatch could be opened to assist with through-flow vehicle ventilation cooling. After landing, the post-landing ventilation fan is turned on manually to circulate air through the vehicle to continue to remove heat soakback through the thermal protection system tiles to the vehicle structure.

Studies were conducted to define the Spaceplane requirement for reentry cooling. The reentry profile (Sandia) shown in Figure 4-25 was assumed. In addition, the following assumptions were made:

Figure 4-22

(To Be Supplied)

Figure 4-23

(To Be Supplied)

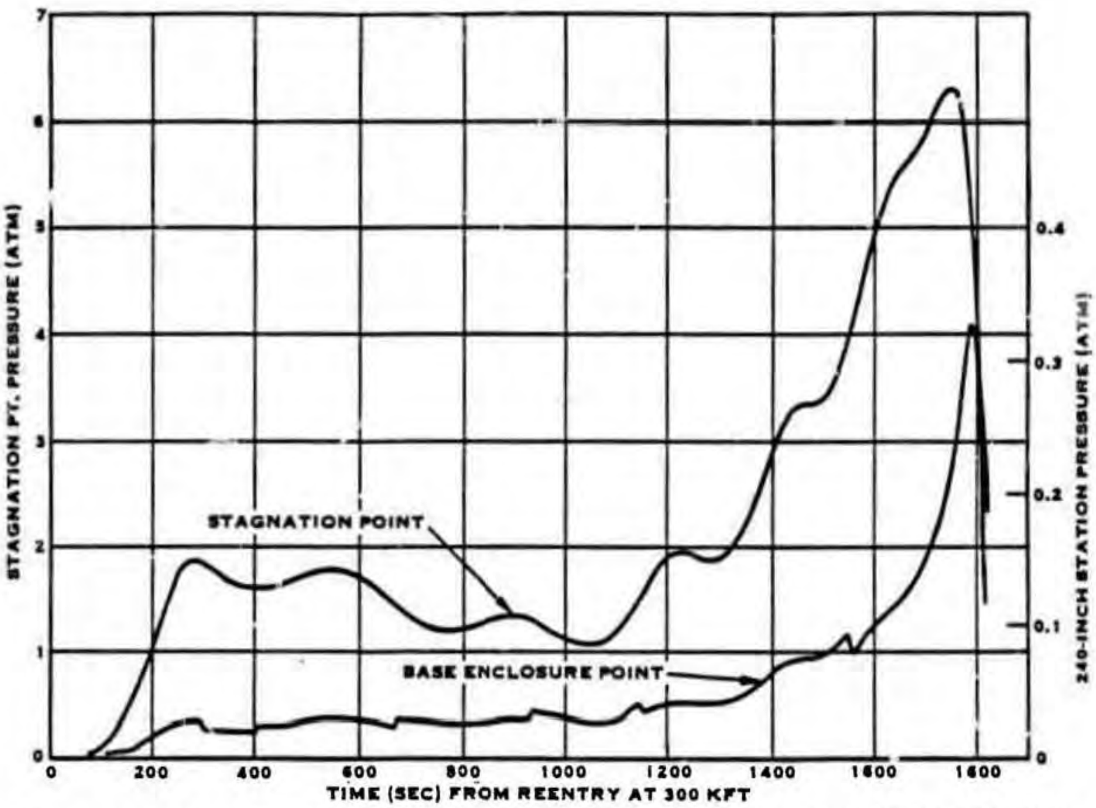


Figure 4-24. **SPACEPLANE SURFACE PRESSURES DURING REENTRY (TWO STATIONS)**

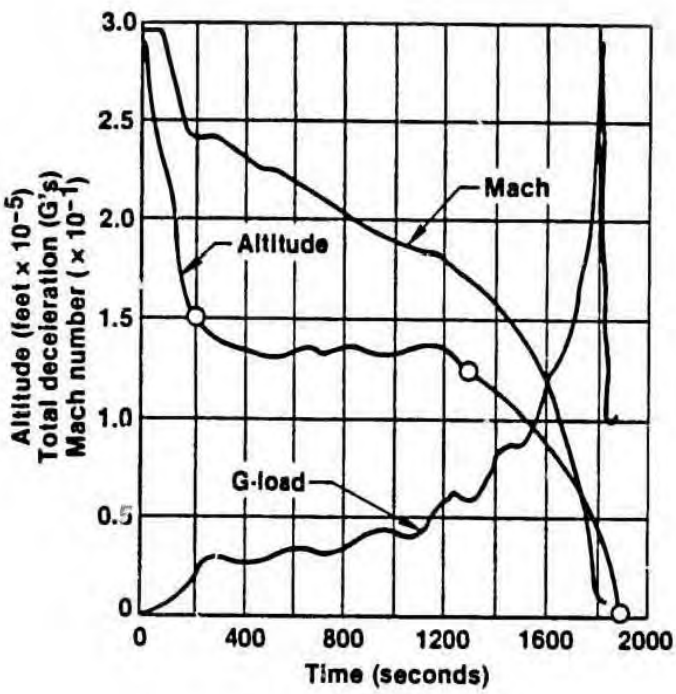


Figure 4-25. REENTRY PROFILE

Assumptions for Reentry Analysis

- Sandia reentry profile

- Reentry heat load (Btu/hr)

Payload	300
Avionics and communications	1,100
Life support	600
Metabolic	1,000
APUs and electrical power	1,000
Aerodynamic heating	7,300
TOTAL	11,300

- EC/LSS thermal mass 47.45 Btu/°F

- Spaceplane base pressure 0.055 psi (based on BIM data, 125,000 feet altitude).

The first analysis was to determine if adequate pilot cooling can be maintained by relying entirely on EC/LSS thermal mass following evaporator/water cutout between 150-100 thousand feet. Figure 4-26 shows the results of this analysis, which indicate that if water/evaporator cooling cuts out at 125,000 feet, the coolant temperature to the pilot will rise in excess of 90°F prior to touchdown. Ammonia cooling is therefore recommended during reentry below the altitude where water evaporator cooling can be provided. Experience shows that ammonia is a good choice of evaporator fluid at the lower altitudes because of its lower boiling point when compared with water.

Characteristics of hardware components to be used in the EC/LSS of Figure 4-20 are given in Table 4-V. This hardware is either Shuttle EMU Hardware or derived from other space-qualified hardware. In the few isolated cases where new hardware components will be required, this hardware will be based on proven technology. Pictures of derivative Space shuttle and EMU hardware, listed in Table 4-V and planned for use in the EC/LSS subsystem, are shown in Appendix C.

4.5.5.3 EC/LSS Subsystem Hardware Specification

The EC/LSS subsystem hardware equipment shall meet the following specifications:

Operating Life

The unit shall be designed for a continuous operating life of (TBD) years.

Physical Characteristics

Maximum Operating Pressure	(TBD) psid
Proof Pressure	(TBD) psid
Burst Pressure	(TBD) psid

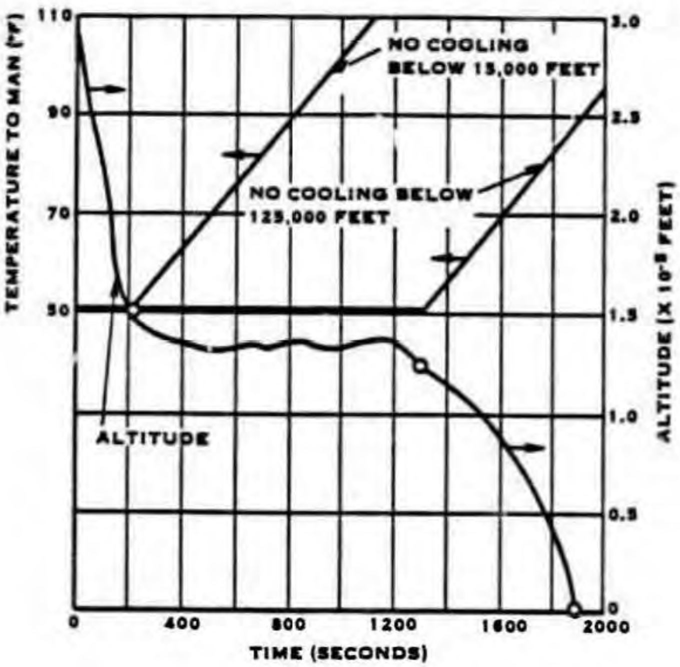


Figure 4-26. **REENTRY COOLING**

TABLE 4-V
EC/LSS SUBSYSTEM COMPONENTS

COMPONENT	QTY	TOTAL VOL (IN^3)	TOTAL DRY WT (LB)	DERIVATION
O_2 TANK	2	1200.00	20.00	STANDARD SPHERICAL TANK
O_2 CONTROL	3	180.00	5.40	EMU O_2 REG. MODIFIED
NH_3 TANK	2	753.00	8.00	SHUTTLE H_2O ACCUM.
H_2O TANK	2	5962.00	50.00	CONFORMAL LIKE EMU
COOLANT FLOW CONTROL MODULE:	1	180.00	10.00	EMU COMPONENTS
S/O VALVE (H_2O)	6	—	—	EMU
CYCLING VALVE (NH_3)	3	—	—	EMU
REG. VALVE (H_2O)	3	—	—	EMU
STACKED EVAPORATOR	1	128.00	11.00	SUBLIMATOR/AMMONIA BOILER
CCC (LiOH)	1	3360.00	31.00	SHUTTLE MOD.
PUMP/FAN/H_2O SEP	2	270.00	7.00	EMU
POST LANDING VENT FAN	1	90.00	2.50	SHUTTLE MOD.
GAS TRAP	2	7.70	1.35	EMU
CO_2 SENSOR	2	44.00	1.54	EMU
CONDENSATE VALVE VENT LOOP	2	5.00	1.00	EMU

TABLE 4-V(CONT)
EC/LSS SUBSYSTEM COMPONENTS

COMPONENT	QTY	TOTAL VOL (IN^3)	TOTAL DRY WT (LB)	DERIVATION
CHECK VALVE	3	15.00	1.50	EMU
PURGE VALVE	1	0.50	0.10	EMU
POS. PRESS. RELIEF	2	2.35	0.70	EMU
CONDENSATE CHECK VALVE	6	3.00	2.10	EMU
VENT FLOW SENSOR	1	2.50	0.70	EMU
BYPASS COOLING CONTROL	1	0.50	1.00	EMU
NEG. PRESS. RELIEF	2	0.50	1.00	EMU
VENT BYPASS	1	0.50	1.00	SHUTTLE
LIQUID LOOP CHECK VALVE	2	2.00	0.70	EMU
PUMP/MOTOR	1	90.00	3.00	STANDARD CENTRIFUGAL
SUBTOTALS		12296.50	160.59	
MISC. BRACKETRY AND TUBES		1200.00	30.00	
TOTALS		13496.50 IN^3 (7.8 FT^3)	190.59 LB DRY	

Leakage

External Leakage

Water leakage in the coolant loop shall not exceed (TBD) cc/hr at (TBD) psid per loop. Oxygen leakage in the vent loops shall not exceed (TBD) cc/hr at a loop pressure of (TBD) psid.

Weight

The dry unit weight shall not exceed (TBD) lbs.

Power

The total pump and fan operating power shall not exceed (TBD) watts.

Performance (Design Point)

a. Heat Rejection - The coolant loop shall be capable of rejecting 4,000 Btu/hr average Spaceplane heat load, with a peak Spaceplane heat load of (TBD) Btu/Hr.

b. Coolant - The coolant loop shall supply coolant at a rate of 240 lb/hr, at a temperature of 50° to the crewman.

c. Oxygen Supply - The vent loop shall provide maximum oxygen makeup at a rate of (TBD) lb/hr and maintain a total loop pressure of (TBD) psid.

d. CO_2 Removal Rate - The vent loop shall maintain CO_2 partial pressure at not to exceed (TBD) psi.

e. Vent Loop Flow - The vent loop shall maintain an oxygen flow of (TBD) lb/hr.

f. Humidity - The vent loop relative humidity shall be maintained at (TBD), at a temperature of 70°F.

Flight Environment

The unit shall be capable of meeting the operating requirements, as specified herein, at the following environmental conditions:

a. Ambient Environment

Atmosphere	Air to space vacuum
Pressure	15.23 psia to 1×10^{-10} torr
Temperature	-25 to 180°F

(Conductive/Convective/Radiation Environments)

b. Acceleration - $\pm$3g's in any attitude

c. Humidity

Dew Point Temperature 0 to 84°F

(Relative humidity can reach 100%)

d. Acoustic and Random Vibration - Consistent with Orbiter payload bay levels.

e. Shock Loads - $\pm$ (TBD) lbs force.

4.5.6 EMU

The EMU proposed for Spaceplane is an 8 psi system that would be a derivative of the existing Shuttle EMU. Only those features of the Spaceplane EMU that differ from the Shuttle EMU are discussed in detail here.

4.5.6.1 EMU Specification

The Spaceplane shall satisfy the following specifications:

- Provide self-contained Environmental control and Life Support.
- Provide required work mobility and glove dexterity.
- Interface with the vehicle.
- Incorporate food and water supply.
- Incorporate Waste Management Systems.
- Incorporate Voice Controller and Heads-Up Display system.
- Include detachable PLSS provision.

4.5.6.2 EMU Description

The Spaceplane EMU performs two basic functions for the pilot. It provides the life support function by way of a controlled, pressurized environment; clean oxygen for breathing; temperature control, waste management; and food and water supply. Secondly, it provides a mobile enclosure, the spacesuit, which permits tasks to be performed in the vacuum of space.

The current Shuttle EMU contains end items that would also be included in the Spaceplane 8 psi EMU, but with some differences in certain cases. The end items in which major differences are seen will now be discussed.

Portable Life Support System (PLSS). The portable life support system backpack, mounted on the rear of the spacesuit upper torso, contains the life support subsystem hardware and expendables. The PLSS structure, check valves, and regulators will be designed to operate at 8 psid.

Simplified schematics for the Spaceplane PLSS vent and liquid loop are shown in Figures 4-27 and 4-28. The umbilical interface with the vehicle EC/LSS is shown at the left of each figure.

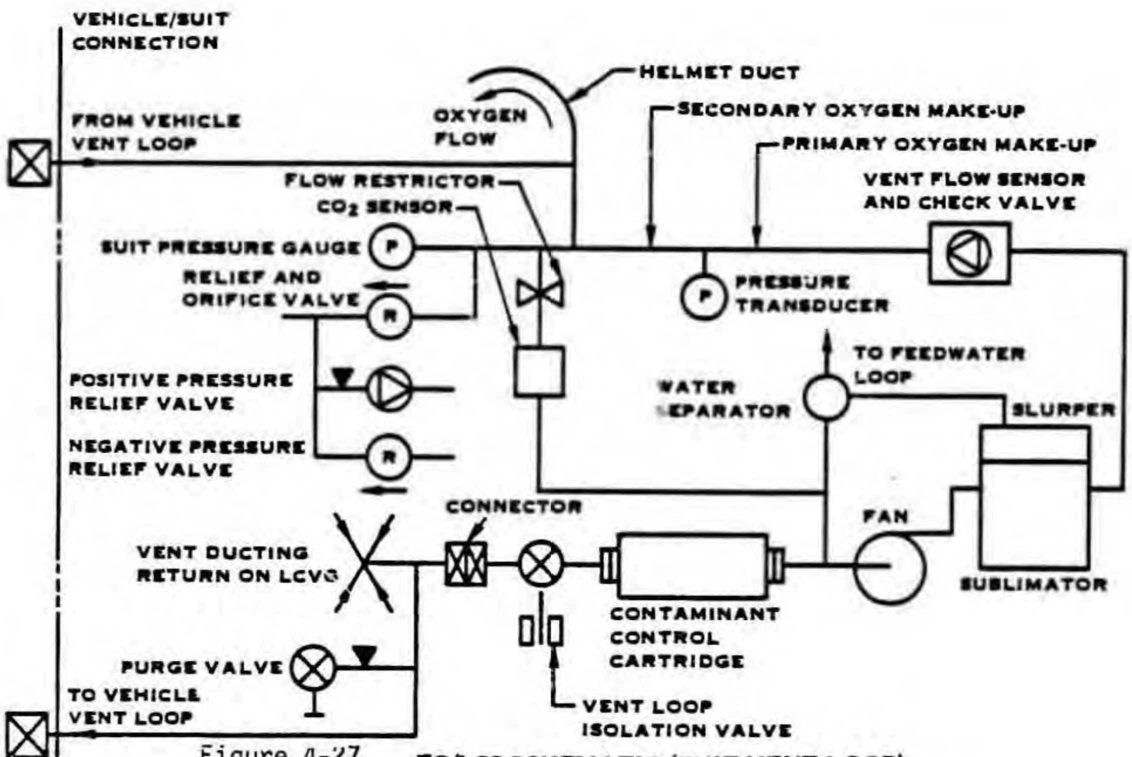


Figure 4-27. EC/LSS SCHEMATIC (SUIT VENT LOOP)

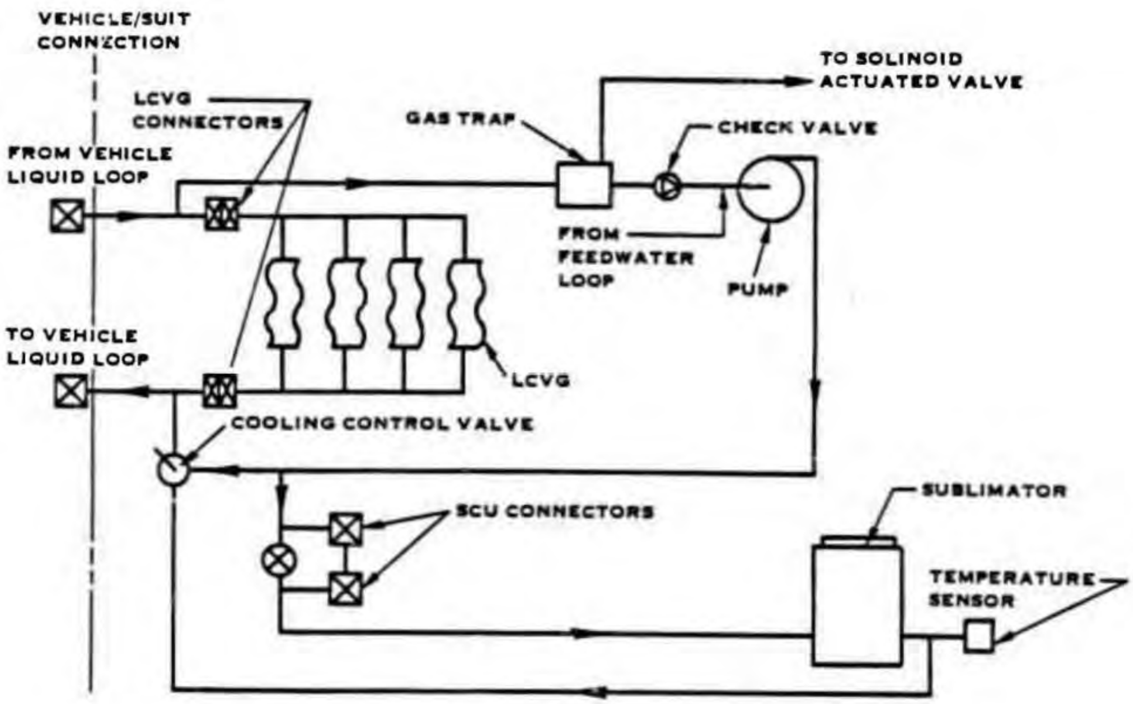


Figure 4-28. EC/LSS SCHEMATIC (SUIT LIQUID LOOP)

The existing Shuttle EMU does have an umbilical provision for Shuttle airlock operations providing for EMU checkout and service, which the Spaceplane EMU would also require. In addition, the Spaceplane EMU will require vent loop and liquid loop umbilicals, each providing a supply and return line between the vehicle EC/LSS and the EMU. In addition, some minor plumbing modifications will be required within the EMU spacesuit.

During EVA, the Spaceplane EMU will provide environmental control and life support through the PLSS. Following is a description of PLSS operation.

Oxygen Ventilating Circuit. The Oxygen Ventilating Circuit, shown in Figure 4-27, forms a closed loop with the spacesuit assembly within which oxygen for breathing and suit pressurization is circulated and purified. This oxygen picks up heat, humidity, CO_2, and other contaminants from the astronaut. These metabolic by-products are removed from the oxygen by the Oxygen Ventilating Circuit components.

The Oxygen Ventilating Circuit operates over a range of pressures controlled by the Primary Oxygen Subsystem or the Secondary Oxygen Pack (SOP).

A centrifugal fan (integrated into a fan/separator/pump assembly) running at 14,000 rpm to 20,000 rpm drives the ventilation flow through the spacesuit. The gas flow picks up CO_2 and trace contaminants, humidity, and sensible heat from the astronaut. The Contaminant Control Cartridge removes CO_2 and trace contaminants. The gas stream continues through the Sublimator, where it is cooled. Excess humidity is removed from the ventilation oxygen circuit and condensed in the Sublimator.

The CO_2 level is measured in the ventilating circuit upstream of the Sublimator by an electrochemical CO_2 sensor. A check valve/flow sensor is located downstream of the Sublimator. The check valve assures that when the fan is unpowered, all purge flow is directed to the helmet. The flow sensor monitors flow levels.

The ventilation circuit flow then passes to the HUT, through hard-plumbed connections and then to the helmet. Ventilating circuit ducting in the HUT superheats the inflow O_2 sufficiently to prevent visor fogging. Within the helmet, the flow passes downward over the visor and head to provide CO_2 removal and head cooling, then continues to the pressure garment. Ventilation flow is picked up at the extremities and returned to the HUT via a manifold that is part of the LCVG.

During emergency, or other conditions requiring open-loop operation, the ventilating circuit can be operated in a purge mode. Purge, checkout, and pressure relief are all accomplished with valves mounted in the DCM. The purge valve is manually activated and passes O_2 from the ventilating circuit. Positive pressure relief protection against failed-open SOP or Primary Oxygen Subsystem regulators is effected by a relief valve. A Negative Pressure Relief Valve permits suit internal pressure to rise at the same rate as the Orbiter airlock during an emergency repressurization.

The Condensate Circuit removes the condensate generated by the ventilating circuit and returns it to the feedwater circuit.

Condensate collected by the slurper is passed to the water separator with a small amount of ventilation gas. A gas and water mixture also enters the water separator from the gas crap through the pitot-actuated valve. The water separator separates the gas/water mixture, returns the gas to the ventilating circuit and pumps the water through a condensate water relief valve into the feedwater circuit.

Feedwater Circuit. The Feedwater Circuit stores and supplies feedwater for LCVG makeup, provides water for EMU sublimator cooling, and condenses and delivers condensate to supplement the feedwater supply. Feedwater is stored in a water reservoir and pressurized by the Primary Oxygen Circuit. Feedwater passes through a pressure regulator that reduces pressure to top off and pressurize the LCVG loop and to feed the sublimator. Feedwater passes into the sublimator, freezes in the stainless steel, porous plate and sublimates to space, cooling the LCVG coolant water as it flows through the sublimator.

Liquid Transport Circuit. The Liquid Transport Circuit, as shown in Figure 4-28, circulates water through the LCVG and sublimator via a centrifugal pump, thereby performing crewman temperature control. The LCVG pressure level is regulated by the Feedwater Circuit. A Cooling Control Valve, located on the DCM, provides temperature control by permitting the astronaut to bypass a variable amount of water around the sublimator or ALSS heat sink back to the LCVG.

The PLSS Ventilating Circuit heat loads are transferred to the Liquid Transport Circuit in the sublimator, which acts as a gas-to-liquid heat exchanger in this mode. The PLSS pump provides the water circulation through the Liquid Transport Circuit.

Two regulators provide oxygen at usable pressures. One regulator pressurizes the feedwater circuit and the other pressurizes the Ventilating Circuit for EVA operations.

Operating Pressure. Preliminary results of a prebreathe study, which are of interest in the context of an 8 psi spacesuit for Spaceplane, are shown in Figure 4-29. Figure 4-29 first assumes that a crewman, acclimated to an earthlike environment at some time prior to zero, enters an 8 psi spacesuit of pure oxygen and, after 10 minutes, purges the suit of the nitrogen removed from the lung cavities during breathing. At time zero, the crewmember then continues tissue denitrogenization in the spacesuit environment of essentially pure oxygen at a total pressure of 8 psi. Results, shown in Figure 4-29, indicate the elapsed time required in the 8 psi spacesuit to safely depressurize in a a single step to lower suit pressures of 6, 5, and 4 psi. The advantages of a step-wise spacesuit depressurization provision on certain missions might be that the crewmember could safely reduce suit pressure prior to going EVA, which would improve work mobility and, perhaps, improve spacesuit life and extend the time before oxygen toxicity effects occur.

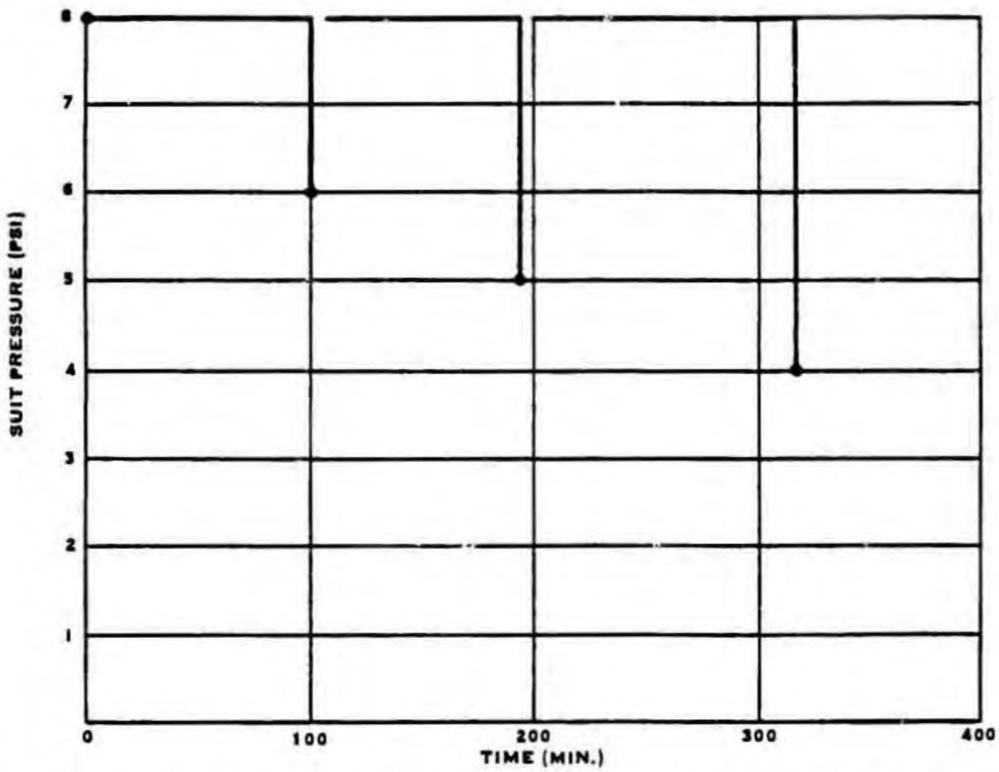


Figure 4-29. VARIABLE SUIT PRES-URE PROVISION (PRELIMINARY RESULTS)

However, with the current development of 8 psi spacesuit joints, such a scheme is probably unnecessary.

Space Suit Assembly (SSA). The SSA consists of end items forming an anthropromorphic pressure vessel that protects the extravehicular crewmen from extreme temperatures, micrometeoroids, and space vacuum. The major task in development of an 8 psi EMU will be development of new suit joints and gloves to provide adequate pilot mobility and dexterity at the higher pressure. Ongoing work indicates that this technology is achievable.

Food and Water Supply. A diet adequate to satisfy the pilot throughout the 24-hour mission should provide approximately 3,000 to 4,000 calories of energy, with adequate quantities of proteins, fats, carbohydrates, minerals, and vitamins. Exact caloric requirements for pilots vary, based on the actual body weight, age, height, and level of activity for the specific mission. Thirty-six (36) ounces of water, contained in a drink bag, and sixteen (16) ounces of food sticks would be integrated into the EMU suit and accessible to the pilot in the EMU Helmet area.

Waste Management. Waste management consists of provisions for urine, vomitus, and feces collection and storage. The urine collection device would have a 2,000 cc capacity to provide for the 24-hour mission duration. The urine collector device (UCD) consists of a urine receiver, shutoff valve, and urine collection bag. The urine receiver is a roll-on rubber sleeve. Just inside the opening of the receiver is a one-way check valve that is forced open by pressure of the in-flowing urine. The UCD is integrated within the spacesuit and can be changed during post-mission maintenance.

The fecal defecation collection system consists primarily of a strong, lightweight plastic bag. The collecting bag is a nylon-ployethylene, laminated plastic bag. The circular opening of the bag is about 4 inches in diameter and is encompassed by a 1-3/4-inch circular flange that is covered with a surgical-adhesive tape for attaching the collecting bag to the body. After defecation, the bag is removed from the body. The plastic pouch would contain a liquid germicide, ruptured by pressure exerted from outside the collecting bag, for mixing with the fecal material.

A vomitus collection device has been concepted that would be incorporated within the EMU.

The design features required of the Vomitus Containment System (VCS) are as follows:

- Container to hold .5 liters (30.5 cu. in.) of vomitus.
- Requirement to accept gas expelled behind vomitus.
- Use EMC purge flow to propel vomitus into receptacle.
- Mouthpiece requires minimal movement to reach it.
- Package complete assembly in helmet.

Figure 4-30 shows the schematic of the vomitus control system. Vomitus is pulled into the hydrophobic bag with suction provided by manual application of the existing purge valve. When purging is required and vomitus is not present, purge flow enters the mouthpiece and goes through the bag with little additional resistance.

In using the system, the crewman places his open mouth over the mouthpiece, presses against it, and regurgitates. During or immediately before this, he actuates the purge valve, exposing the hard bag container to a negative pressure. As the hydrophobic bag becomes clogged, the vacuum opens it against the hard container, drawing vomitus into it. An elastomeric flapper check valve keeps the vomitus in the bag with no purge flow.

Figure 4-31 is a pictorial sketch of the conceptual design. The mouthpiece is located between the purge valve and the bite valve. The VCS is arranged over and behind the crewman's head, which places it under the visor assembly. This position does not restrict visibility.

To change the bag, the mouthpiece and bag are removed and a new assembly, including the mouthpiece, is reinserted into the hard bag.

4.5.6.3 EMU Hardware Specification

The EMU spacesuit system hardware for Spaceplane will be a derivative of existing Shuttle EMU hardware. The EMU hardware for Spaceplane will satisfy the following specifications:

<u>Operating Life</u>

The unit shall be designed for an operating life of (TBD) years.

<u>Physical Characteristics</u>

Suit Operating Pressure - 8 psid.

<u>External Leakage</u>

Oxygen leakage from the suit shall not exceed (TBD) cc/hr at 8 psid.

<u>Weight</u>

The dry unit weight shall not exceed 250 lbs.

<u>Size</u>

Sized to pass through a 20" x 30" opening.

<u>Power</u>

The EMU shall be operable from self-contained batteries.

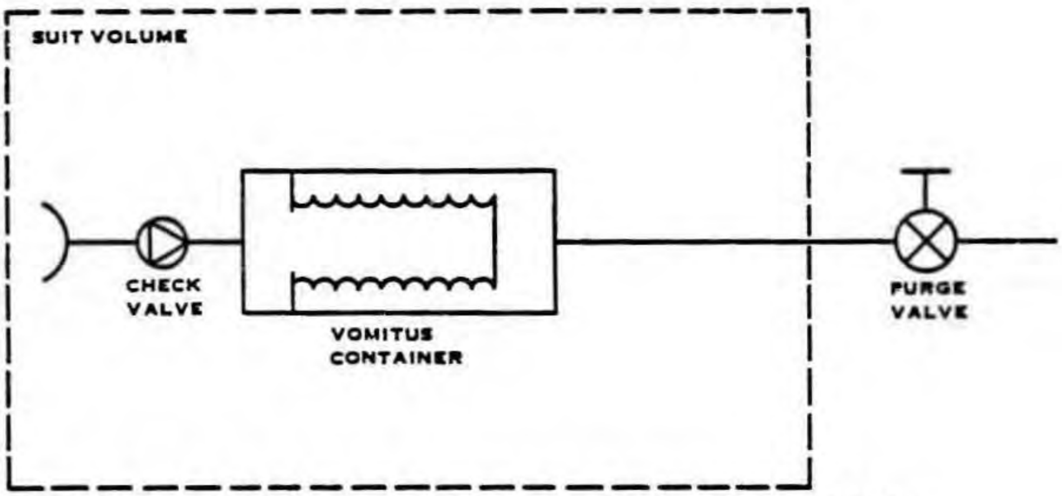


Figure 4-30. **VOMITUS CONTAINMENT SYSTEM (SCHEMATIC)**

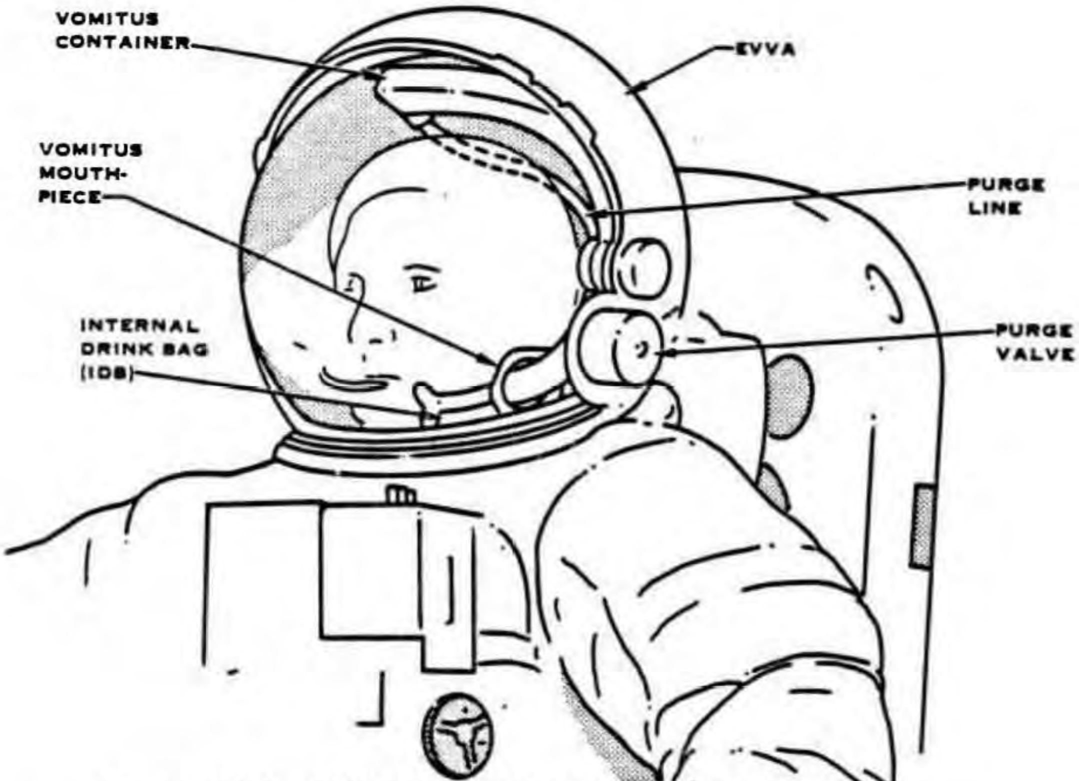


Figure 4-31A. **VOMITUS CONTAINMENT SYSTEM (HARDWARE)**

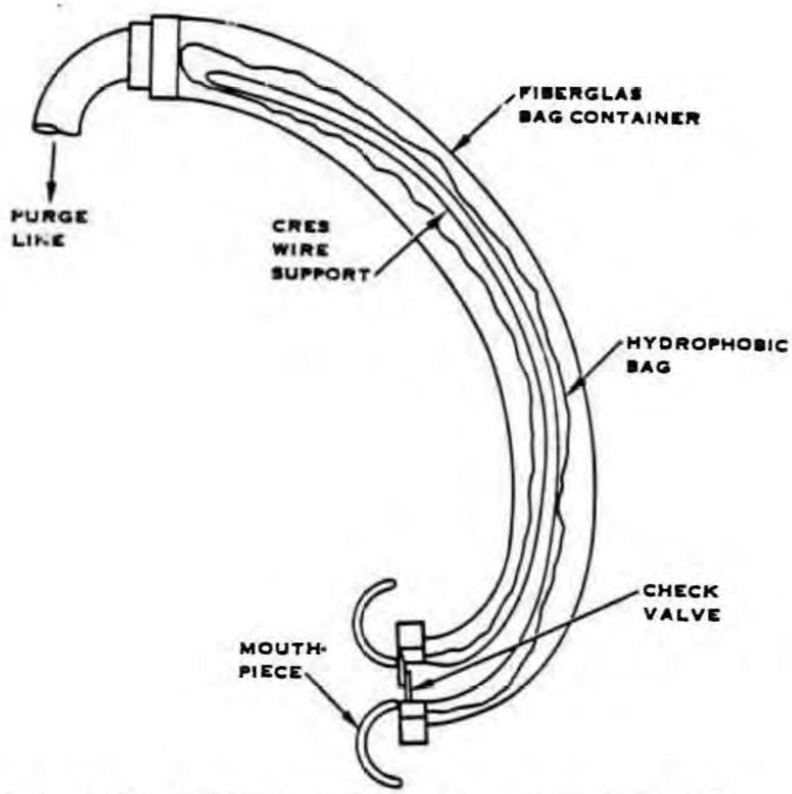


Figure 4-31B. VOMITUS CONTAINMENT SYSTEM (DETIALS)

Flight Environment

The unit shall be capable of meeting the operating requirements, as specified herein, at the following environmental conditions:

a. Ambient Environment

Atmosphere	Air to space vacuum
Pressure	15.23 psia to 1×10^{-10} torr
Temperature	-25 to + 180°F

(Conductive/Convective/Radiation Environment)

b. Acceleration

±3 g's in any attitude

c. Humidity

Dew Point Temperature 0 to 84°F
(relative humidity can reach 100%)

d. Acoustic and Random Vibration

Consistent with Orbiter payload bay levels.

Performance

Function	Provide pressurized spacesuit for support of pilot in Spaceplane for Extravehicular Activities
EVA Life Support	Seven-hours duration EVA (at 1,000 Btu/Hr or 2×10^{6} Joules/hour metabolic heat rejection rate
Emergency Life Support	30 minutes
End Items	(TBD) different items - Life Support System; Spacesuit Assembly
Sizing	Modular assembly to fit 5th to 95th percentile male pilots

Food and Water

An in-suit drink bag shall store 36 ounces of water and have a tube projecting up into the helmet to permit the crewman to drink while suited. Food sticks (16 ounces) shall be integrated into the helmet to provide food to the pilot.

Waste Management

An in-suit, 2,000 cc capacity, urine collection device consisting of adaptor tubing, storage bag, and disconnect hardware, will be provided.

A vomitus collection device with (TBD) cc capacity will be integrated with the helmet.

Checkout and Operation	Microprocessor-assisted, isolates problems and provides corrective instructions
Donning	5 minutes
Construction Features	Helmet and Visors - polycarbonate

Suit Materials

- main restraint - dacron
- bladder - urethane-coated nylon
- thermal/meteoroid garment
    - outer layer (orthofabric) - Gortex, Nomex, and Kevlar
    - 4 layers aluminized Mylar
    - inner layer - Neoprene-coated nylon

Hard Upper Torso - fiberglass and metal construction

Joints - TBD

Life Support Expendables - replaceable or rechargeable on orbit

The Extravehicular Mobility Unit for the Spaceplane will be modularized to fit 5th to 95th percentile men while providing mobility and protection for working in space.

The EMU for the Spaceplane will be a derivative of the Shuttle EMU, shown in Figures 4-32 and 4-33, sharing many common features.

4.5.7 Couch Specification

The Spaceplane couch shall provide the pilot comfort and protection throughout all mission phases. The couch shall satisfy the following specifications:

- Integrate with the EMU/PLSS
- Provide pilot restraint with mobility
- Provide G-load protection
- Include shock attenuation features
- Include deployment mechanism for ease of pilot ingress and egress and on-orbit viewing

Figure 4-32. **SHUTTLE EMU**

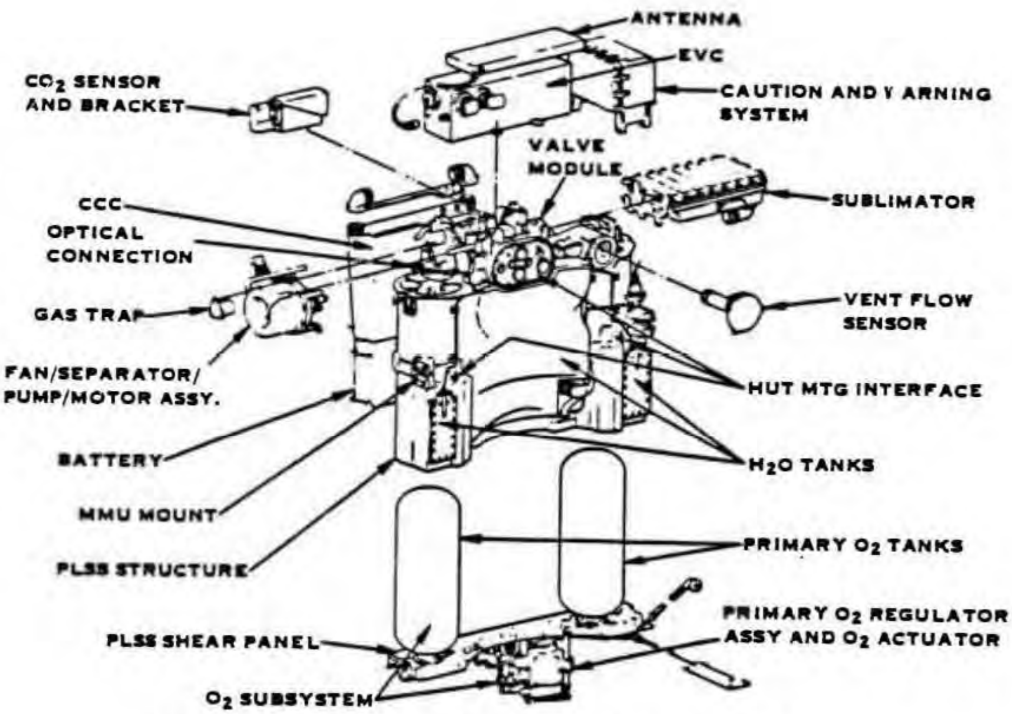


Figure 4-33. **PLSS COMPONENT ARRANGEMENT**

4.5.7.1 Couch Description

The Spaceplane couch is designed to accommodate a 95th percentile pilot suited in the 8 psi EMU with the approximate dimensions indicated in Figure 4-34. The attachment points between the EMU and couch are the same, because the attachment points on the EMU will be standardized for all size suits.

The couch, as shown in Figure 4-35, includes lightweight metal structural members with four hard attachment points to which the EMU/PLSS mounts. The same interface mechanism is planned that is currently used between the Shuttle EMU and the Shuttle Airlock Adapter plate, shown in Figure 4-36. The couch seat would be of nylon webbing similar to that used in the Apollo spacecraft couch.

Padding for crew comfort will be provided within the upper spacesuit torso and the buttocks and legs will be supported by the lower couch, permitting movement and repositioning for comfort and vehicle control. The restraint harness system used with the Apollo couch, as shown in Figure 4-37, might be used with Spaceplane in modified form. The principal change would be to design couch disconnects compatible with EMU arm/shoulder mobility and pressurized glove dexterity.

An EVA consideration is pilot-induced vehicle dynamics. Although the pilot is restrained in his couch on IVA, analysis indicates that changes in momentum resulting from torso or head motions can be effectively nullified by the on-board attitude control system momentum wheels. However, when free from the couch restraint system (for example, during EVA egress through the hatch), pilot motions can induce vehicle moments on the order of 36 in.-lbs. For a vehicle weighing 5,000 lbs with the geometries studied, these moments could result in vehicle yaw, pitch, and roll rates in excess of 30 degrees/minute. In Tables 4-VI and 4-VII, forces and torques that were exerted in Skylab EVA are shown. It is expected that Spaceplane mission EVA could involve similar tasks and associated forces/torques. Ideally, should the pilot impart dynamics to the vehicle when on EVA, vehicle dynamics would be nullified when the pilot again stabilizes through physical contact with the vehicle.

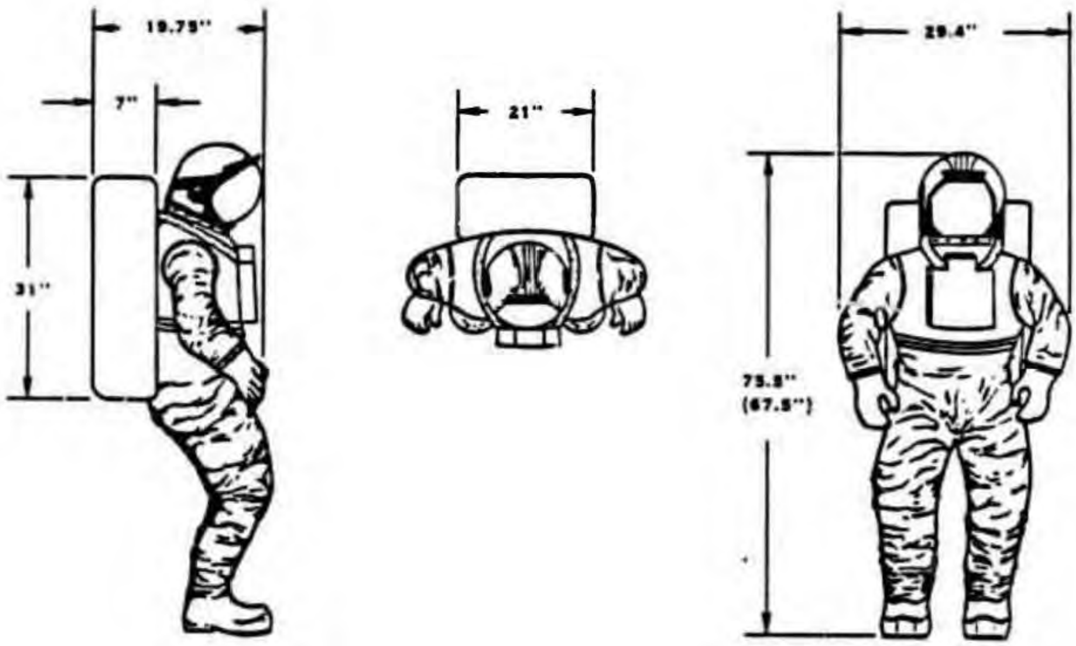


Figure 4-34. **SHUTTLE EMU DIMENSIONS (95TH PERCENTILE MAN)**

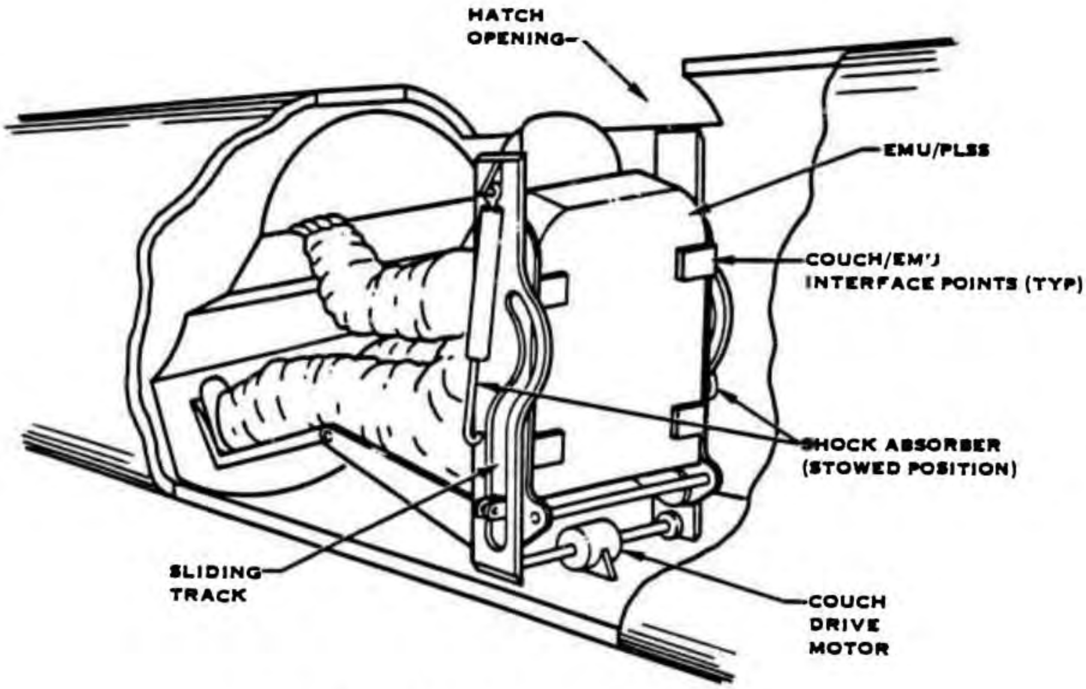


Figure 4-35. COUCH IN FLYING POSITION

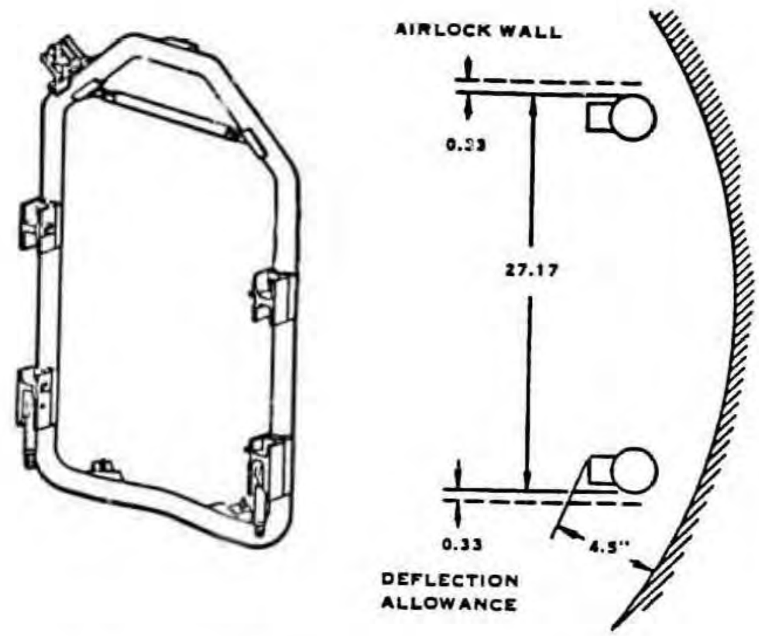


Figure 4-36. **AIRLOCK ADAPTER PLATE**

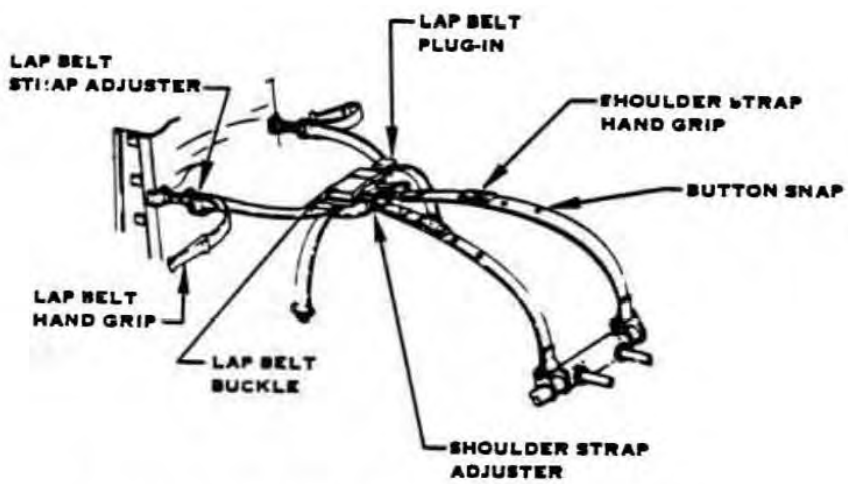


Figure 4-37 APOLLO COUCH RESTRAINT-HARNESS

Table 4-VI REPRESENTATIVE FORCE VALUES FOR SKYLAB EVA HARDWARE

OPERATION	FORCE REQUIRED (LBS)	COMMENTS
OPEN-CLOSE EVA HATCH	45.0	
OPEN-CLOSE INTERIOR HATCHES	35.0	
ACTUATE HATCH LATCHES	25.0	LEVER TYPE HANDLE
OPEN-CLOSE ATM ACCESS DOORS	10.0	
LATCH-UNLATCH S052 LATCH HANDLE	3.5 4.0	PUSH-PULL
INSTALL ATM CAMERA TO INSTRUMENT LATCHING MECHANISM	10.0	S052, S056, H ALPHA 1 EXPERIMENTS PUSH-PULL REQUIRED BY CREWMAN
OVERCOME ATM S082 LATCH LEVER DETENTS	12.0 16.0	DETENTS RESTRAIN THE EXPERIMENT WHEN IN THE UNLOCKED POSITION - PUSH-PULL REQUIRED
OPERATE S082 LATCHING HANDLES SUSTAINED IMPULSE	 18.0 36.0	TANGENTIAL FORCE REQUIRED
ACTUATE S082 CONTAINER LOCKING MECHANISM	3.0	THUMB OPERATED LEVER DEVICE
OPERATE S082 CONTAINER OPEN-CLOSE LEVER	15.0	APPLIED AT 8.5 IN. FROM PIVOT
OPEN S082 CONTAINER LID	3.0 8.0	PULL AGAINST A FRICTION HINGE
ACTUATE S056 TRIGGER MECHANISM (HAND SQUEEZE)	30.0	REQUIRED A SQUEEZING MOTION

TABLE 4-VII
REPRESENTATIVE TORQUE VALUES FOR SKYLAB EVA HARDWARE

OPERATION	TORQUE REQUIRED (IN-LB)	COMMENTS
ROTATION CONTROL LEVER (ATM PANEL 160)	25.0 4.0–6.0	MAXIMUM OPERATIONAL
REGULATOR SELECTOR VALVE (PRESSURE CONTROL UNIT)	15.0	OPERATING TORQUE
FLOW CONTROL VALVE (PRESSURE CONTROL UNIT)	15.0	OPERATING BREAKAWAY TORQUE
FILL AND SHUTOFF VALVE (SECONDARY OXYGEN PACK)	25.0	MAX. OPERATING TORQUE
GAS FILL CONNECTOR (MANNED MANEUVERING UNIT)	10.0	+/-140 DEG. ROTATION
PROPELLANT SUPPLY CONNECTOR (MANNED MANEUVERING UNIT)	30.0	+/-3.32 REVOLUTIONS
ELECTRICAL CONNECTOR (LIFE SUPPORT UMBILICAL TO EVA PANEL)	36.0 7.0	MAXIMUM MINIMUM

However, a net angular displacement of the vehicle from its original attitude would have occurred in the interim. The approach taken may be to allow vehicle attitude perturbations during EVA. When the pilot reenters the vehicle following EVA, the vehicle guidance system would be reactivated to reestablish vehicle attitude.

The couch is designed with deployment provision for ease of pilot ingress and egress from Spaceplane. The couch drive motor mechanism with the seat in different positions is shown in Figure 4-35 and Figures 4-38 through 4-40. The drive motor can raise or lower the upper couch back, which rides in the couch side-track structural members affixed to Spaceplane. The lower couch leg and seat supports are attached to the couch back and move together with the couch back during deployment. When on-orbit, the pilot could deploy the couch to provide unobstructed visibility during such operations as rendezvous and docking with a satellite. Couch deployment would also aid EVA egress.

G-loads - The maximum g-loads to be sustained during Spaceplane missions with Shuttle launch are sustained during reentry, as indicated in Figure 4-41. Preliminary data indicate that, for a maximum 3 g-load directed along the Spaceplane centerline and sustained for a time interval of less than 3 minutes in duration, the pilot in the flying position of Figure 4-19 would not be endangered. The structural design of the couch would be required to support approximately 450 lbs of pilot/EMU during the g-loads sustained in all flight maneuvers.

For an expendable booster ground-launch mode, g-loads in excess of those for the DRM (3-g's) can be expected. Data from AMRL at Wright-Patterson AFB in Table 4-VIII indicate that human tolerance time limits decrease with the level of the g-load sustained. For ground-booster launch of Spaceplane, it will be required that the pilot be repositioned to withstand the high g-loads. Work at AMRL provides some basis for the expectation that a pilot positioned as shown in Figure 4-40 can withstand g-loads expected from an MX launch. The couch in Figure 4-40 has been deployed and locked in position with the pilot extended through the open vehicle hatch but protected beneath the launch vehicle shroud. With the high g-load directed along the Spaceplane centerline, the recommended pilot position would be:

- Upper torso inclined forward, forming a 65° angle with Spaceplane centerline
- Upper torso and thigh included angle of 100°
- Thigh and leg calf included angle of 100°

Finally, the couch design includes shock attenuation provision to protect the pilot during landing. Although the paraglide recovery system provides for canopy flare-out to provide a soft landing, provision must be made for a sustained shock load at touchdown. In Figure 4-39, the couch shock absorber system is shown. A shock absorber is located on each side of the couch, which swings out to a stowed position during all mission phases other than landing.

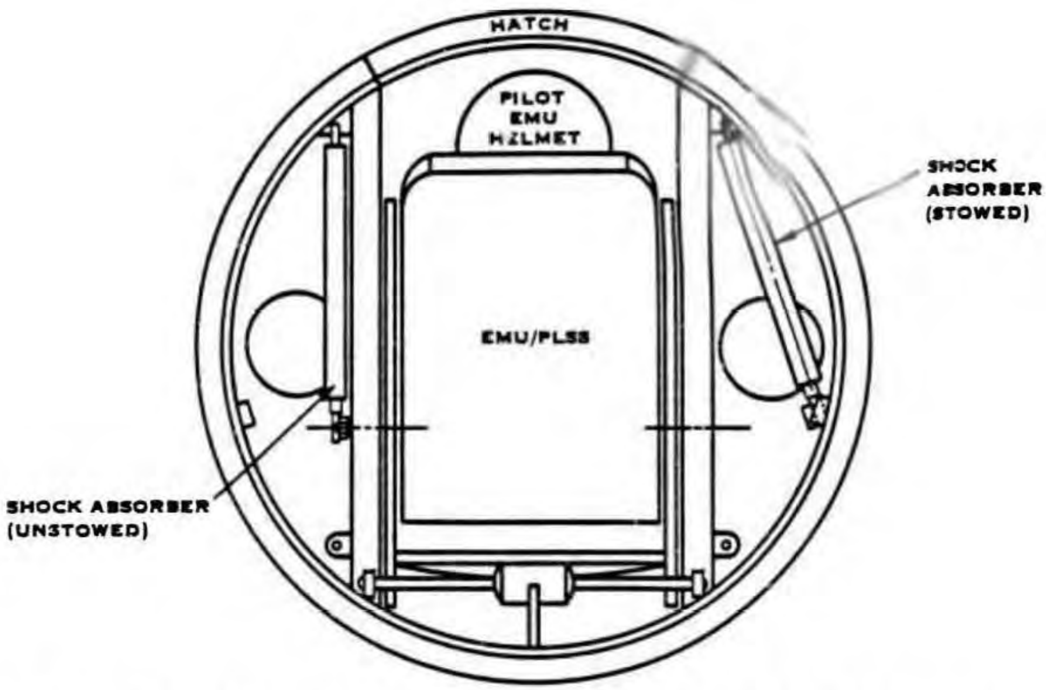


Figure 4-38. **CROSS-SECTION AFT OF PILOT COUCH IN FLYING POSITION**

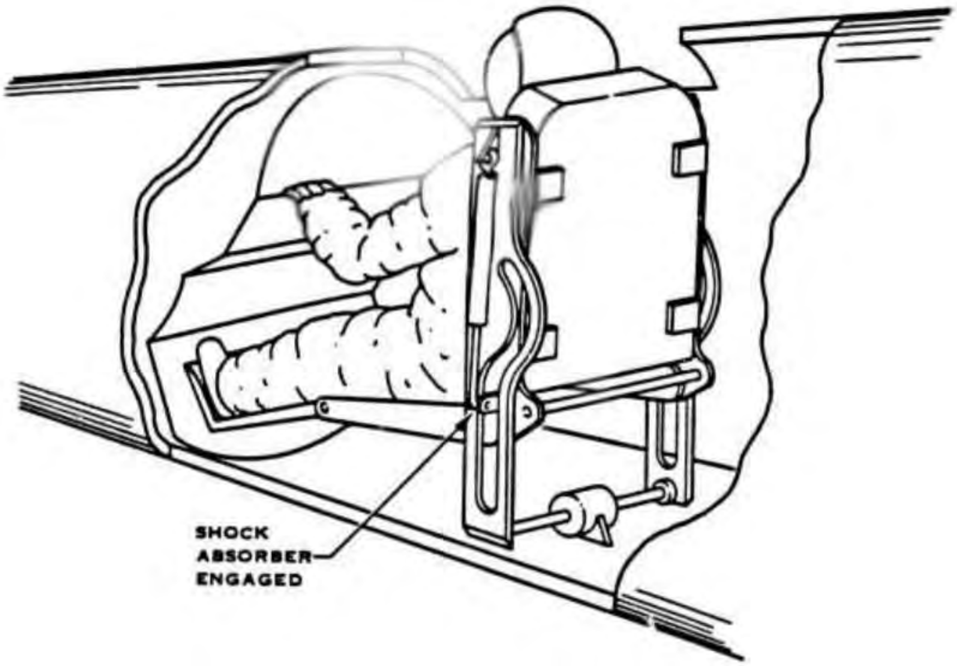


Figure 4-39. **COUCH IN LANDING POSITION (PARTIALLY DEPLOYED)**

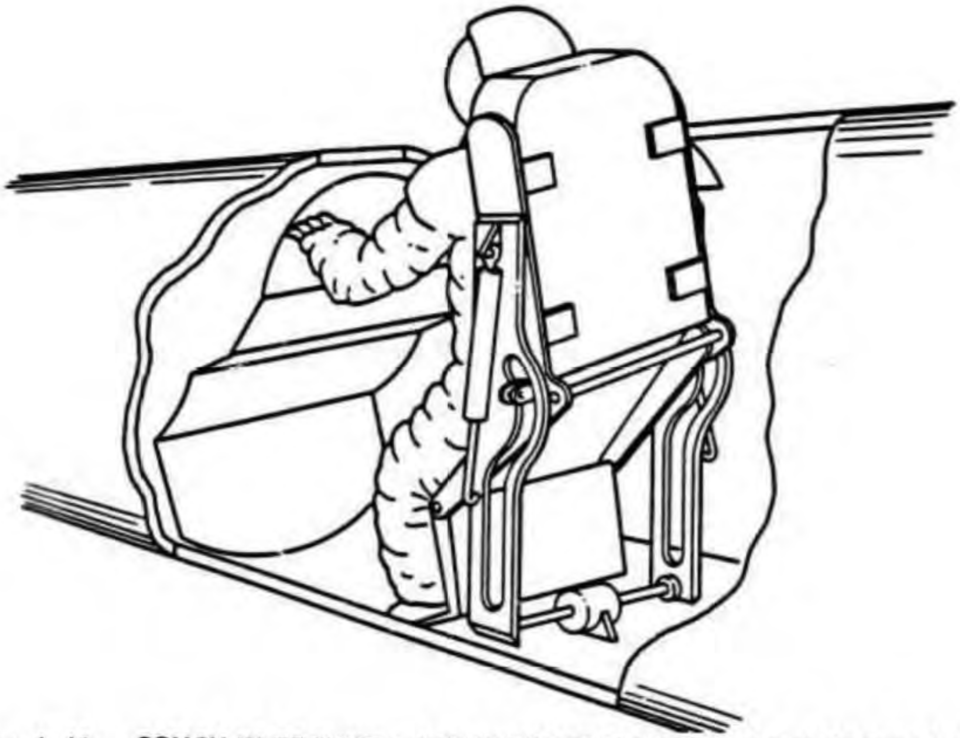


Figure 4-40. **COUCH IN GROUND LAUNCH/SPACE FLIGHT POSITION (DEPLOYED)**

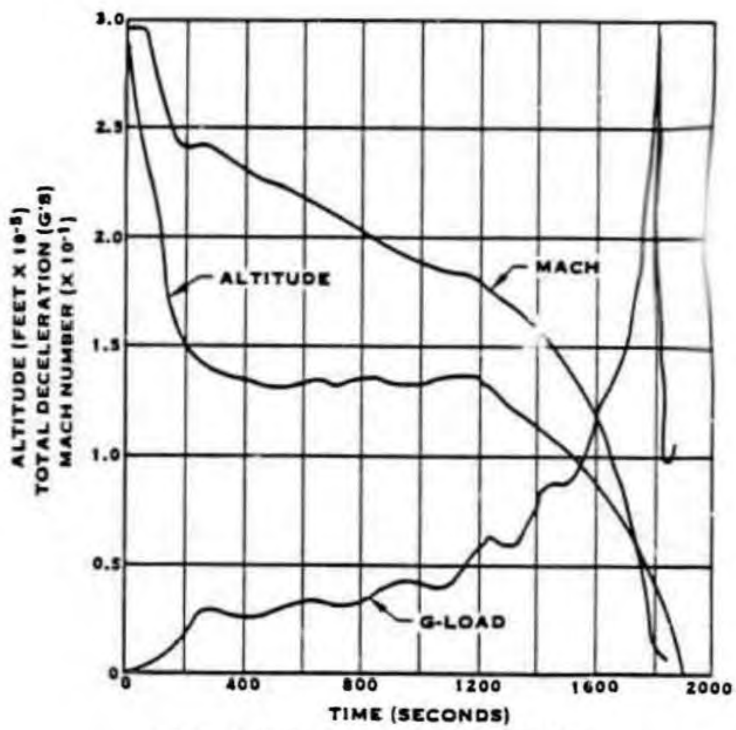


Figure 4-41. **REENTRY PROFILE**

TABLE 4-VIII
HUMAN TOLERANCE TO SUSTAINED ACCELERATION LEVELS

G-LEVEL	DURATION[1]
2	24 HRS
3	1 HR
4	20 MIN
5 WITH G-SUIT	4 MIN
7 WITH G-SUIT	1-2 MIN

[1] USEFUL CONSCIOUSNESS -ADEQUATE PERFORMANCE

During reentry, following deployment of the recovery paraglide canopy, the pilot would open the Spaceplane hatch and partially deploy the couch. The two shock absorbers would then be unstowed by the pilot and swung into the seat upper frame and secured in place. In this position, the shock absorbers would have the required stroke to absorb landing shocks.

4.5.7.2 Couch Hardware Specification

The couch for Spaceplane shall meet the following specifications:

Physical Characteristics

Deployment Range (TBD)
Weight not to exceed (TBD)

Power

Drive motor power not to exceed (TBD)

Flight Environment

The unit shall be capable of meeting the operating requirements, as specified herein, at the following environmental conditions:

a. Ambient electronics Environment

Atmosphere	Air to space vacuum
Pressure	15.23 psia to 1×10^{-10} torr
Temperature	-25 to +180°F

(Conductive/Convective/Radiation Environment)

b. Acceleration

+10 g's in any attitude

c. Humidity

Dew Point Temperature 0 to 84°F

(Relative humidity can reach 100%)

d. Acoustic and Random Vibration

Consistent with Orbiter payload bay levels

e. Shock loads (TBD)

4.5.8 Displays and Controls

4.5.8.1 Displays and Controls Specifications

Without proper design, a proliferation of displays and controls will confront the Spaceplane pilot from which he has to collect, collate, integrate and interpret data. A systematic display and controls design approach to the Spaceplane cockpit is required. The subsystems for Spaceplane shall include:

- Necessary sensors and instrumentation, with appropriate signal conditioning to provide pilot capability for controlling and monitoring subsystems during any mode of mission operations.

- Appropriate Display and Control interfaces compatible with a space suited pilot.

4.5.8.2 Displays and Controls Description

The interaction of the space suited pilot and the vehicle while engaged in IVA and EVA requires consideration with regard to displays and controls which must be designed to be accessible, operable and communicative. Any Spaceplane manual controls (e.g., levers, buttons, switches) must be designed to be compatible with the pilot's pressurized glove geometry and dexterity limits. The use of voice control and optical system heads-up-displays integrated into the pilot's helmet system is recommended and offers many advantages in performance, flexibility and weight and volume savings.

Manual Controls

Views of the Spaceplane cockpit displays and controls panel are shown in Figures 4-26 to 4-28. The panel is located so that the space suited pilot can reach and operate controls under the full range of potential mission g-loads. With the hatch open and the couch deployed on-orbit, the panel is designed to rotate up to provide the pilot with visibility of displays in that deployed position.

All displays must be situated and illuminated to provide clear visibility to the space suited pilot in the extreme lighting conditions of space. Manual EC/LSS controls consist of knobs that can be rotated with the pressurized EMU glove and toggle switches and push buttons providing the pilot access to the onboard computer. The displays might include an LCD display to verify computer entry data and AC-plasma or CRT displays for mission data. AC-Plasma displays are now under development for military tactical systems. A one meter (diagonally) flat panel AC-plasma display has a low electrical drive power requirement and can be driven by a small battery. An AC-plasma display only 4 inches deep including all required drive electronics could provide a resolution of over 50 pixels per linear inch.

Current research efforts are enhancing the capabilities of AC-plasma devices. A three-meter (diagonal) display is being developed with an unprecendented 125 pixels per linear-inche resolution. Also, work is being done to extend color capabilities from monocolor orange to multicolor red, blue, and green. This display technology should be available for Spaceplane application.

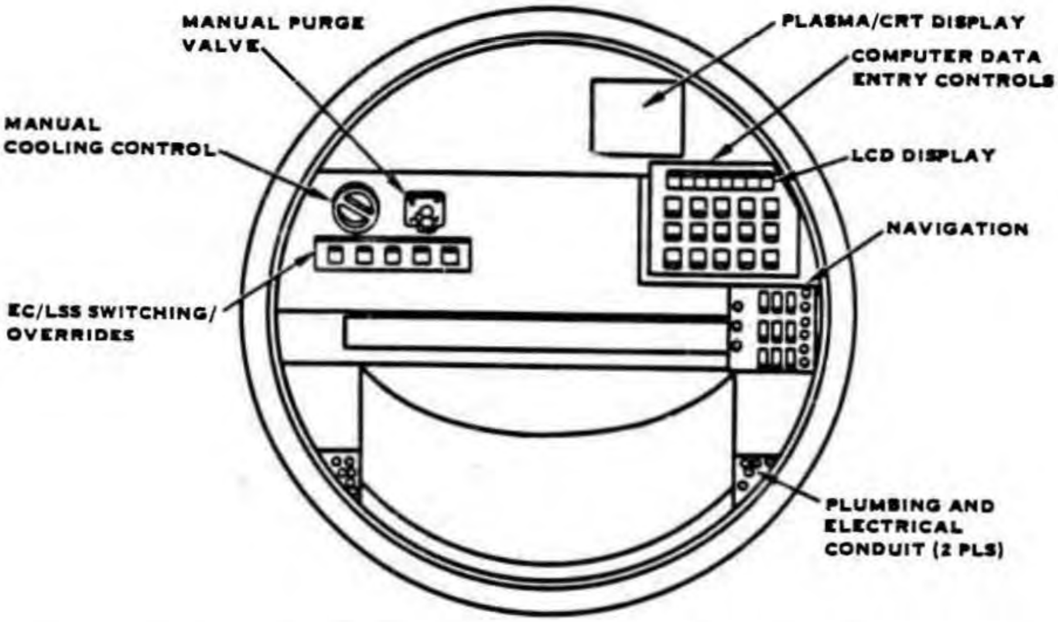


Figure 4-42. **SPACEPLANE CONTROLS AND DISPLAYS PANEL**

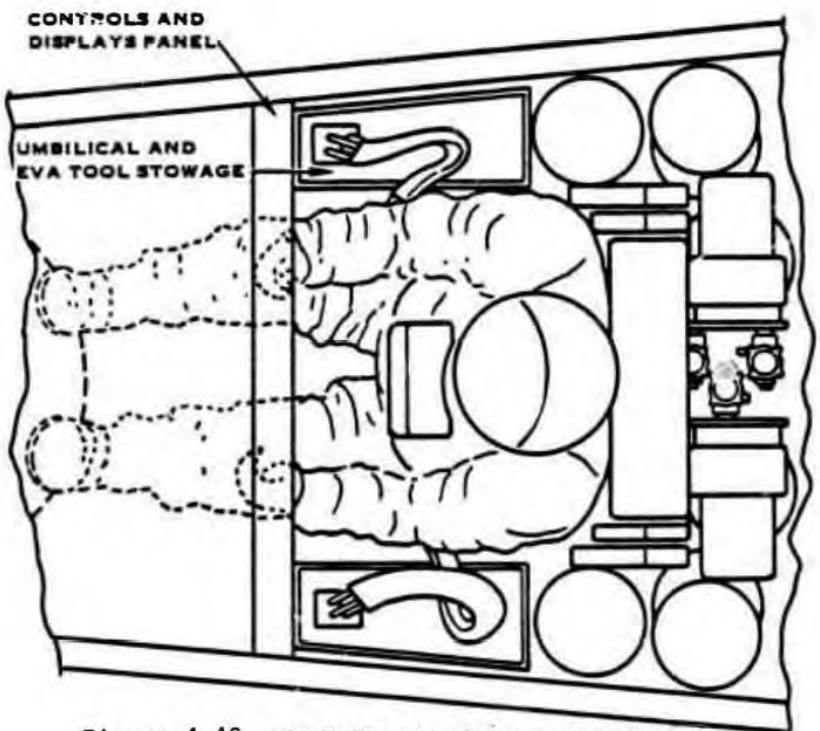


Figure 4-43. **SPACEPLANE COCKPIT (TOP VIEW)**

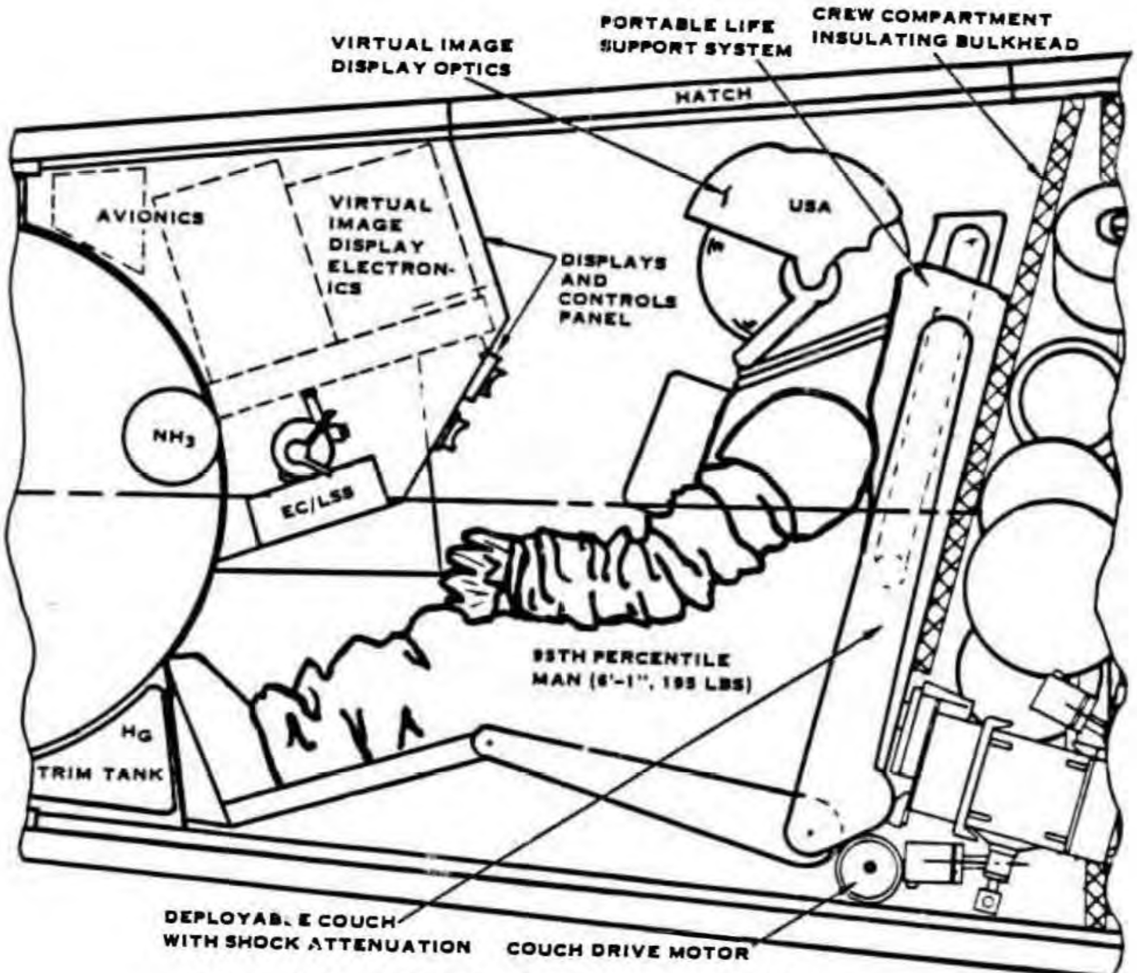


Figure 4-44. **SPACEPLANE COCKPIT (SIDE VIEW)**

Voice Controller - A voice controller for the Spaceplane pilot to control subsystems would offer mission flexibility. For example, while conducting EVA the pilot could override manual controls with the voice controller and thereby continue to control Spaceplane systems. Use of a voice controller would allow both of the pilots hands to be free for EVA work activity.

Through recent advances in semiconductor technology, the economics of speech recognition are beginning to improve rapidly including extending speech to man-machine interactions. A new generation of lower-priced speech-recognition modules aimed specifically at original equipment manufacturers is coming to market and, for the first time, dozens of companies are looking at building this capability into their products. Speech recognition has until recently been an experimental effort, but the technology is now ready for commercialization.

The signal flow for a Spaceplane voice controller is shown in Figure 4-29. Sounds are converted into digital codes and stored in the computer memory. When a user then utters a command, the digital code for the new sound is compared with the stored data until a close match is found. Accuracies in laboratory tests often approach 99%; however, performance may drop if the environment is noisy or if the speaker is subjected to a stressful situation such as high g-loads. Work is being done to understand the effects on speech of stressful activity.

Semiconductor advances are making speech recognization technology available at much lower cost. As thousands more circuit functions are packed onto solid state chips of silicone, the costs of processing the speech signal and storing the results are dropping. It will soon be possible to replace hundreds of individual components with a single chip for converting voice sounds into data.

Heads-Up Display

The Spaceplane pilot will be required to perform a myriad of control and information processing functions by interacting with an assortment of cockpit controls and displays. The pilot will have to attend to information from flight systems (e.g., engine status, fuel, altitude, attitude, velocity, etc.), as well as from navigation, communications, radar, and imaging sensor systems. Simultaneously, he must initiate control actions to direct various subsystems and payloads. Many of these operations must be performed and directed from inside the cockpit with no direct visual contact with the outside environment. While digital avionics systems have provided some degree of automation, it has been at the cost of an increase in the amount of information to be displayed and in the control options available to the pilot.

Since 1966, the USAF Aerospace Medical Research Laboratory has been developing a new family of control/display devices collectively termed "visually-coupled

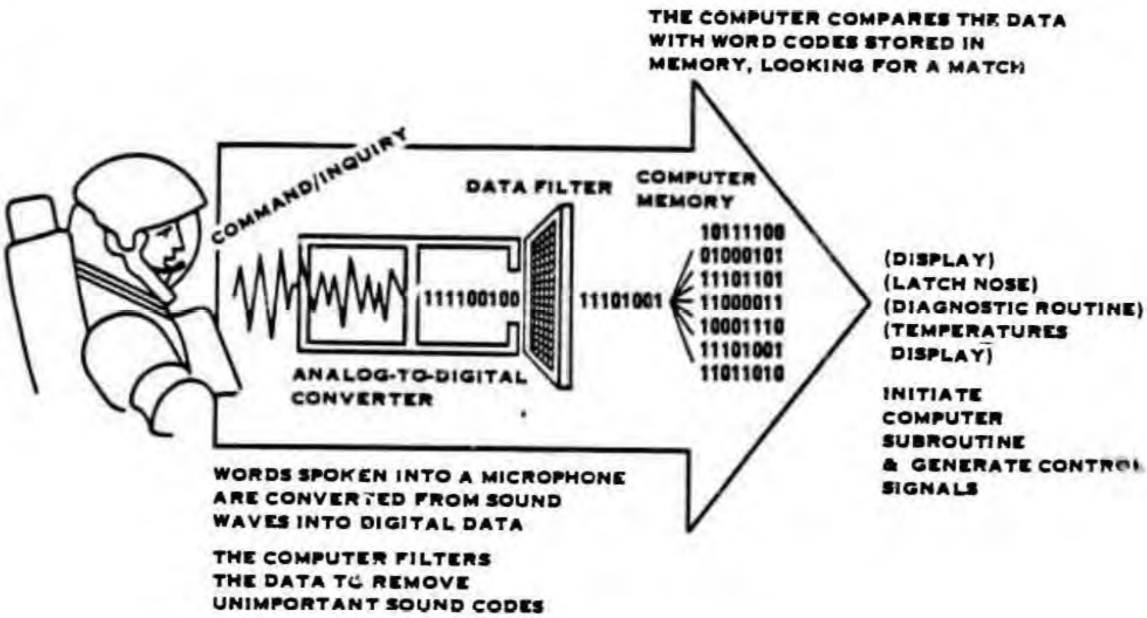


Figure 4-45. **VOICE CONTROLLER SIGNAL FLOW**

systems." This approach takes advantage of the precision with which a pilot can aim his head and direct his gaze. In essence, the interface between the crewmember and the vehicle systems is brought about through the communication of head (and consequently eye) position coordinates in order to designate objects or sensors or activate switches. The two components which comprise visually-coupled systems are: 1) the helmet-mounted sight, which provides line-of-sight data; and 2) the helmet-mounted display, which provides feedback via displayed information.

The helmet-mounted sight would make use of an aiming reticle which, mounted to the pilot's head, is collimated and projected into the crewmember's eye by an optical assembly integrated with the EMU flight helmet. The operator can position the reticle over objects of interest by simply moving his head. The system line-of-sight is determined by measuring the orientation of the helmet in the cockpit. The output signals of the helmet-mounted sight can be used to select various devices and controls in the Spaceplane cockpit. This select control device could also be activated using the voice controller described above. The pilot can purposefully shift his line-of-sight to labeled control surfaces in order to initiate various Spaceplane subsystem functions (e.g., selecting temperature, display modes, etc.). For example, he need only position the helmet-mounted reticle over a symbol box designated "temperature status" in order to receive EC/LSS temperature information.

Although the Honeywell electro-optical system represents a first generation helmet-mounted sight, such a system is capable of measuring helmet orientaton with an accuracy of better than 1 degree virtually anywhere in the cockpit.

The helmet-mounted display presents a virtual image which can be viewed continuously, regardless of head orientation. Figure 4-30 depicts the operation of the Honeywell helmet-mounted display. Pictorial and/or symbolic information is imaged by a miniature, high resolution cathode ray tube integrated with EMU helmet. The active area of the CRT is approximately 19mm in diameter, with a limiting resolution of 1,000 TV lines at a highlight luminence of 1700 cd/m2. The CRT image is magnified and collimated by optical elements on the helmet. It is then projected into the eye of the pilot via a combiner which is integrated into the helmet flight visor. The operator perceives a wide field-of-view image which appears at optical infinity. The fields-of-view of current helmet-mounted displays are up to 40 degrees, being limited in practice by the size of the optical elements. The exit pupil is typically 14-16mm, with an eye-relief of approximately 25-40 mm.

There are several important advantages of visually-coupled systems for Spaceplane application. As described earlier, the EMU helmet-mounted sight allows the pilot to head track/aim assuring a quick reaction time, with no learned skills required. Additionally, the line-of-sight data can provide the pilot a heads-up view external to Spaceplane as seen by an external camera or TV monitor. The effect perceived by the pilot is analogous to viewing the outside space environment through a movable window.

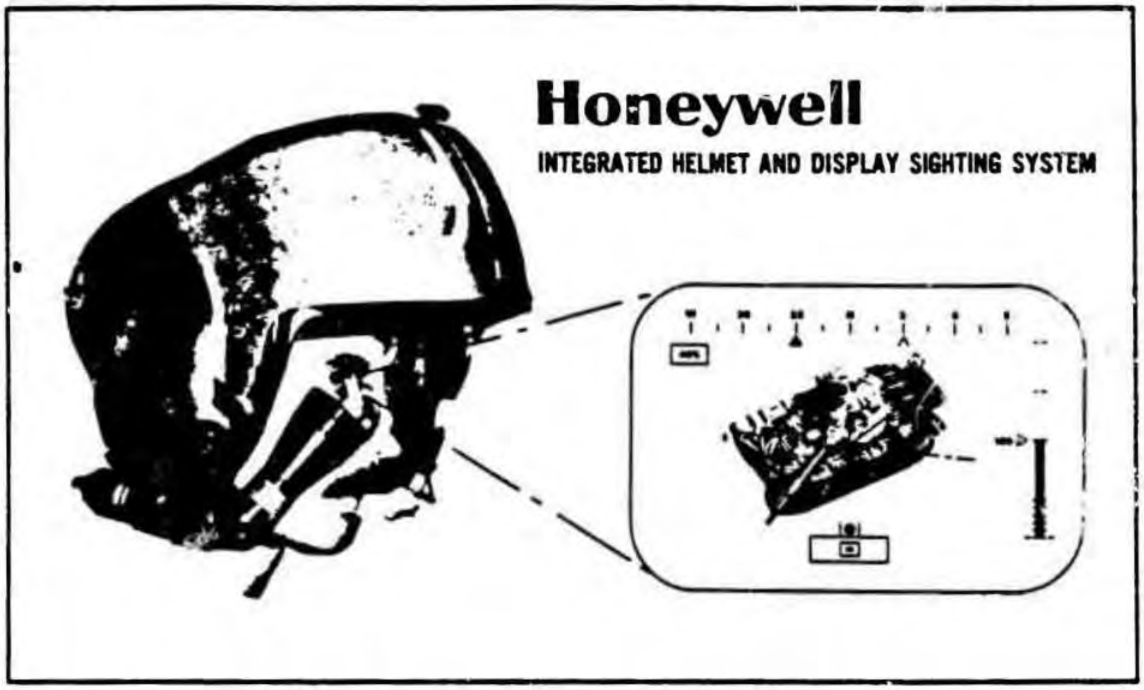


Figure 4-46

The helmet-mounted display offers the additional advantage of a presentation of visual information regardless of the operator's direction. For example, when on EVA, the pilot could take along the cockpit control panel in the form of a computer-driven virtual image. Head-up display-type symbols plus sensor images can be presented simultaneously. Since the optics form a virtual image, the image angle is greater than that which can be achieved with panel-mounted displays. Additionally, more information elements can be conveyed to the operator than using conventional panel-mounted displays. This distinction is very important in the display of sensor imagery which is usually limited in quality by poor resolution. Another key advantage of the helmet-mounted display in the crew station design is that it occupies little space in the small volume Spaceplane cockpit.

Symbolic images which normally appear on a heads-up display in fixed locations relative to the airframe can also be presented on the helmet-mounted display. However, the symbols can be made to appear stationary relative to Spaceplane and actually move in and out of the field-of-view of the helmet-mounted display as the pilot moves his head beyond a certain position. There are three advantages to a "visually-coupled" heads-up display. First, the instantaneous field-of-view of the helmet-mounted display is much larger than that of most head-up displays; therefore, more information can be displayed. Second, the visually-coupled system will not require valuable Spaceplane cockpit area required by a conventional head-up display. Third, additional display modes are possible. One of these modes can be termed a "wrap-around" or hemispherical display. It is possible, for example, to present the pilot with a continuous artificial horizon which is super-imposed over the real horizon and which can be viewed at any azimuthal angle.

4.5.8.3 Displays and Controls Hardware Specification

The displays and controls voice controller and heads-up display hardware described in the previous sections must satisfy the following specifications.

Operating Life

The unit shall be designated for a continuous operating life of (TBD) years.

Physical Characteristics

Vocabulary (TBD)
Data Rate Maximum (TBD)
Resolution (TBD)

Weight

The unit weight shall not exceed (TBD)

Power

The unit power requirement shall not exceed (TBD)

Flight Environment

The unit shall be capable of meeting the operating requirements as specified herein at the following environmental conditions:

Ambient Environment

Atmosphere	Air to space vacuum
Pressure	15.23 psia to 1 x 10-10 torr
Temperature	-25 to +122°F

(Conductive/Convective/Radiation Environment)

Acceleration

±10 g's in any attitude

Humidity

Dew Point Temperature	0 to 84°F

(Relative humidity can reach 100%)

Acoustic and Random Vibration

Consistent with Orbiter payload bay levels

4.5.9 Support Equipme

Equipment will be required to support Spaceplane EC/LSS during launch phases and during post mission activity. This equipment is launch mode dependent and falls into two categories - Ground Support Equipment (GSE) and Flight Support Equipment (FSE). The DRM launch mode assumes a Spaceplane launch from the Space Shuttle. FSE equipment required to support the Shuttle launch mode, as well as that for alternative modes, is discussed below. GSE for post mission activity is also discussed.

4.5.9.1 Ground Support Equipment (GSE) Specification

The GSE must support EC/LSS aboard the Spaceplane throughout prelaunch and post-mission phases. The GSE must support Spaceplane during ground service and checkout operations and final checkout in the Shuttle payload bay while in the Vehicle Assembly Building at the launch site. The GSE must satisfy the following specification·

1. Provide Avionics/payload cooling during service and checkout of systems.

2. Recharge of EC/LSS subsystem consumables (water, ammonia, oxygen).

3. Recharge of EMU consumables (waste management devices, water, oxygen, LiOH, drinking water, foodsticks).

4.5.9.2 GSE Description

A GSE cart is shown in Figure 4-47 for use in ground service of Spaceplane. A schematic for the GSE for Spaceplane to replenish vehicle EC/LSS consumables of water, ammonia and oxygen is shown in Figure 4-48. As shown, the GSE would include a water accumulator supply reservoir pressurized with nitrogen, a high pressur oxygen recharge system and a nitrogen pressurized liquid ammonia system. schematic for GSE cooling, which employs a Freon-12 loop and condenser cooled by ambient air is shown in Figure 4-48c. The interface with the
Spaceplane EC/LSS subsystem uses umbilicals connected to two service disconnect ports in the EC/LSS.

The GSE described in the preceeding paragraph could also be used for service of the EMU. The high pressure oxygen source would replenish oxygen in the PLSS and the pressurized water source would replenish PLSS sublimator water. Waste management service would involve the exchange of insuit waste collectors (vomitus, feces, urine). In addition, foodsticks, drinking water and LiOH cannister would be replaced.

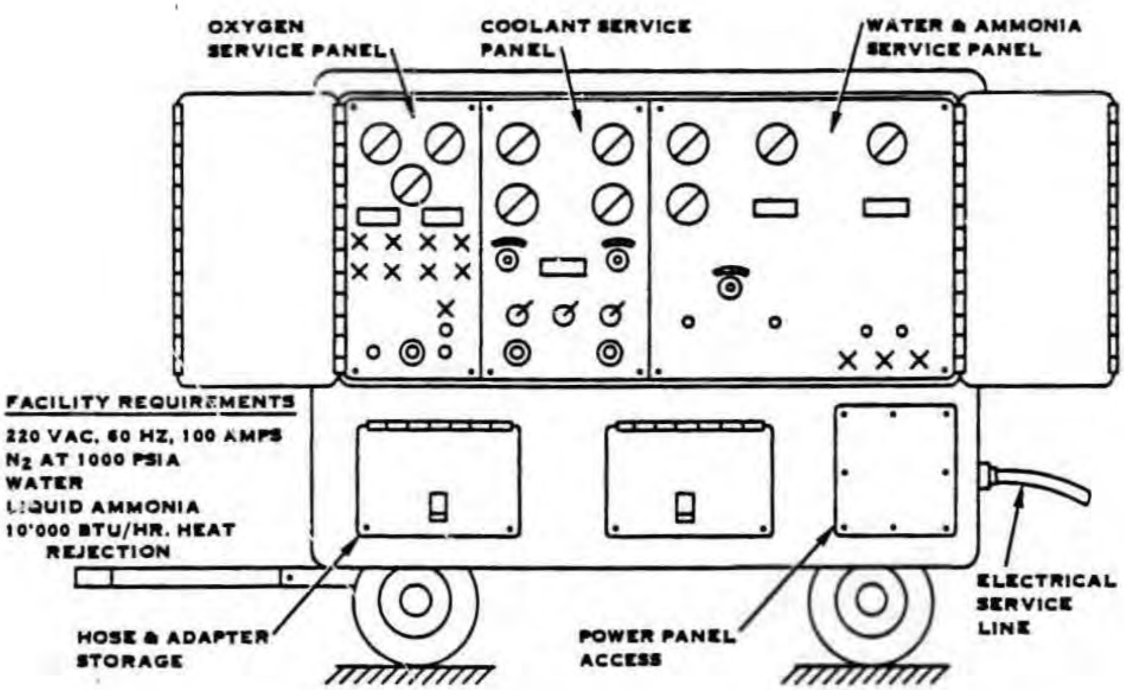


Figure 4-47. **GSE CART FOR SPACEPLANE**

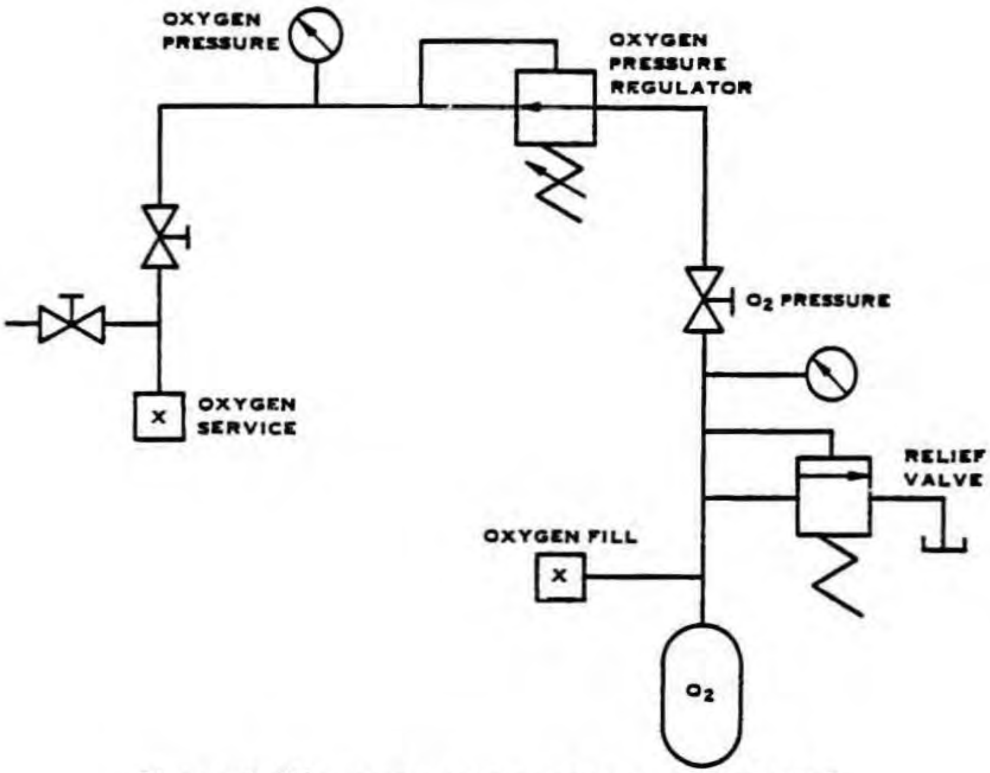


Figure 4-48A. **GSE RECHARGE SCHEMATIC (OXYGEN)**

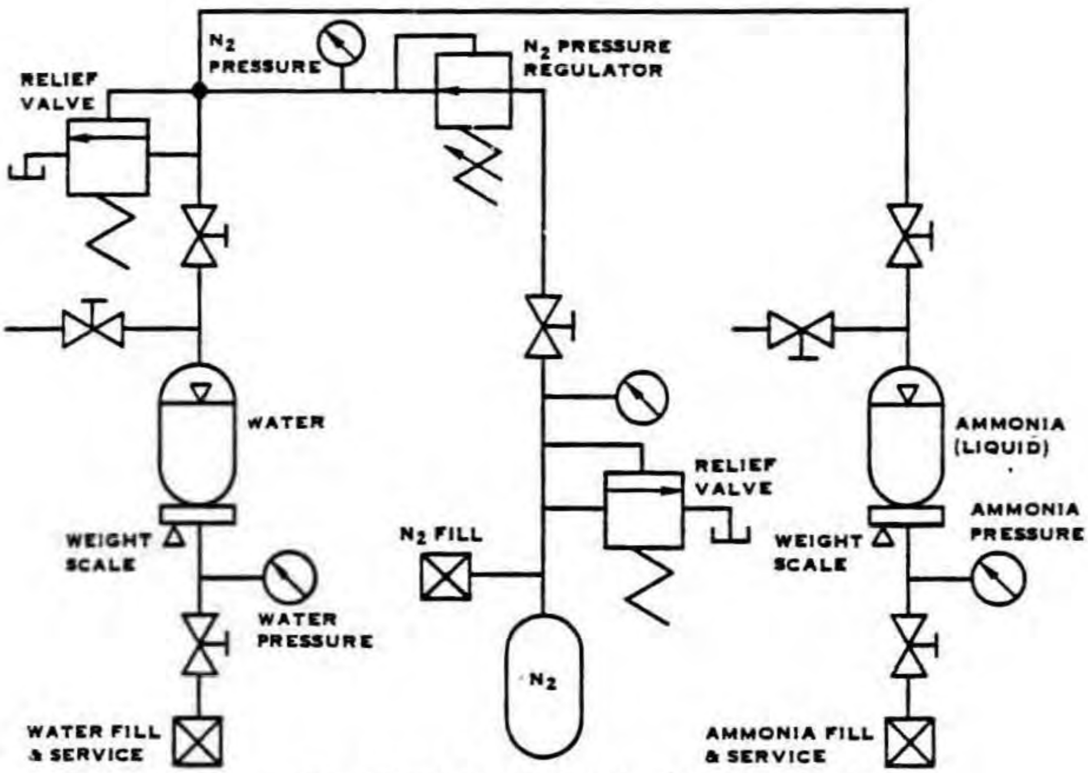


Figure 4-48B. GSE RECHARGE SCHEMATIC (WATER, AMMONIA)

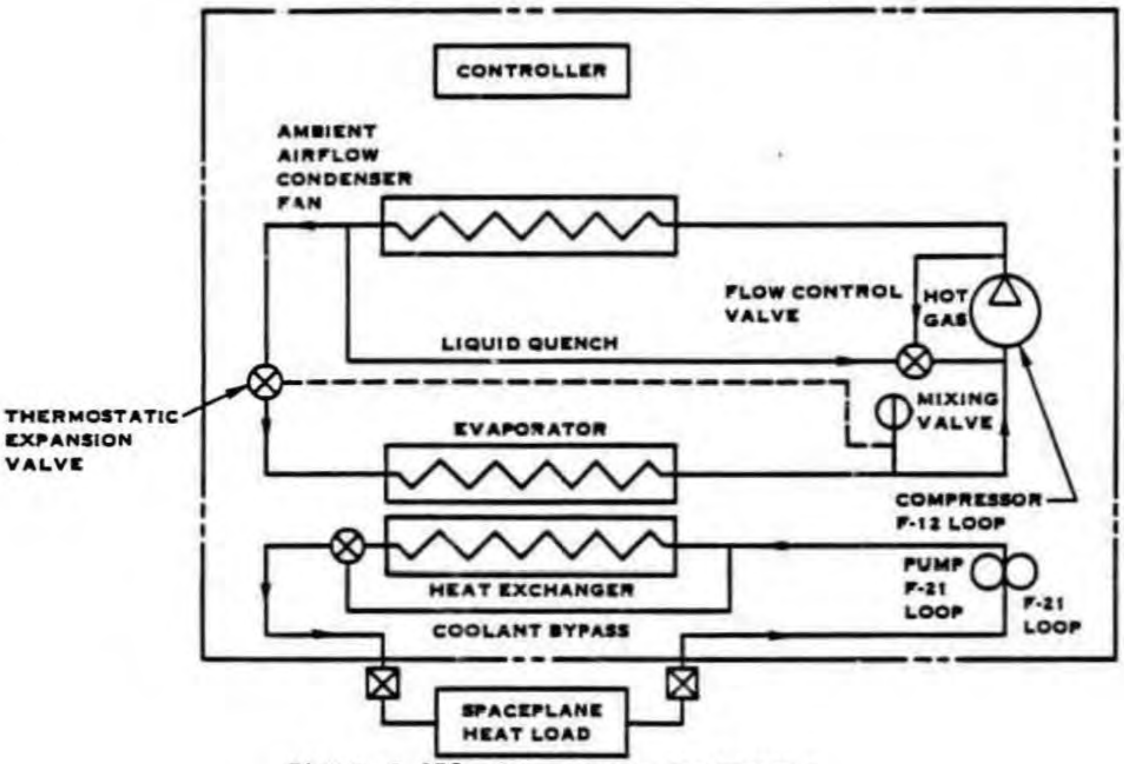


Figure 4-48C. GSE COOLING SCHEMATIC

4.5.9.3 GSE Hardware Specification

Specification for the GSE equipment is as follows:

EC/LSS Subsystem GSE Service/Checkout Specification

Water:

Temperature	-	34F
Flow	-	240 lb/hr

Recharge Consumables

Water:

Weight	-	300 lbs
Supply Pressure	-	0-1000 psia

Ammonia:

Weight	-	20 lbs
Supply Pressure	-	(TBD)

Oxygen:

Weight	-	20 lbs
Supply Pressure	-	3000 psia

4.5.9.4 Flight Support Equipment Specification (FSE)

During earth-to-orbit Shuttle launch and prior to Spaceplane deployment from the Shuttle payload bay, it is assumed that Spaceplane Avionics/payload thermal control will be supplied by the Space Shuttle active thermal control system through an umbilical interface provision. However, FSE will be required to support service of of the EC/LSS aboard the Spaceplane between Spaceplane sorties operating from the Space Shuttle.

The FSE must satisfy the following specification:

1. Provide on-orbit recharge of Spaceplane EC/LSS subsystem consumables (water, ammonia, oxygen)

4.5.9.5 FSE Description

A sketch of a FSE kit for on-orbit service of Spaceplane is shown in Figure 4-49. The kit would include a Remote Manipulator System (RMS) grapple pin for transport of the kit in the Shuttle payload bay to the Spaceplane during service operations. A schematic for FSE designed to replenish EC/LSS consumables is shown in Figure 4-50. As shown, the FSE, similar to the GSE cart, would include provision for recharging Spaceplane water, ammonia and oxygen in the Spaceplane EC/LSS. Service of the EMU would be accomplished

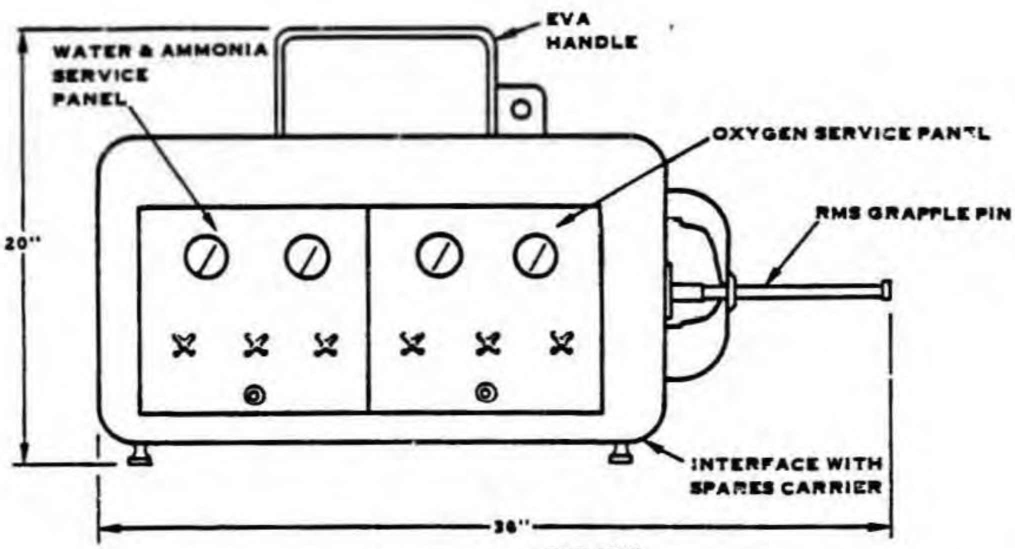


Figure 4-49. FSE KIT

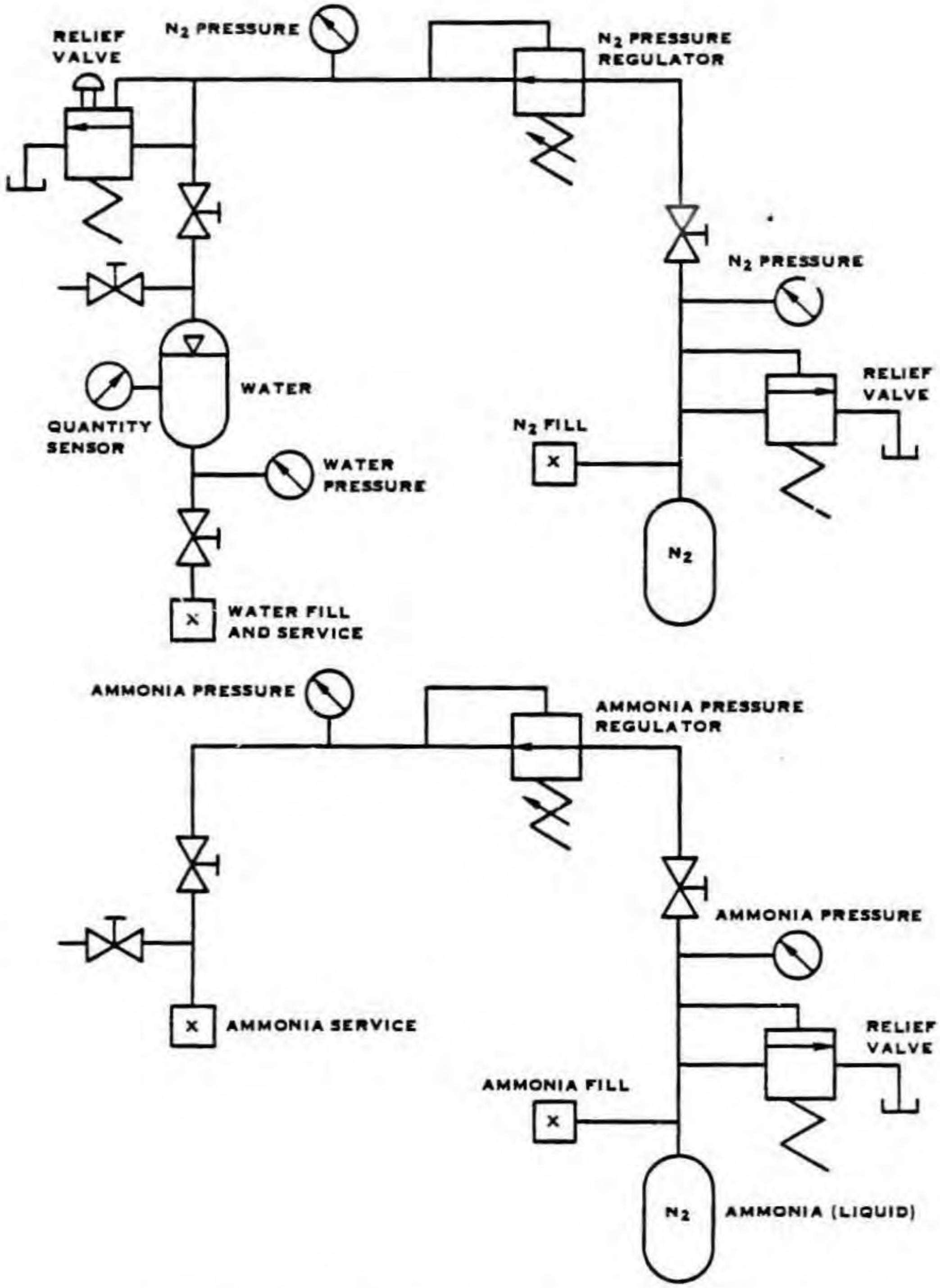


Figure 4-50A. **FSE RECHARGE KIT (WATER, AMMONIA)**

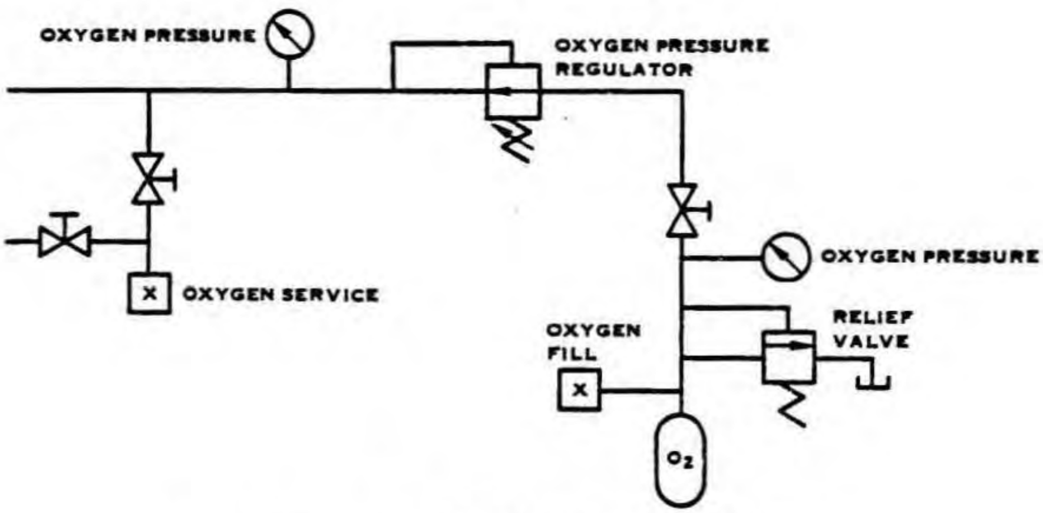


Figure 4-50B. **FSE RECHARGE KIT (OXYGEN)** using the existing Shuttle Airlock EMU service provisions. Between Spaceplane sorties operating from the Shuttle replacement LiOH canisters for Spaceplane would be stored in the Shuttle payload bay.

4.5.9.6 FSE Hardware Specification

Specification of FSE equipment is as follows:

Recharge Consumables

Water:

Weight	-	(TBD)
Supply Pressure	-	0-1000 psia

Ammonia:

Weight	-	(TBD) lb
Supply Pressure	-	1000 psia

Oxygen:

Weight	-	(TBD) lb
Supply Pressure	-	3000 psi

Flight Environment

The unit shall be capable of meeting the operating requirements as specified herein at the following environmental conditions:

a. Ambient Environment

Atmosphere	Air to space vacuum
Pressure	15.23 psia to 1 x 10-10 torr
Temperature	-25 to +122°F

(Conductive/Convective/Radiation Environment)

b. Acceleration

± 3 g's in any attitude

c. Humidity

Dew Point Temperature 0 to 84°F

(Relative humidity can reach 100%)

d. Acoustic and Random Vibration

Consistent with Orbiter payload bay levels.

4.6 SPACEPLANE TECHNOLOGY PROGRAM PLAN

This program plan for the Spaceplane Technology Program was developed in conjunction with the preceding studies. The program, based on our previous EC/LSS experience on the Shuttle orbiter and the EMU and recent EVA studies, permits us to offer a complete and efficient program that could produce flight-qualified hardware for use in the Spaceplane within 2 years from program go-ahead.

4.6.1 Objective

The objective of the Hamilton Standard program described herein is to define, design, manufacture, test, and deliver flight-qualified technology items. Elements of this task include:

- 8 psi EMU
- EMU with Detachable PLSS
- EMU Solid Waste Systems
- Stacked Evaporator
- EMU Heads-Up Display

Part of our objective is the utilization of as much existing Shuttle and EMU hardware as possible, reducing overall program costs thereby and allowing near-term completion of the program as planned. The write-ups that follow are typical of the development programs required for each of the above items.

4.6.2 Work Breakdown Structure

Figure 4-51 defines the typical work breakdown structure (WBS) that would be used for each technology item. The figure represents the major program phases and the elements over which program financial control will be exercised. For progress and technical activity monitoring, the task elements will be further detailed in the form of technical status schedules.

Although all other elements will be performed solely by Hamilton Standard, the program for the EMU Heads-Up Display will be performed by a subcontractor. The system integration and WBS Tasks 1.0, 2.0, and 5.0 will be mutual efforts, with the subcontractor performing WBS Tasks 3.0 and 4.0.

4.6.3 Program End Products

The end products of these programs will include documentation described herein and delivery of one each of the technology items.

4.6.4 Program Schedule

The typical schedule for accomplishing the task elements identified in the WBS is presented in Figure 4-52. Also identified are the major program milestones. Figure 4-53 describes the program logic with the relationship between various tasks.

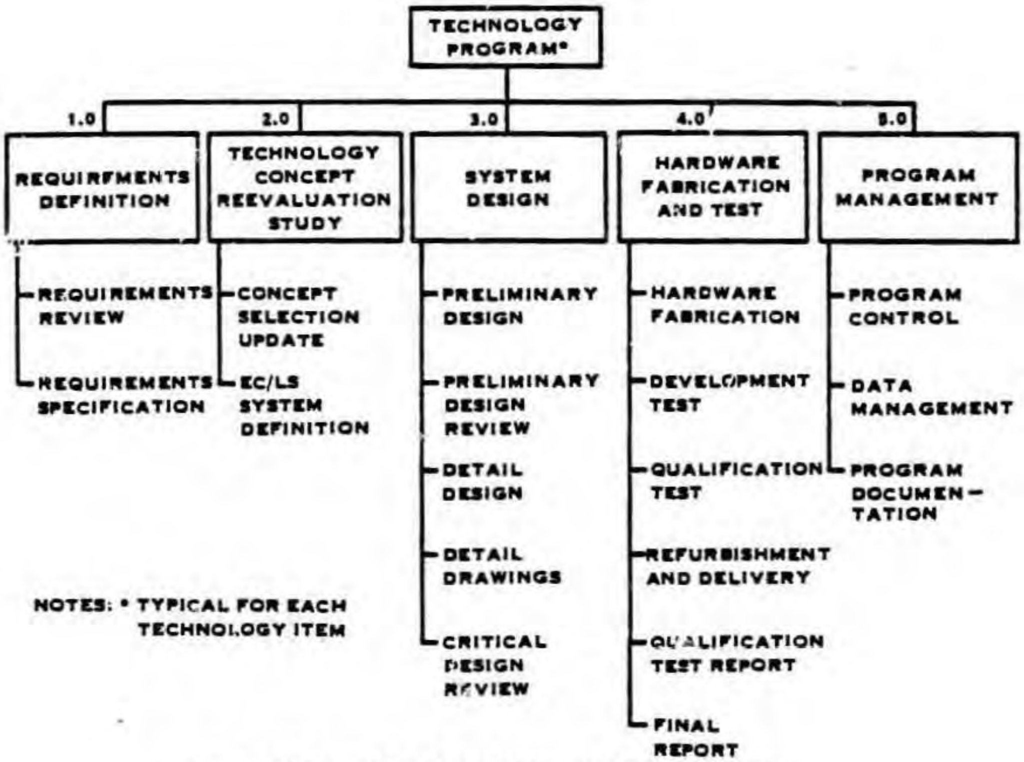


Figure 4-51. WORK BREAKDOWN STRUCUTRE

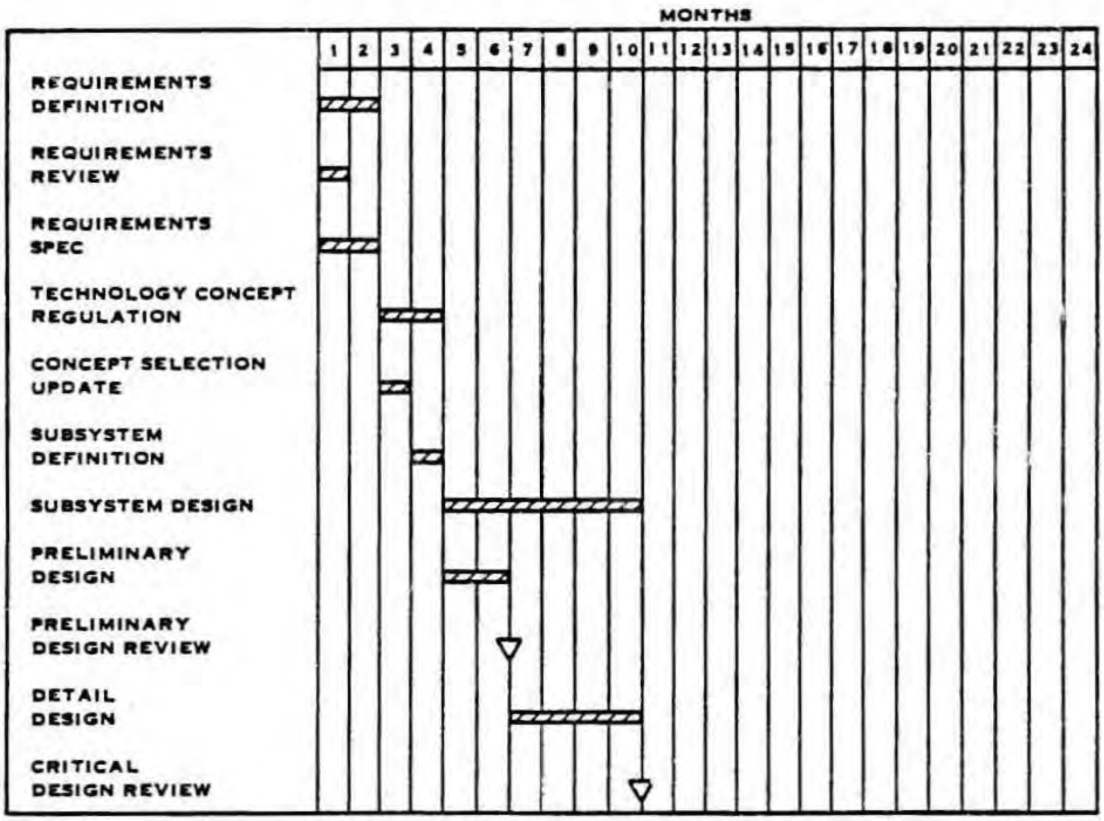


Figure 4-52. TECHNOLOGY SCHEDULE

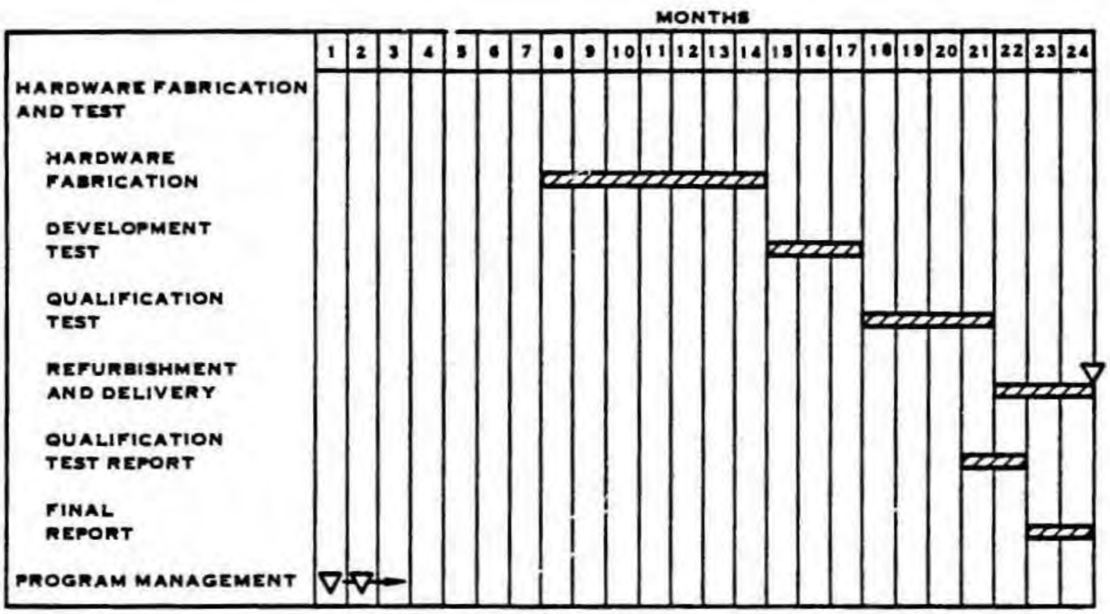


Figure 4-52 (cont.) TECHNOLOGY SCHEDULE (CONT'D)

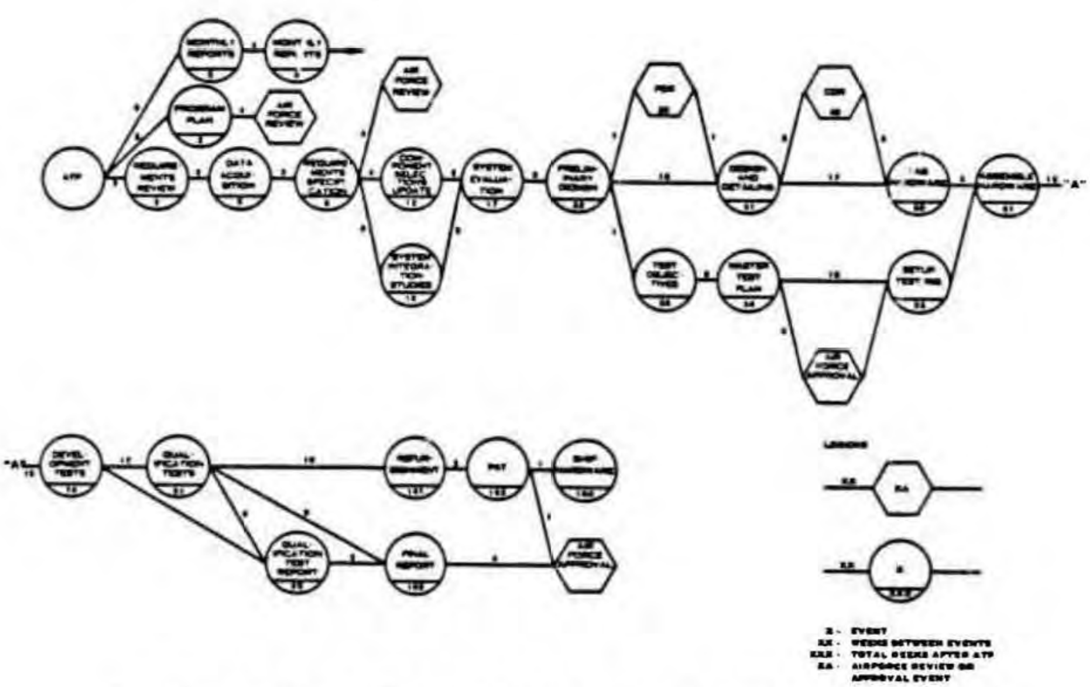


Figure 4-53. **NEW TECHNOLOGY ITEMS LOGIC FLOW**

4.6.5 Task Descriptions

As previously stated, Hamilton Standard will conduct the technology programs in accordance with the WBS defined in Figure 4-51. All in-house financial data collection will be against WBS task element effort. In addition, to assist in the day-to-day administration of the program, each WBS element will have a detailed technical status schedule. The preparation of these schedules is the responsibility of the program project engineer. The schedules will be reviewed on a continuous basis by Program Management. All active technical status schedules will be included in the first monthly progress report and updated on a monthly basis thereafter as part of the monthly report.

The following discussion describes each of the WBS task elements and the program flow in accordance with the program logic chart (Figure 4-53).

4.6.5.1 Requirements Definition

Under this task, the complete requirements for the EC/LSS will be documented in a design specification. This specification would be common to technology items at the system level but, at the subsystem level, would be specific.

4.6.5.2 Requirements Review

For this task element, all available information and data, including mission definitions and alternates, vehicle environmental input to subsystem, operational requirements, contingency and docking considerations, EVA requirements, and other demands impacting the technology item requirements, will be used. To obtain this information and to better understand mission requirements and vehicle interfaces, discussions and meetings will be held with appropriate organizations, as required, during the program.

4.6.5.3 Requirements Specification

Based on the requirements review, a detailed specification for each technology item will be generated. Quantitative requirements in the specification will include data from the Preliminary Spaceplane Study, mission definition, vehicle, crew, equipment, cabin environment, EVA, and other considerations. Qualitative requirements will include reasonable design objectives, based on previous spacecraft and spacesuit experience and Spaceplane Study results.

Upon completion of this program element, the design specification generated will be reviewed with the contract technical monitor and used as a basis for the conceptual design of the technology item.

4.6.5.4 Technology Concept Evaluation Study

In this phase, the technology concept selection will be reevaluated, together with all Spaceplane-related items, against the Requirements Specification, to define a complete EC/LSS.

4.6.5.2.1 Concept Selection Update

The selected technology concept will be reevaluated against the Requirements Specification to assure that all mission objectives can be met. Every effort will be made to continue to utilize existing technology and space-qualified hardware in the resulting design to minimize both the lead-time and total program costs.

4.6.5.2.2 Technology Subsystem Definition

Utilizing the concept selected above, a complete technology subsystem will be defined. Included in this definition will be the integration of all the new technology items and all ancillary equipment, including EVA and support equipment to assure all interfaces are compatible within the Spaceplane during all mission phases. Also to be considered is the acceptability to the crew member and such factors as safety, reliability, maintainability, and commonality.

4.6.5.3 Subsystem Design

Under this task, the technology items will be designed, detail drawings prepared as required, and a preliminary design review (PDR) and a Critical Design Review (CDR) held.

4.6.5.3.1 Preliminary Design

The design phase effort is initiated in this task element. The objective is selection of an optimum design configuration, including system level performance, safety, weight, volume, reliability, maintainability, and consideration of interface requirements utilizing design layouts, packaging drawings, etc. Recommendations concerning applicable flight controls and displays and ground-test instrumentation will be made. Thermal and performance analyses required, as applicable, to evaluate each selection will also be made to assure proper system integration at the detail level.

4.6.5.3.2 Preliminary Design Review

A Preliminary Design Review will be held to review all work completed to date and to assure agreement on all elements of the design before it is initiated. Personnel from the Air Force and the contractor are expected to attend this review.

4.6.5.3.3 Detail Design

The design task element covers the generation of detailed technical definitions for the system from the configuration concept approved in the design-setting phase. The primary effort is involved in preparation of the detail and assembly drawings for fabrication, procurement, and final assembly of all elements of the subsystem. The effort will also include the additional functional, thermal, and structural analyses of the system to assure compliance with the requirements of the SOW. The task will terminate with conduct of the CDR and incorporation of CDR comments resulting in the approval of the engineering drawings.

Hamilton Standard will conduct a materials control program to identify all nonmetallic materials to ensure compatibility with the environment in the subsystem. This effort will result in the generation of the Nonmetallic Materials List and corrosion-protection summary to be included in the Final Report. Although the effort is scheduled for completion by CDR, on-going overview will be maintained by Materials Engineering throughout the duration of the program.

Intrinsic reliability is a system characteristic that Hamilton Standard recognizes as being of major importance. All design-phase effort will include inputs from our Reliability Engineering Group to assure attainment of the maximum practical reliability. The major effort for this task element will be preparation of the Failure Mode and Effect Analysis. A preliminary analysis willbe available at CDR.

4.6.3.4 Critical Design Review

Upon completion of the design effort, a Critical Design Review will be held with the representatives from the Air Force and the contractor. At this review, the final design will be reviewed in detail for comments and/or changes. Drawings for selected long-lead items may be released prior to this meeting in order to maintain schedules.

4.6.5.4 Hardware Fabrication and Test

This task includes all effort and material necessary to manufacture, assemble, and test hardware for this program. Also included is the effort to refurbish the hardware prior to delivery to the customer.

4.6.5.4.1 Hardware Fabrication

This fabrication phase includes all effort and materials necessary to manufacture and assemble one complete set of hardware for this program. The phase also includes inspection of raw materials, fabricated hardware, and the complete assemblies. Included is the Manufacturing Engineering and Control effort necessary to prepare all manufacturing documentation and to expedite the processing of the hardware. Because it is planned to deliver this hardware to the customer, all necessary controls and paperwork will be inspected and documented.

4.6.5.4.2 Development Tests

This task element includes the effort to conduct development tests to evaluate the effect of modifications on existing hardware and to establish the performance of all items in the mission operating range. Also included is the effort for the special test equipment to conduct the above tests, as well as any qualification and production acceptance tests.

A master test plan defining the complete test program will be prepared under this task element. It will be submitted to the contractor for comment and approval at least 90 days before testing is to begin.

4.6.5.4.3 Qualification Tests

Within this task element, verification of the qualification test requirements will be accomplished, either by actual testing or by documentation that shows that existing or modified flight hardware is capable of meeting the requirements defined in the master test plan.

4.6.5.4.4 Refurbishment and Delivery

This task element covers the effort necessary to inspect the unit thoroughly after completion of the subsystem testing, to perform any required refurbishment and to prepare the unit for shipment. Because there are no limited-life items in the subsystem, this effort is expected to be minimal. Acceptance tests will be conducted on all items prior to delivery.

4.6.5.4.5 Qualification Test Report

Upon completion of the qualification testing, a qualification test report, fully documenting all test data and conditions, will be prepared. The report will include photographs of the test setups and hardware.

4.6.5.4.6 Final Report

A final report documenting all activity accomplished during this program will be prepared. It will include, as appendices, the various test plans and reports generated during the program. The report will be submitted to the contractor for his review and approval.

4.6.5.5 Program Management

The objective of this work task is to provide the manpower necessary to direct, manage, administer, control, and report upon Hamilton Standard's performance of the Spaceplane Technology Program.

4.6.5.5.1 Program Control

An accounting and scheduling system built upon the WBS will be established and included as a part of the first monthly progress report. Resource budgets will also be established and maintained, with actual expenditures identified in all subsequent monthly progress reports. Progress against the predetermined milestones will be maintained continuously. Program control information in the form of the Technical Status Schedules will be a part of the monthly report. Monthly Progress Report and Monthly Financial Management Report will be prepared and submitted in accordance with contract requirements.

4.6.5.5.2 Data Management

Under this WBS task element, the existing Hamilton Standard data management capability will control the collection, preparation, publication, quality, assessment, distribution, and maintenance of all data items, in accordance with the contract Statement of Work.

4.6.5.5.3 Program Documentation

This task element provides effort for preparation of that documentation required by the Statement of Work Data Requirement List (DRL) and not specifically prepared under other task elements.

4.7 SPACEPLANE EC/LSS PROGRAM PLAN

This program plan for the Spaceplane Environmental Control and Life Support System (EC/LSS) was developed in conjunction with the preceding studies. The program, based on our previous Shuttle and EMU EC/LSS experience and recent EVA studies, permits us to offer a complete and efficient program that could produce flight-qualified hardware for use in the Spaceplane within 3 years from program go-ahead, assuming a 1-year head start on the Spaceplane Technology Items.

4.7.1 Objective

The objective of the Hamilton Standard program described herein is to define, design, manufacture, test, and deliver a flight-qualified EC/LSS, essentially as defined in Figure 4-54, and a couch for use in the Spaceplane. Also included in this program are vehicle EC/LSS control and display requirements, flammability control and suppression hardware, necessary ground and flight support equipment, and integration of new technology items into the Spaceplane.

Part of our objective is the utilization of as much existing Shuttle and EMU hardware as possible to reduce overall program costs, assuring completion of the program as planned.

4.7.2 Work Breakdown Structure

Figure 4-55 defines the typical work breakdown structure (WBS) that represents the major program phases and the elements over which program financial control will be exercised. For progress and technical activity monitoring, the task elements will be further detailed in the form of technical status schedules.

4.7.3 Program End Products

The end products of these programs will include documentation described herein and delivery of one EC/LSS and one couch, which will be refurbished prior to delivery to the Air Force.

4.7.4 Program Schedule

The schedule for accomplishing the task elements identified in the WBS is presented in Figure 4-56. Also identified are the major program milestones. Figure 4-57 describes the program logic with the relationship between various tasks. As noted, initiation of this program is paced by definition of five (5) new technology items. These items are:

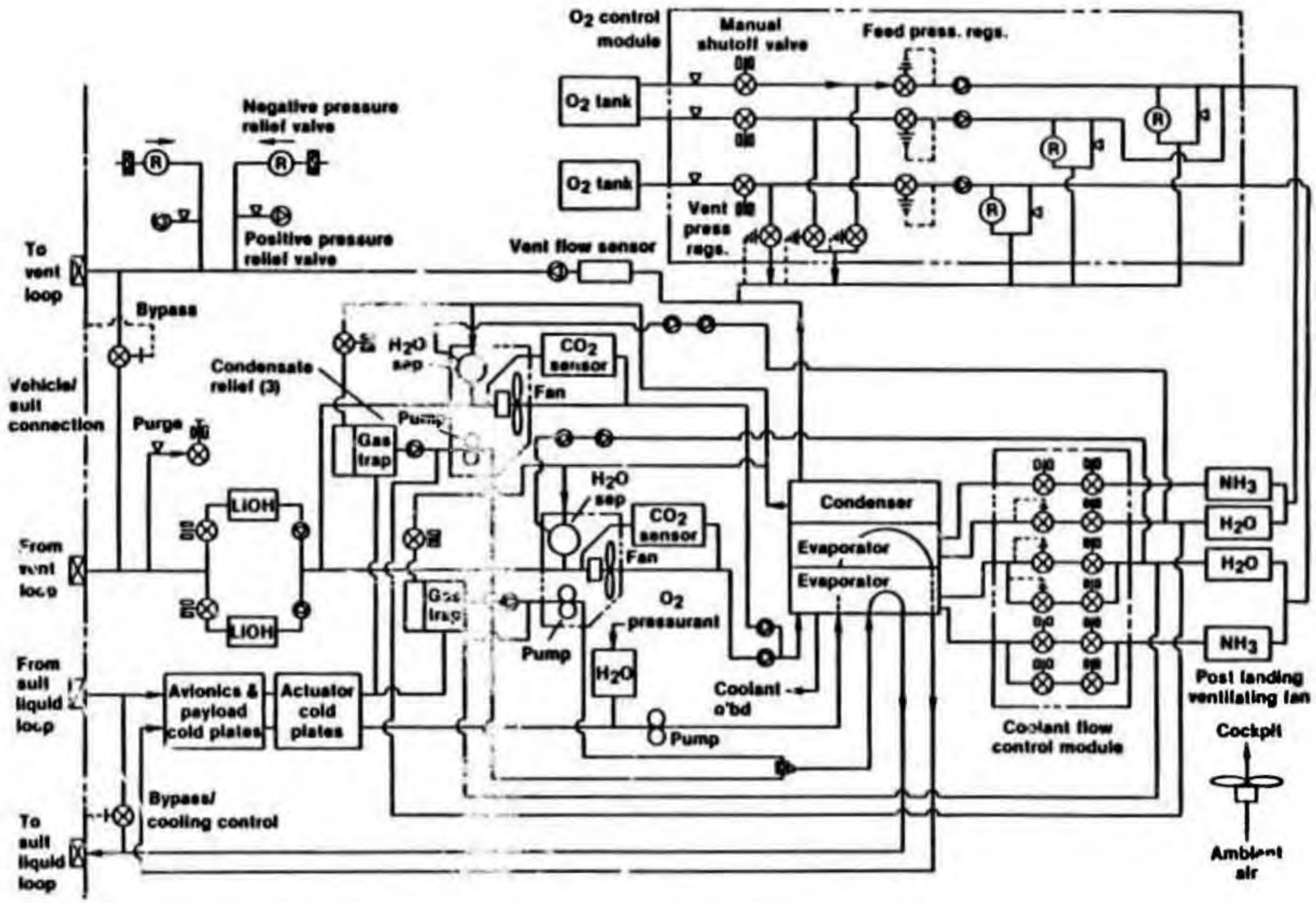


Figure 4-54. SPACEPLANE EC/LSS SUBSYSTEM SCHEMATIC

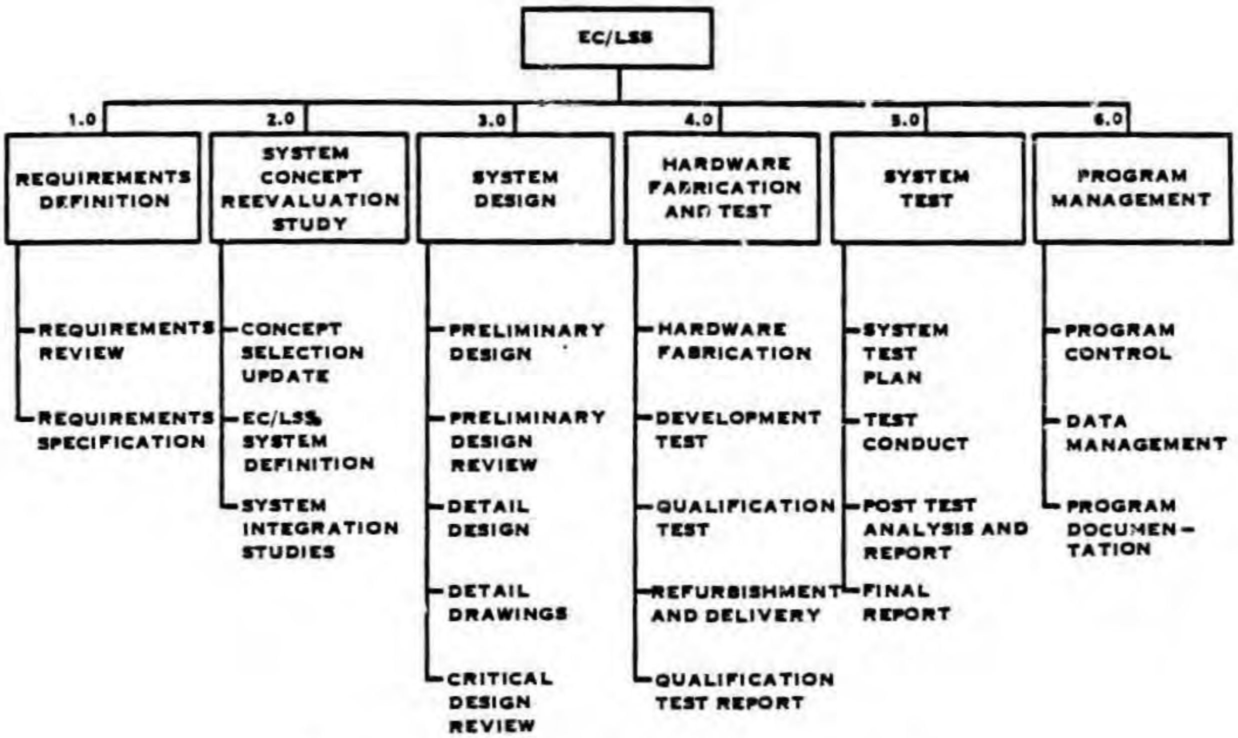


Figure 4-55. WORK BREAKDOWN STRUCTURE

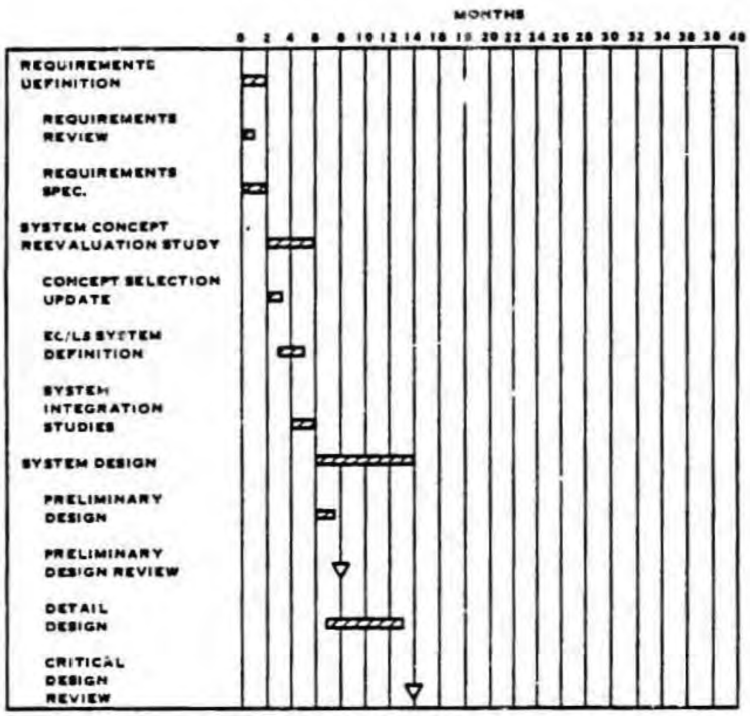
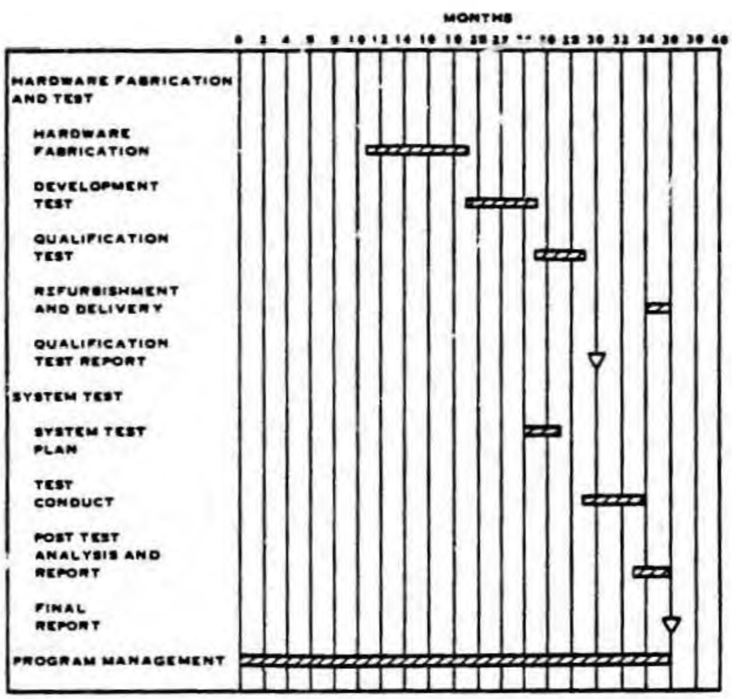


Figure 4-56. **EC/LSS SCHEDULE**



Fgiure 4-56. (CON'T) EC/LSS SCHEDULE

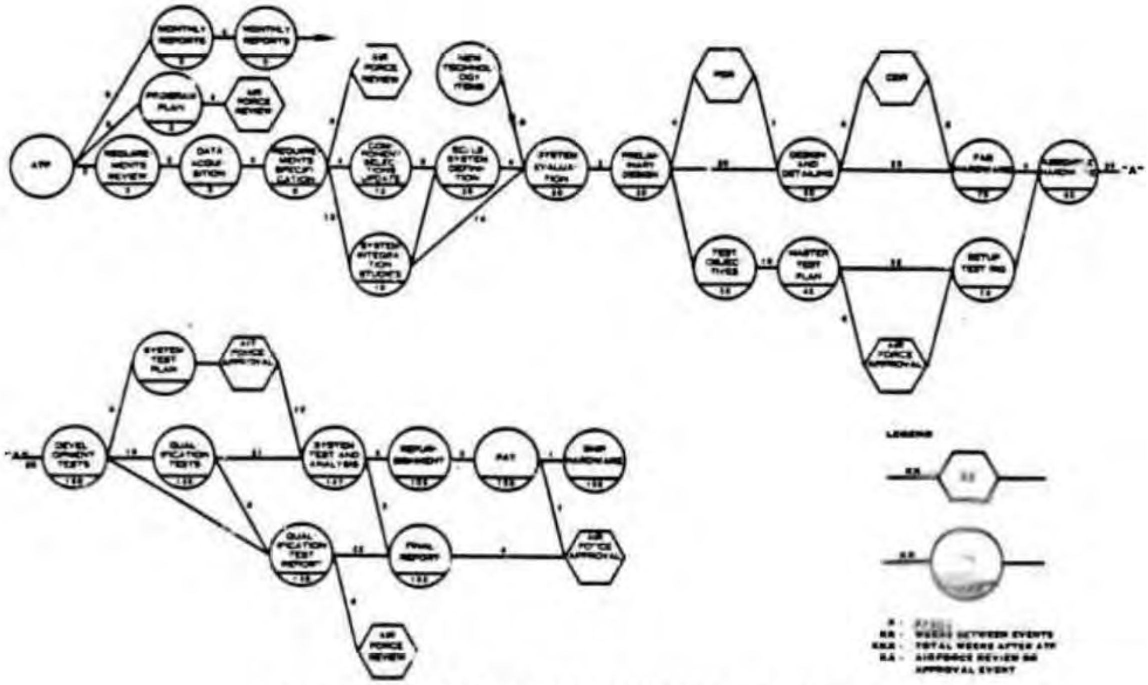


Figure 4-57. **EC/LSS PROGRAM LOGIC FLOW**

- EMU Heads-Up Display
- 8 psi EMU
- EMU Detachable PLSS
- EMU Solid Waste Systems
- Stacked Evaporator

4.7.5 Task Descriptions

As previously stated, Hamilton Standard will conduct the EC/LSS Program in accordance with the WBS defined in Figure 4-55. All in-house financial data collection will be against WBS task element effort. In addition, to assist in the day-to-day administration of the program, each WBS element will have a detailed technical status schedule. The preparation of these schedules is the responsibility of the program project engineer. The schedules will be reviewed on a continuous basis by Program Management. All active technical status schedules will be included in the first monthly progress report and updated on a monthly basis thereafter as part of the monthly report.

The following discussion describes each of the WBS task elements and the program flow, in accordance with the program logic chart (Figure 4-57).

4.7.5.1 Requirements Definition

Under this task, the complete requirements for the EC/LSS will be documented in a design specification.

4.7.5.1.1 Requirements Review

For this task element, Hamilton Standard will use all available information and data, including mission definitions and alternates, vehicle environmental input to subsystem, operational requirements, contingency and docking considerations, EVA requirements, and other demands impacting basic EC/LSS requirements. To obtain this information and to better understand mission requirements and vehicle interfaces, discussions and meetings will be held with appropriate organizations, as required, during the program.

4.7.5.1.2 Requirements Specification

Based on the requirements review, a detailed EC/LSS specification will be generated. Quantitative requirements in the specification will include data from the Preliminary Spaceplane Study, mission definition, vehicle, crew, equipmet, cabin environment, EVA, and other considerations. Qualitative requirements will include reasonable design objectives, based on previous craft and spacesuit experience and Spaceplane Study results.

Upon completion of this program element, the design specification generated will be reviewed with the contract technical monitor and used as a basis for the conceptual design of the Spaceplane EC/LSS.

4.7.5.2 System Concept Reevaluation Study

In this phase, the concept selection will be reevaluated, together with all Spaceplane-related items, against the EC/LSS Requirements Specification, to define a complete EC/LSS.

4.7.5.2.1 Concept Selection Update

The selected EC/LSS concept will be reevaluated against the requirements specification to assure that all mission objectives can be met. Every effort will be made to continue to utilize existing technology and space-qualified hardware in the resulting design to minimize both lead-time and total program costs.

4.7.5.2.2 EC/LSS Definition

Utilizing the concept selected above, a complete EC/LSS will be defined. Included in this definition will be the integration of the new technology items and all ancillary equipment, including EVA and support equipment, to assure all interfaces are compatible within the Spaceplane during all mission phases. Also to be considered is the acceptability to the crew member and such factors as safety, reliability, maintainability, and commonality.

4.7.5.2.3 System Integration Studies

As part of this task element, the following equipment will be defined in detail:

- EVA equipment
- Vehicle EC/LSS control and display requirements
- Flammability control and suppression hardware
- Ground and flight support equipment

The definition will include a list of requirements, a description of the hardware and its availability, and identification of any long-lead-time items.

A study will also be conducted to show that the new technology items defined under the Program Schedule paragraph above will properly integrate with the EC/LSS defined by this program.

4.7.5.3 System Design

Under this task, the EC/LSS and the couch will be designed, detail drawings prepared as required, and a Preliminary Design Review (PDR) and a Critical Design Review (CDR) held.

4.7.5.3.1 Preliminary Design

The design phase effort is initiated in this task element. The objective is selection of an optimum design configuration, including system level performance, safety, weight, volume reliability, maintainability, and consideration of interface and requirements utilizing design layouts, packaging drawings, etc. Recommendations concerning flight controls and displays and ground-test instrumentation will be made. Individual components will be identified and selected for all functions in the EC/LSS. Thermal and performance analyses required to evaluate each components selection will also be made to assure proper system integration at the detail level.

4.7.5.3.2 Preliminary Design Review

A preliminary design review will be held to review all work completed to date and to assure agreement on all elements of the design before it is initiated. Personnel from the Air Force and the contractor are expected to attend this review.

4.7.5.3.3 Detail Design

The design task element covers the generation of detailed technical definition for the system from the configuration concept approved in the design-setting phase. The primary effort is involved in preparation of the detail and assembly drawings for fabrication, procurement, and final assembly of all elements of the subsystem. The effort will also include the additional functional, thermal, and structural analyses of the system to assure compliance with the requirements of the SOW. The task will terminate with conduct of the CDR and incorporation of CDR comments resulting in the approval of the engineering drawings.

Hamilton Standard will conduct a materials control program to identify all nonmetallic materials to ensure compatibility with the environment in the subsystem. This effort will result in the generation of the Nonmetallic Materials List and corrosion-protection summary to be included in the Final Report. Although the effort is scheduled for completion by CDR, on-going overview will be maintained by Materials Engineering throughout the duration of the program.

Intrinsic reliability is a system characteristic that Hamilton Standard recognizes as being of major importance. All design-phase effort will include inputs from our Reliability Engineering Group to assure attainment of the maximum practical reliability. The major effort for this task element will be preparation of the Failure Mode and Effect Analysis. A preliminary analysis will be available at CDR.

4.7.5.3.4 Critical Design Review

Upon completion of the design effort, a Critical Design Review will be held with the representatives from the Air Force and the contractor. At this review, the final design will be reviewed in detail for comments and/or changes. Drawings for selected long-lead items may be released prior to this meeting in order to maintain schedules.

4.7.5.4 Hardware Fabrication and Test

This task includes all effort and material necessary to manufacture, assemble, and test hardware for this program. Also included is the effort to refurbish the hardware prior to delivery to the customer.

4.7.5.4.1 Hardware Fabrication

This fabrication phase includes all effort and materials necessary to manufacture and assemble one complete set of hardware for this program. The phase also includes inspection of raw materials, fabricated hardware, and the complete assemblies. Included is the Manufacturing Engineering and Control effort necessary to prepare all manufacturing documentation and to expedite the processing of the hardware. Because it is planned to deliver this hardware to the customer, all necessary controls and paperwork will be inspected and documented.

4.7.5.4.2 Development Tests

This task element includes the effort to conduct development tests to evaluate the effect of modifications on existing hardware and to establish the performance of all items in the mission operating range. Also included is the effort for the special test equipment to conduct the above tests, as well as any qualification and production acceptance tests.

A master test plan defining the complete test program will be prepared under this task element. It will be submitted to the contractor for comment and approval at least 90 days before testing is to begin.

4.7.5.4.3 Qualification Tests

Within this task element, verification of the qualification test requirements will be accomplished, either by actual testing or by documentation that shows that existing or modified flight hardware is capable of meeting the requirements defined in the master test plan.

4.7.5.4.4 Refurbishment and Delivery

This task element covers the effort necessary to inspect the unit thoroughly after completion of the subsystem testing, to perform any required refurbishment and to prepare the unit for shipment. Because there are no limited-life items in the subsystem, this effort is expected to be minimal. Acceptance tests will be conducted on all items prior to delivery.

4.7.5.4.5 Qualification Test Report

Upon completion of the qualification testing, a qualification test report, fully documenting all test data and conditions, will be prepared. The report will include photographs of the test setups and hardware.

4.7.5.5 System Test

An unmanned system test will be set up and conducted to demonstrate that all items within the EC/LSS integrate and function properly and that the desired system performance is obtained under various performance operating ranges.

4.7.5.5.1 System Test Plan

A test plan will be prepared defining the test setup, operating conditions and parameters, and the required data. This plan will be submitted to the contractor for his approval at least 120 days before testing begins.

APPENDIX A
EC/LSS PERFORMANCE, DESIGN, DEVELOPMENT
AND
VERIFICATION REQUIREMENTS

EC/LSS REQUIREMENTS DEFINITION

SCOPE

This specification establishes a general set of performance, design, development and verification requirements for equipment to be used in the Life Support and Environmental Control Subsystem for Spaceplane.

APPLICABLE DOCUMENTS

The following documents of the exact date of issue shown (current issue of Hamilton Standard documents) form a part of this specification to the extent specified herein. In the event of a conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall take precedence.

Government Documents

Specifications

Federal

QQ-A-367 G (1) 23 May 1968	Aluminum Alloy, Forging
QQ-A-596 26 May 1966	Aluminum Alloy Permanent and Semi-permanent Mold Castings

Military

MIL-T-5021D 1 February 1974	Tests, Aircraft and Missile, Welding Operators Qualifications
MIL-B-5087B (2) 31 August 1970	Bonding, Electrical, and Lighting Protection, for Aerospace Systems
MIL-S-7742	Screw threads, Standard, Optimum Selected Series
MIL-S-8879	Screw Threads, Controlled Radius Root
MIL-N-25027C 22 March 1965	Nut, Self-Locking, 250F, 450F and 800F, 125 KSI FTU, 60 KSI FTU, and 30 KSI FTU.

MIL-C-6021 G 9 September 1967	Casting, Classification and Inspection of
MIL-H-6088 E 9 June 1972	Heat Treatment of Aluminum Alloys
MIL-H-6875 F 17 January 1972	Heat Treatment of Steels (process for Aircraft Practice)
MIL-F-7190 A 18 September 1958	Forgings, Steel for Aircraft and Special Ordinance Aplications
MIL-B-7883 B 20 February 1965	Brazing of Steel, Copper Alloys, Nickel Alloys, Aluminum and Aluminum Alloys
MIL-A-21180 C (1) 26 February 1965	Aluminum Alloy Casting, High Strength
MIL-A-22771 C (1) 25 August 1969	Aluminum Alloy Forgings, Heat Treated
MIL-I-26860 B 17 May 1972	Indicator, Humidity, Plug, Color Change
MIL-H-83282 A 22 February 1974	Hydraulic Fluid, Fire Resistant Synthetic Hydrocarbon Base, Aircraft
MIL-L-23699 B 1 October 1969	Lubricating Oil, Aircraft Turbine Engines, Synthetic Base
MIL-H-81200 A (1) 24 March 1969	Heat Treatment of Titanium and Titanium Alloys
MIL-P-25732B 11 January 1967	Packing, Preformed, Petroleum Hydraulic Fluid Resistant, 275F
MIL-F-83142A 1 December 1969	Forgings, Titanium Alloy, for Aircraft and Aerospace Applications

<u>NASA</u>

SP-R-0022 A 9 September 1974	Vacuum Stability Requirements of Polymeric Materials for Spacecraft Application
MSFC-SPEC-250 (1) February 1964	Protective Finishes for Space Vehicle Structures and Associated Flight Equipment, General Specification for

MSFC Dwg. 10M33107B 8 August 1975	Design Guidelines for Controlling Stress Corrosion Cracking
NHB 8060.1 February 1974	Flammability, Odor, and Offgassing Requirements and Test Procedures for Materials in Environments that Support Combustion
MSFC-SPEC-469 February 1968	Titanium and Titanium Alloy, Heat Treatment of
SC-D-0001	Metal Foil Decals, Manned Spacecraft and Related Flight Crew Equipment, Specification for
SC-M-0003	Markings, Labeling and Color, Manned Spacecraft and Related Flight Crew Equipment, Functional Design Requirements for
SD-W-0020	Water Quality
SE-R-0006A	General Specification for Materials and Processes

Standards

Federal

FED-STD-101B (2) 8 October 1971	Preservation, Packaging, and Packing Materials, Test Procedures
FED-STD-209B 24 April 1973	Clean Room and Work Station Requirements Controlled Environment

Military

MIL-STD-129F 30 March 1973	Marking for Shipment and Storage
MIL-STD-143 B 12 November 1960	Standards and Specification, Order of Precedence for the Selection of
MIL-STD-454C 1 December 1971	Standards, General Requirements for Electronic Equipment (Requirement Numbers 4 and 11 only)

MIL-STD-130	Identification and Marking of U.S. Military Property
MIL-STD-280 A 7 July 1969	Definitions of Item Levels, Item Interchangeability, Models and Related Terms
MIL-STD-794 D (1) 25 May 1973	Parts and Equipments, Procurement for Packaging and Packing for
MIL-STD-810 B (4) 21 September 1970	Environment Test Methods
MIL-STD-1246 A	Product Cleanliness Levels and Contamination Control Program
MIL-STD-1247 B 20 December 1968	Marketing and Functions and Hazard Designations for Hose, Pipe and Tube Lines for Aircraft, Missile, and Space Systems
MIL-STD-1472 A 15 May 1970	Human Engineering Design Criteria Military Systems, Equipment, and Facilities

Handbooks

Military

MIL-HDBD-5B (4) 1 September 1971	Strength of Metal Aircraft Elements

NASA

NHB 6000.1 (1A) December, 1969	Requirements for Packaging, Handling and Transportation for Aeronautical and Space Systems Equipment and Associated Components
TMX-64589 10 May 1971	Terrestial Environment (Climatic) Criteria Guidelines for use in Space Vehicle Development, 1971 Revision

CHARACTERISTICS

Performance

Spaceplane EC/LSS equipment will provide selected crew functions in life support. Operation will include service during all orbital mission phases. Nominally, this shall include support of a crew of one for a maximum duration of 24 hours.

Oxygen Partial Pressure

The oxygen partial pressure shall be maintained at 8 $\pm$ 0.25 psia.

Carbon Dioxide Partial Pressure

Carbon dioxide partial pressure will be maintained at or below 3.0 mm Hg during nominal operations.

Trace Contaminants

a. Normal atmosphere leakage shall not be considered as a method of controlling the level of airborne trace contaminants.

b. Expected airborne trace contaminants, initial and nominal production rates and their maximum allowable concentrations during nominal operational levels are (TBD).

c. The MAC (maximum allowable concentration) of total organics, exclusive of fluorocarbons, is 100 ppm n-pentane equivalents.

d. The MAC of total fluorocarbons is 100 ppm.

Airborne Particles Contaminant Control

The EC/LSS shall adhere to airborne particle control requirements as follows:

Airborne particle filtering	(TBD) micron nominal

Drinking Water

Drinking water shall meet the requirements of the water quality Specification: NAS-MSCFC specification SD-W-0020.

Life Requirements

The equipment shall be designed to provide for (TBD) years the most cost effective life capability, considering minimum maintenance and/or refurbishment, as well as state-of-the-art hardware design.

Operating LIfe

The equipment shall be capable of performing all operations specified herein for a minimum of 30 days under mission or standby conditions.

Useful Life/Shelf LIfe

As a design objective, the equipment shall have a useful life of (TBD) years when exposed to space environments with a minimum of preventive maintenance, servicing, repair, and parts replacement. In those cases where age-sensitive materials cannot be avoided, scheduled maintenance shall be permitted.

Limited Life Items

Limited life items (i.e., items whose expected life is less than the specified minimum operating life of the equipment of which the part is a member) shall not be incorporated in the design unless justified to and approved by the prime contractor. When used, limited life items shall be controlled from date of manufacture through delivery to the prime contractor. A list of limited life items used in the design shall be established. The list shall include:

a. Item name
b. Item number
c. Life limits
d. Operating parameter limit
e. Refurbishment limit
f. Handling and storage requirements
g. Special requirements not covered above

A record of accrued calendar time and operating time on limited life items by part name, part number, and serial number shall accompany the delivery of the hardware in which the limited life items are included.

Reliability

The equipment shall be fail-operational/fail-safe as a minimum. Fail-operational/fail-safe is the ability of each subsystem to sustain a double known failure and not induce a failure in another subsystem, or compromise crew safety. Additional tolerance to failures utilizing further redundancy shall be considered when sufficient rationale warrants.

Fail-Safe Provisions

The following provisions shall be incorporated to prevent failure propagation of those failure modes identified as critical in the failure modes and effects analysis.

a. Failure modes which could induce a failure in another subsystem shall have single backup provisions.

b. Failure modes which could compromise crew safety shall have dual backup provisions.

c. Backup provisions shall include redundancy and/or automatic sensing and shutdown equipment. No crew action shall be required to place the subsystem in a safe configuration subsequent to a failure.

d. Should 'on-line' redundancy be employed, means of identifying a failure in the primary equipment shall be incorporated.

e. Instrumentation and components used for subsystem operational control shall not be used for failure detection and shutdown. The shutdown equipment shall be independent of the subsystem control equipment and powered by a separate emergency power source.

f. It shall be the responsibility of each subsystem to monitor its critical interfaces and to initiate shutdown should a loss of that interface result in a failure propagation.

g. Safety critical shutdowns shall have manual override provisions.

h. Each redundant or shutdown circuit shall be capable of verification as part of the ground checkout procedures.

i. Valve position indicators (visual or electrical) should be considered to verify operation during checkout sequences.

j. A warning signal will be provided during periods when shutdown features are inhibited as part of the startup sequence.

k. Manual startup and shutdown sequences shall be fail safe. At least two procedural operations shall be required for initiation of safety-critical functions. A hazard shall not result from any single procedural error.

Instrumentation

The instrumentation requirements are delineated below:

a. Instrumentation shall be limited to that necessary for normal operation, system level performance test monitoring, and subsystem level fault detection.

b. Process control and emergency shutdown shall be performed at the subsystem level by the central computer.

c. Local limit checking shall be performed on safety critical instrumentation and warning signals shall be issued when emergency life support parameters are exceeded.

d. When the time from failure to critical effect is sufficient to permit crew action, lower anticipatory limits shall be established and early warning signals shall be provided at pre-selected intervals prior to automatic shutdown.

Maintainability

The crewman will perform minimal inflight maintenance and repair to sustain the integrated Environmental Control and Life Support Subsystem operation throughout the mission.

Scheduled Maintenance

Scheduled maintenance shall be limited to disposition of disposables, replenishment of consumables, cockpit cleaning, coolant loop and electronics checkout, and oxygen, water, ammonia, LiOH canisters and food and drinking water and coolant loop recharging.

In the event equipment design cannot avoid preventive maintenance requirements, all preventive maintenance tasks shall be identified (i.e., removal and replacement of filters, removal and replacement of rotating components).

Unscheduled Maintenance

Installation of subsystems equipment shall be designed so that corrective maintenance can be accomplished during vehicle turnaround by replacement of flight replaceable units (FRU). A flight replaceable unit is a combination of components, units, parts, assemblies, subassemblies, etc., that are contained in one package or are so arranged that together the combination is common to one mounting, and provides a complete function to the larger entity within which it operates.

Special Tools

The equipment shall be designed to preclude, to the maximum extent possible the use of special tools and equipment for maintenance and repairs.

Special tools, if required, shall be provisioned as GSE or FSE and shall be designed to withstand the intended use throughout the life of the equipment.

Temperature Sensors

Temperature sensor installation in liquid line ports shall be designed in a manner to permit removal and replacement with no fluid leakage.

Servicing

Servicing and sampling provisions for fluid systems shall be provided and shall be accessible without requiring removal of access plates or covers except service caps.

The equipment shall not require constant/periodic circulating during powered down situations, and shall be designed to permit flushing and drying to accommodate cleaning of the vehicle.

The equipment shall be designed so as to minimize the need for on-line support equipment.

No on-vehicle adjustments or calibration shall be permitted except as identified in detail specifications. Calibration controls, when required, shall be accessible and clearly marked for major functions.

Servicing and test points shall be accessible and clearly marked.

Attachment fittings for components routinely removed shall be operable without hand tools.

QUALITY ASSURANCE PROVISIONS

Subcontractor's Quality Program

Equipment furnished in compliance with this specification shall be produced under the controls established in the contractor's Quality Program Plan. This plan shall be prepared and submitted for approval in accordance with the quality requirements specified in the Contract.

Verification

All hardware furnished in accordance with this specification shall be verified that each specific design requirement delineated in Sections 3 and 5 of this document has been satisfied.

Verifiction may be accomplished by means of test, analysis, or assessment techniques or combinations of all three. Selection of the method of verification shall be based on hardware class (i.e., new existing as-is or existing modified) as well as the results of an in-depth hardware evaluation. This evaluation shall investigate all available material related to the testing of usage of the equipment (i.e., history, prior use, previous test results, etc.) and compare it to specific equipment requirements.

A. Verification By Analysis

This method shall use existing analysis or may include additional analysis and design techniques to support the verification. When this method is selected, it shall be incumbent upon the contractor to provide substantiation of the method, including documented evidence. Analysis methods include:

1. Similarity

Comparison of configuration, design and environmental capability based on prior test results, with similar requirements.

2. Design Correlation

Engineering analyses, data extrapolation or computer modeling methods, to show that the design is compatible with a nonsimulatable flight condition.

3. Data Extrapolation

The use of analysis to extrapolate test data to the on time spectrum of flight conditions, where it is not economically feasible to test.

B. Verification By Assessment

This method applies to hardware acceptance and checkout. Assessment methods include:

1. Inspection

   Verification of construction features, drawing compliance, and physical condition of the end item.

2. Design Review

   Verification of hardware requirements during formal review.

C. Verification By Test

1. If certification by test is chosen, it shall include tests as dictated by the hardware class, suitability evaluation and Spaceplane requirements, and shall comply with the applicable paragraph of this specification.

2. As a general guideline off-limit testing will not be conducted. However, off-limit testing will be considered:

   a. When design margins are relatively small with respect to off-nominal abort conditions.

   b. When uncertainty exists in the definition of the design criteria.

   c. When single point failure modes exist.

   d. When failure mode analysis indicated that a credible probability of associated hardware failures will create an off-limit condition.

   Testing of this nature must have considered preservation of at least one specimen of certification hardware for later testing.

3. Acceptance tests, procedures, equipment and test levels shall be proven and verified during development testing.

4. Where new materials or existing materials under new conditions are to be used, adequate testing shall be performed to statistically identify material property values.

5. Application of non-destructive evaluation techniques shall be proven and verified during the development test program.

6. Certification shall be structured to verify the full range of the design requirements under mission environments.

7. All test specimens shall be processed through specified acceptance testing prior to subsequent test.

8. Where redundancy in design exists, each redundancy shall be verified through normal output sources designed for that purpose.

9. Wherever practical and technically sound, accelerated life test techniques shall be utilized.

10. Testing shall be conducted at the most effective level of assembly.

Test Responsibility and Location

The EC/LSS subcontractor shall be responsible for implementing the quality assurance requirements specified herein at the CEI level and higher. For lower levels of component identity, the EC/LSS subcontractor may implement the process or assign share elements with lower level suppliers. Each component specification shall delineate the assignment of responsibilities. Except as otherwise noted, EC/LSS subcontractor facilities or any commercial laboratory approved by the subcontractor may be used for the conduct of tests.

Classification of Tests

Items covered by this specifiction shall be subjected, as required, to the following tests as defined by detail specifications to determine compliance with all specified requirements:.

a. Development Tests

1. Design feasibility (Breadboard)
2. Design evaluation
3. Individual Requirements
4. Structural
5. Electromagnetic compatibility
6. Maintainability demonstration

b. Acceptance

1. Individual tests

   a. Examination of product
   b. Base point tests
   c. Operational
   d. Fixed duration burn-in/run-in

Development Tests

Development encompasses engineering evaluations of manufacturing processes and techniques for the proposed hardware, software, manufacturing processes, and techniques for the purpose of acquiring engineering data; identifying sensitive parameters; evaluating the development configuration; providing the necessary confidence that the hardware will meet the specificatiion requirements; and assurance that the manufacturing process will produce an acceptable product. Development objectives shall encompass the following as a minimum.

a. Design and performance capability, including redundancy.

b. Ability to meet mission requirements with adequate design margin.

c. Integration of each component and subsystem with other components, subsystems, facilities, support equipment, etc.

d. Establishment of processes, procedures, equipment and test levels for manufacturing, acceptance testing, maintenance, checkout, and operational phases of the program.

e. Identification of significant failure mode and effects.

f. Determination of the effect of various combinations of tolerance and drift design parameters.

g. Determination of the effect of combinations and sequences of environments and varying stress levels.

h. Identification of safety hazards, parameters, requirements, and procedures.

Design Feasibility

Design Feasibility tests shall be conducted as required on selected material, parts and circuits both on the bench and under simulated critical environments to verify feasibility of design and aid in the selection of material and parts. This shall include fluid compatibility tests where applicable. A summary outline of testing to be accomplished shall be prepared. Tests reports shall be developed which summarize test results.

Design Evaluation

Design evaluation tests shall be performed to demonstrate attainment of functional and performance design requirements. All FRU's embodying new components or concepts shall be subjected to this test. The test item shall be a full-up prototype of the end item fabricated to approved production type drawings.

Test Requirements

In general, tests and sequences to be conducted shall conform to the requirements of the detail specifications. Tests shall include the following.

Vibration

The vibration tests shall be performed to determine if the equipment is constructed to withstand expected dynamic vibration stresses and to insure that performance degradations or malfunctions will not be produced by the dynamic environment during operation.

The test item shall be in flight and reentry modes except for critical electronic components which should be operated to allow continuity checks, even if they are turned off during portions of the normal flight envelope. Prior to and after the vibration test, the equipment shall be visually examined and functionally tested to check performance.

The equipment shall be subjected to the vibration environment to be encountered during mission phases.

Electromagnetic Compatibility

Qualification tests for electromagnetic compatibility, where applicable shall be conducted in accordance with SL-E-0001 and SL-E-0002.

Maintainability Demonstration

Hamilton Standard will specify a test to demonstrate the following, as applicable:

a. Fluid and gas replenishment.

b. Access capability to all Line Replaceable Unit (LRU) attaching points, fluid line connections and interconnects, and handling equipment attachment points and fittings.

c. All difficult, hazardous and repetitive tasks on installed section.

d. Trouble shooting procedures.

Acceptance Tests

Acceptance tests shall be performed to verify that all equipment supplied meets the performance requirements of Section 3 in all respects; including construction, workmanship, and quality. Acceptance or approval of material during the course of manufacture shall not be construed as a guarantee of its acceptance in the finished product. Evidence of non-compliance with the above include the following elements, as applicable, for each Contract End Item (CEI). The CEI specification will delineate the specific requirements which are applicable in each case. Abridged component acceptance tests, based on the same element definition, shall be considered and applied as required to preclude assembly of a complete package CEI with defective or marginal components.

General Requirements

Test Duration

The total time for acceptance testing shall be recorded.

Inspection After Test

Upon completion of the acceptance test, the unit shall be subjected to a visual inspection. If any part is found to be defective, an approved part shall be supplied to replace it and a suitable penalty test may be conducted.

Description of Acceptance Tests

The acceptance tests described below shall be conducted on the equipment as specified in the detail specifications.

Examination of Product

All equipment shall be examined carefully during appropriate stages of manufacture and after final assembly to determine that the material, workmanship and mass properties requirements have been met and that all internal components and subassemblies are properly coded, located and affixed and that all external devices such as flanges, mounting provisions and connector locations are correct as specified herein and in the applicable drawing. Appropriate records shall be reviewed after assembly to assure completeness of product examination.

Weight

The dry test article shall be weighed and its weight shall not exceed that specified in the applicable detail specification.

Burn-in/Run-in Tests

Each specified subsystem shall be subject to a burn-in/run-in test to reduce latent defects. The optimum time/cycle limit to be imposed on each type of hardware to effectively expose workmanship and component defects or to provide the necessary wear-in period to assure proper seating or conditioning is TBD.

Base Point Tests

Base point tests shall be used as baseline test to establish functional parameters which can be used for comparison purposes to detect degradation in equipment operation or equipment leakage as a result of Burn-in/Run-in Tests. Base point tests shall consist of the following, as appropriate:

a. Performance
b. Leakage
c. Calibration
d. Handle torque
e. Other non-destructive tests as described in the detail specifications.

Performance Tests

Performance tests shall demonstrate that the unit under test is capable of operating within specified design performance requirements of the applicable detail specification. The performance test shall ensure that the unit under test is functionally monitored full time for possible discontinuity and intermittent conditions. Performance tests shall be conducted before, during and after each test or sequence of tests as appropriate.

The test shall demonstrate that the unit is operating within specified tolerances throughout its operational range. The test shall consist of, as a minimu, an abridged mission profile in which all components are exercised including redundant mode operation.

Leakage Tests

Leakage tests shall be made on assemblies and applicable components of those assemblies to the requirements specified in detail specifictions. Leakage tests shall be performed utilizing helium or nitrogen unless otherwise specified in the detail specifications.

Calibration Tests

The equipment shall be operated to establish full compliance with the detailed performance requirements of Section 3. Calibration, if required, shall be conducted and set points recorded and become part of the data package for each unit.

Valve Torques

Valve operational torque tests shall be conducted on all manual valves. Loads shall be applied in all operating directions and shall not exceed 10 in-lb torque.

Random Vibration

The objective of the acceptance vibration testing is to locate latent material and workmanship defects in equipment of proven design before they are integrated into the Spaceplane.

Prior to and after the vibration test, the equipment shall be visually examined and functionally tested to verify performance integrity.

The test item shall be subjected to the random vibration level (TBD) along each of three mutually perpendicular axes.

The vibration test duration shall not be less than 30 seconds or greater than 5 minutes per axis. Should reruns be required in any axis, the total accumulative vibration test time in that axis shall not exceed five minutes. The package shall be tested on the same fixture used for the vibration development tests. Functional tests shall be conducted on all items before and after acceptance environmental tests. Functional verification during the vibration tests shall be made for those outputs significant to the mode of flight being simulated. All electrical circuits shall be monitored for continuity during the vibration test to check for intermittents and opens.

Proof Pressure Test

Each specified CEI shall be subject to a proof pressure test. Testing shall not accumulate in excess of a maximum number of proof pressure cycles, as noted in the detail specification. pressure and tank temperature vs. time shall be recorded (from strip chart of equivalent) at all pressure levels above 15 psig. Evidence of permanent set or deformation, excessive leakage, or degradation of performance shall be cause for rejection.

Dielectric Strength

Dielectric strength tests shall be conducted in accordance with SL-E-0002 when required by the detail specification.

Insulation Resistance

Insulation resistance shall be conducted in accordance with SL-E-0002 when required by the detail specifications.

Special Tests

Special tests shall be conducted, as required, for the purpose of checking the effect of any design or material change on the performance of the equipment and to assure adequate quality control. The items selected for special tests as part of development and/or acceptance tests are identified in the applicable detail specifications.

Special Test Descriptions

Performance Maps

During development, performance testing shall be conducted to generate performance maps for both normal and selected off-design conditions. The tests, in general, shall consist of the following:

a. Fans

Determine variation in fan performance with flow inlet temperature, input voltages, and system pressure drop. Determine resulting variations in flow, fan speed and input current. Transients for nominal start up conditions shall be determined.

b. Cold Plates and Evaporator

Determine variations in performance with change in flow, change in fluid operating temperatures and, where applicable, change in operating pressures.

Stability/Response

Development tests shall demonstrate that check valves, relief valves, regulators and systems do not exhibit signs of instability at design range flow rates. Tests shall also be conducted to demonstrate that pressure regulators shall control pressure within specified limits while reference pressure varies at expected rates.

Noise Tests

Noise generating components shall be subjected to noise level tests as installed in their final FRU level of assembly and interfacing with actual or duplicated inlet and outlet plumbing or ducting, as applicable. Sound test set-up shall include controllable system resistance and frequency range measurement from 40 to 10,000 Hz. Where feasible, Spaceplane equipment orientation and equipment installation shall be simulated. Noise level of equipment shall not exceed that specified in Section 3 of the applicable Detail Specification.

Negative Proof Pressure

Negative proof pressure tests shall be conducted on all items which are subjected to external fluid or gas pressure during ground servicing and/or flight operation. Negative proof pressure shall be imposed on the item and held for at least 2 minutes. Evidence of permanent set or deformation, excessive leakage or degradation of performance shall be cause for rejection.

Drainage

With the test item in the normal launch and landing attitudes the fluid systems shall be completely filled with the working fluid then drained to the maximum extent possible. The fluids remaining shall be determined.

Fluid Compatibility

Tests of the effects on the test unit of the chemical actions of fluids specified to be used shall be determined and proposed for the following:

a. Aging with fluids
b. Drying in air
c. Contact with vapors and most detrimental combinations thereof

Test conditions shall simulate those encountered in actual applications. Qualifying criteria shall be specified in the detail specification.

Overspeed Tests

The test item shall be operated at a sustained overspeed condition to determine if performance degradation will occur. Test to be conducted with nominal design inlet temperatures and nominal system delta pressure (pre-set prior to initiation of overspeed condition). The overspeed condition will be initiated by adjusting input voltage from nominal condition to nominal condition plus 20%.

General Conditions For Tests

These rules shall be in accordance with MIL-STD-810 and the following general test conditions. In the event of a conflict, this specification shall have precedence. These conditions shall apply to qualification and acceptance tests.

Standard Conditions and Tolerances

Standard Conditions for Test Area

Standard ambient conditions for the test area shall be as indicated below, unless otherwise specified.

Temperature:	71.5 ± 9F
Relative Humidity:	70% or less
Barometric Pressure:	13.4 ± 1.6 psi

Cleanliness Conditions:

The test area shall be visibly clean, i.e. visibly free of physical contaminants such as corrosion products, metal or non-metal chip and shavings, oil or grease etc. The test item shall be cleaned before it reenters a class 100,000 cleanroom, FED-STD-209B.

Measurements and Tolerances

All instrumentation used in these tests shall be in current calibration and shall bear appropriate documentation to this effect from an approved calibration laboratory.

The maximum allowable tolerances for test conditions and physical properties shall be as follows, unless otherwise specified by the applicable test section in the particular test specifications. The values are exclusive of instrument accuracy.

a.	Mass	±0.3%
b.	Center of Gravity	±1% on assemblies ±5% on components
c.	Temperature	-58 to 212 (± 3.6F) 212 to 698 (± 7F) -58 to -328 (± 7F)
d.	Acoustic noise OAL 1/3 Octave Bands	 ± 1 dB ± 3 dB
e.	Humidity	0 - 5% R.H.
f.	Sinusoidal Vibration Amplitude Sweep Speed	 ±10% ±5%
g.	Random Vibration (overall RMS level)	±10%
h.	Random Vibration (PSD)	±3 dB
i.	Random Vibration Test Time	0 to +10%
j.	Displacement	±10%
k.	Vibration Frequency	

Frequency Range	Maximum Effective Bandwidth
10 to 100 Hz	6 Hz
100 to 500 Hz	12 Hz
500 to 2,000 Hz	24 Hz

l. Acceleration $\pm 5\%$

m. Pressure $\pm 10\%$

n. Force + TBD

o. Power $\pm 5\%$

p. Flow rates $\pm 5\%$

NOTE:

OAL	Overall Level
RMS	Root mean Square
dB	decibel
PSD	Power Spectral Density

Accuracy of Test Apparatus

The accuracy of instruments and test equipment used to control or measurethe test parameters shall be less than one-third the tolerance for the variable to be measured that is specified in this specification or equivalent to the "state-of-the-art" for measurements that cannot be made within one-third of the tolerance band.

Vibration Testing

The vibration control accelerometer signal and any response accelerometer signals shall be recorded (and identified by voice annotation) on magnetic tape for all acceptance and qualification vibration tests. The tape recorder shall record these signals whenever power is applied to the shaker system. System calibration information (g's/volt, etc.) sufficient to allow analysis of the vibration signals subsequent to the test, shall also be recorded on the magnetic tape and any other applicable documentation. The magnetic tapes shall be maintained as part of the permanent vibration test records. The vibration control accelerometer(s) shall be located to ensure that the specified vibration is being applied to the test specimen.

For random vibration testing, a dynamically similar dummy shall be used in place of the test specimen, when possible, for pre-test equalization(s). The final equalization prior to the test shall be accomplished using the test specimen and shall be conducted at the full specified random vibration level. The time expended during the final equalization shall be counted as part of the required test time for the random vibration test. The final equalization shall be verified by a narrow band analysis prior to initiation of the test, using effective bandwidths not exceeding those specified in paragraph 4.7.1.2. A narrow band spectral analysis shall be performed on the input control accelerometer signal once per hour during the test (analysis may be actually performed subsequent to test run) to demonstrate that the test specimen has been subjected to the specified random spectrum. All random spectral analyses shall be performed as X-Y log-log plots of acceleration spectral analyses density (g^2/Hz) versus frequency (Hz). The major components shall be exposed to the requirements of the detail specification in each of three orthorgonal axes for time periods specified in the detail specification.

PREPARATION FOR DELIVERY

General Requirements

Preservation, packaging, and packing of all equipment and materials shall be in accordance with the requirements of MIL-STD-1246A as required by the detail drawing/specification. Packing/packaging shall be such as to protect against corrosion, deterioration in transit and/or immediate storage. For vendor items delivered to Hamilton Standard, any special requirements of the purchase order shall also apply.

Contract End Item

The following detailed requirements shall govern the preparation for shipment and the transport of all Spaceplane End Items to government facilities and test sites. The methods of preservation, packaging and packing utilized shall adequately protect the equipment against the transportation, handling, and storage environments.

Packaging, handling and transportation shall be in accordance with Level B of MIL-STD-794.

Packaging of Precision Clean Items

Items cleaned to the level specified in Section 3 shall be protected in accordance with MIL-STD-1246A to assure maintenance of the prescribed cleanliness level.

Temporary Closure Identification

All temporary closure devices shall be of a high visibility color (e.g., cerise) and require streamers, if necessary, to ensure that installed plugs and covers are easily identified under casual observation.

Packaging Data

Drawings, Specifications, plans, or other data defining methods of packaging, handling, or transportation shall be supplied only as required by Data Submittal requirements of the Contract/Purchase Order.

Marking for Shipment

Interior and exterior containers for CEI's shall be marked and labeled in accordance with MIL-STD-129 including precautionary marking necessary to ensure safety of personnel and facilities and to ensure safe handling, transport, and storage. For hazardous materials, markings shall also comply with applicable requirements governing packaging and labeling of hazardous materials. Packaging with reuse capability shall be identified with the words "REUSABLE CONTAINER - DO NOT DESTROY - RETAIN FOR REUSE". Identification information on interior and exterior containers shall be in the following format and shall include:

BUYER PART NUMBER ______________________

ITEM NAME ______________________

MANUFACTURER'S TYPE OR PART NUMBER ______________________

TRACEABILITY IDENTIFICATION ______________________

AGE CONTROL MARKING ______________________

CLEANING MARKING THIS PART HAS BEEN CLEANED TO LEVEL ____________

____

SERIAL NUMBER ______________________

MANUFACTURER ______________________

BUYER PURCHASE ORDER NUMBER ______________________

DATE OF PACKAGING ______________________

LEVELS OF PACKAGING AND PACKING ______________________

MANUFACTURER'S PACKAGE PART NUMBER
(NOT REQUIRED FOR OFF-THE-SHELF CONTAINERS) ______________________

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APPENDIX B

EC/LSS COMPUTER MODEL

EC/LSS COMPUTER MODEL

The following charts summarize the thermal EC/LSS computer model constructed for Spaceplane design studies. The charts indicate computer model logic, uses, required input and output.

PERFORM TRADE-OFFS OF SYSTEM CONFIGURATIONS:

- SELECTION OF OPTIMUM EQUIPMENT
- STEADY-STATE SYSTEM PERFORMANCE
- WEIGHT, VOLUME PREDICTIONS

COMPUTER MODEL

USED TO COMPARE VARIATIONS IN

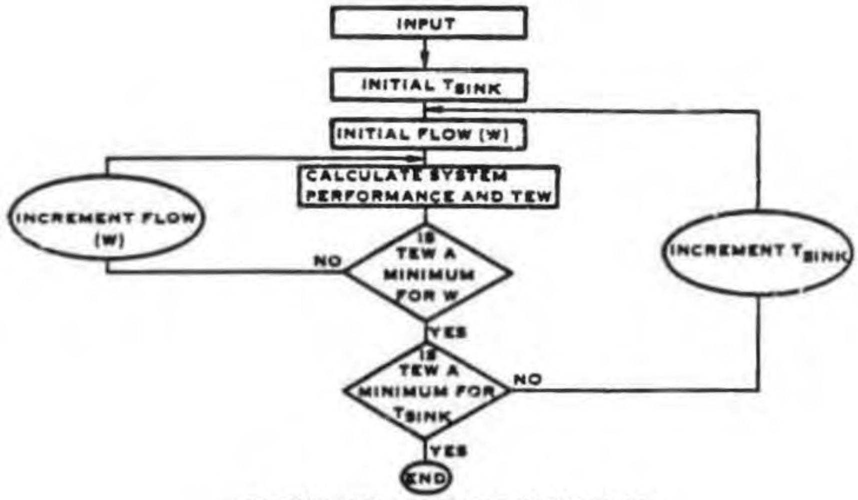
- COMPONENT SELECTION AND ARRANGEMENT
- HEAT LOADS
- TEMPERATURE LIMITS
- COOLANT FLUIDS
- SUIT OR COCKPIT PRESSURE
- SUIT OR COCKPIT ATMOSPHERE COMPOSITION
- POWER PENALTY
- MISSION LENGTH

OPTIMIZATION

INPUT

- BASELINE SYSTEM SCHEMATIC
- DESIGN POINT HEAT LOADS
- SELECTED TEMPERATURES AND PRESSURES
- MINIMUM AND MAXIMUM TEMPERATURES AND HUMIDITIES
- MINIMUM ALLOWABLE FLOW RATES
- POWER PENALTY
- MISSION LENGTH

OPTIMIZATION



OPTIMIZATION LOGIC SEQUENCE

OUTPUT

- COMPONENT WEIGHT, POWER, VOLUME
- COMPONENT DESIGN POINT PERFORMANCE PARAMETERS (E.G., UA)
- SYSTEM VOLUME AND EQUIVALENT WEIGHT
- SYSTEM OPTIMIZED FLOW RATES
- SYSTEM PRESSURE DROPS
- HEAT SINK OUTLET TEMPERATURE

OPTIMIZATION

INPUT

- BASELINE SYSTEM SCHEMATIC AND FLUIDS
- COOLANT FLOW RATE
- HEAT LOADS
- DESIRED CONTROL TEMPERATURE
- HEAT EXCHANGER SIZES
- FAN AND PUMP POWERS
- MISSION LENGTH

LOGIC

- USES MAIN LINE PROGRAM AND COMPONENT SUBROUTINES SIMILAR TO OPTIMIZATION PROGRAM

OUTPUT

- SYSTEM TEMPERATURES AND HUMIDITY
- EXPENDABLE CONSUMPTION

STEADY-STATE SYSTEM PERFORMANCE

TEMPERATURES-DEG F		HEAT LOADS-BTUH		FLOW RATES	
HEAT SINK OUTLET TEMP	34.00	PILOT METABOLIC LOAD	1800.00	LCG COOLANT FLOW-MIN (LB/HR)	20.00
HEAT SINK TEMP	32.00	HYDRAULIC HX LOAD	100.00	MAIN LOOP MIN FLOW (LB/HR)	50.00
MAX HEAT SINK OUTLET TEMP	45.00	LIOH SENSIBLE LOAD	89.40	COOLANT FLOW INCREMENT (LB/HR)	0.10
SUBLIMATOR FEEDWATER TEMP	60.00	LATENT LOAD	43.50	VENT LOOP FLOW (CFM)	6.00
OUTLET TEMP INCREMENT	0.13	WATER SEPARATOR HEAT LOAD	10.00	HYDRAULIC HX FLOW (LB/HR)	100.00
HYD FLUID HOT RETURN TEMP	100.00				

COMPONENT CHARACTERISTICS AND PERFORMANCE		COOLANT PROPERTIES		COMPONENT OPERATING PRESSURES AND PRESSURE DROPS	
		SPECIFIC HEAT BTU/LB/DEG F	1.00		
PIPING INSIDE DIAMETER-INCHES	0.13	DENSITY-LBS/FT**3	62.40	AIR LOOP PRESSURE-PSIA	14.70
TOTAL PIPING LENGTH-FEET	1.00	VISCOSITY-LB/HR-FT	0.02	DESIGN HEAT SINK PRESSURE DROP-PSI	1.00
COLD PLATE WET FACTOR LBS/FT**2	0.50				
CONDUCTANCE-BTU/HR/FT**2/DEG F	40.00			AIR LOOP DUCT PRESSURE DROP-INCHES H2O	1.00
PIPE WEIGHT-LBS/FT OF LENGTH	1.00				
AIR LOOP FAN POWER-WATTS	10.00	MISSION DEFINATION		LIOH PRESSURE DROP-INCHES H2O	1.00
LIOH PACKAGE FIXED WGT-LBS	1.00				
AIR LOOP FAN FIXED WGT-LBS	1.00	MISSION LENGTH-HOURS	1.00	OTHER	
H2O SEPARATOR FIXED WGT-LBS	1.00	POWER PENALTY-LBS/WATT	1.00		

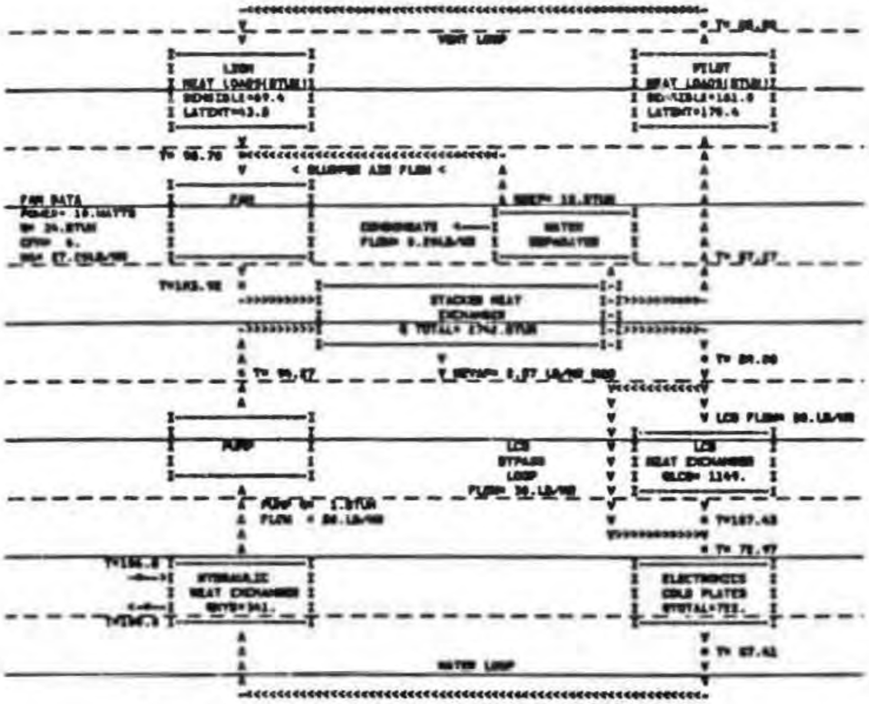
ELECTRONICS COLD PLATE INPUT

BOX #	Q (WATTS)	BASE AREA (FT**2)	BASE TEMP LIMIT (DEG F)
1	[illegible]	0.29	120.00
2	15.00	0.50	120.00
3	45.00	0.44	120.00
4	40.00	0.34	120.00
5	[illegible]	0.39	120.00
6	25.00	0.47	120.00
7	5.00	0.25	120.00
8	25.00	0.79	120.00
9	[illegible]	0.49	120.00
10	5.00	0.63	120.00
11	5.00	0.17	120.00
12	0.00	0.17	120.00
13	[illegible]	[illegible]	120.00
14	15.00	0.50	120.00
15	40.00	0.44	120.00
16	40.00	0.34	120.00
17	[illegible]	0.39	120.00
18	25.00	0.47	120.00
19	5.00	0.25	120.00
20	25.00	0.79	120.00
21	[illegible]	0.71	117.75
22	87.00	0.74	99.00
23	21.00	0.86	101.00
24	14.00	0.66	123.00
25	[illegible]	0.34	101.00
26	74.00	0.99	126.00
27	34.00	0.09	118.00

PAYLOAD COLD PLATE INPUTS

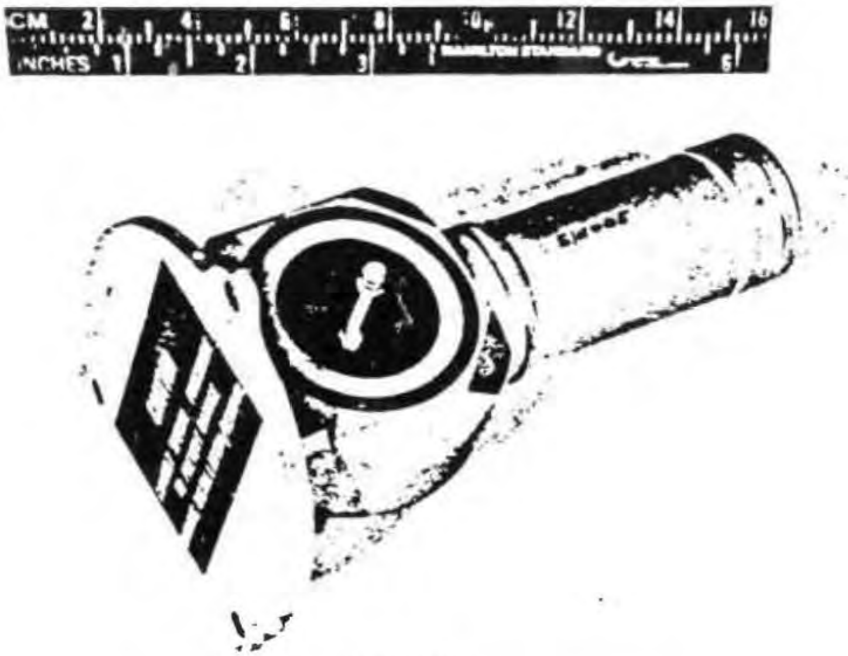
BOX #	Q (WATTS)	BASE AREA (FT**2)	BASE TEMP LIMIT (DEG F)
1	10.00	0.18	120.00
2	20.00	0.12	90.00
3	30.00	0.13	75.00
4	45.00	0.98	99.00
5	67.00	0.86	104.00
6	42.00	6.84	199.00
7	63.00	0.18	100.00
8	77.00	0.46	88.00
9	99.00	0.76	99.50
10	17.00	0.77	120.00
11	40.00	0.00	119.00
12	60.00	0.48	112.00
13	33.00	0.65	109.00
14	77.00	0.33	102.00
15	99.00	0.07	120.00
16	33.00	0.77	129.00
17	[illegible]	0.34	99.00
18	57.00	0.00	103.00
19	83.00	1.00	111.00
20	88.00	0.67	120.00
21	23.00	0.11	141.00
22	93.00	0.46	121.00
23	67.00	0.09	212.00
24	88.00	0.60	119.00
25	88.00	0.75	117.00
26	77.00	0.67	106.00
27	86.00	0.67	109.00

SPACEPLANE COMPUTER PROGRAM
(SAMPLE INPUT DATA)



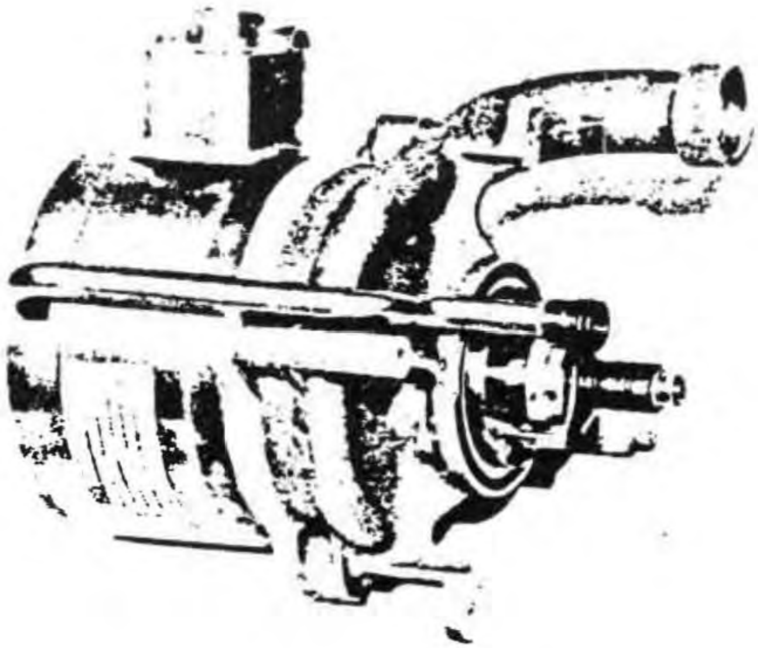
SPACEPLANE COMPUTER PROGRAM SCHEMATIC
(SAMPLE OUTPUT DATA)

APPENDIX C
EC/LSS COMPONENT PICTURES



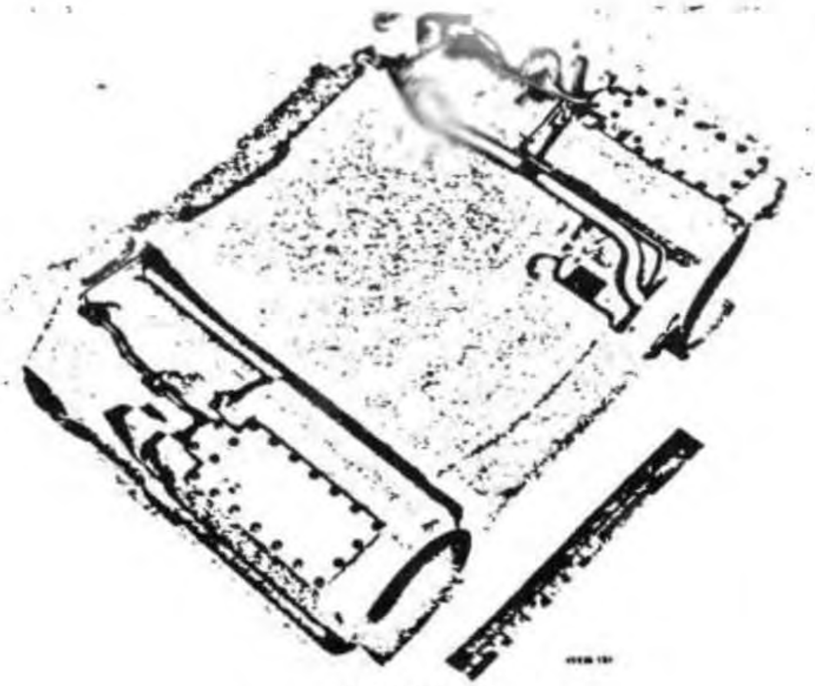
ITEM 121
CHECK VALVE AND VENT FLOW SENSOR

ITEM 122
CARBON DIOXIDE TRANSDUCER



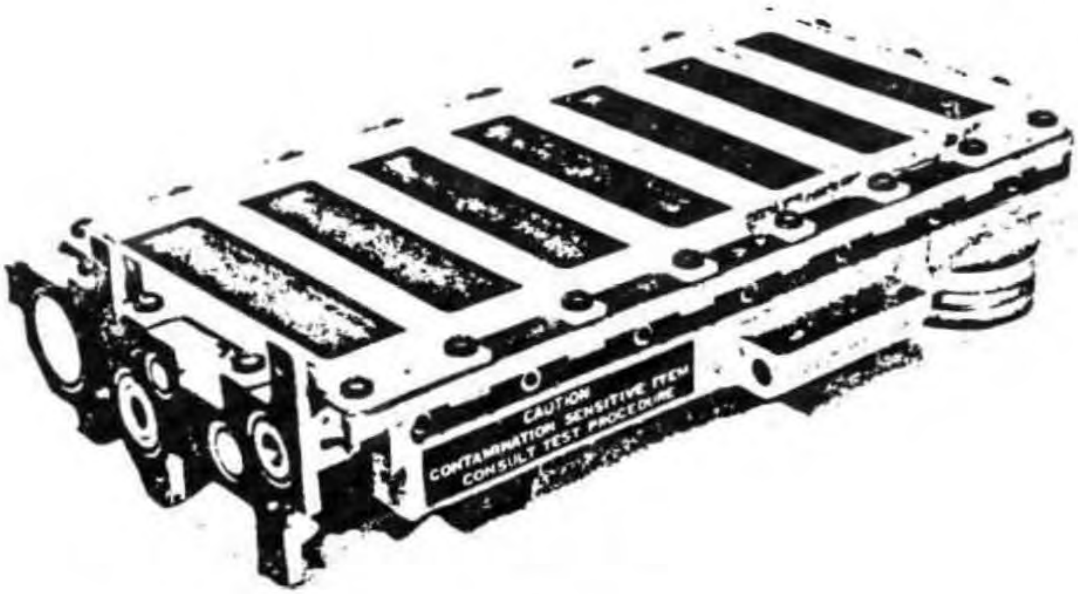
FAN/SEPARATOR/PUMP ASSEMBLY

ITEM 128
CHECK VALVE AND HOUSING



ITEM 131
PRIMARY WATER TANK #1

ITEM 136
FEEDWATER PRESSURE REGULATOR


CAUTION
CONTAMINATION SENSITIVE ITEM
CONSULT TEST PROCEDURE

SUBLIMATOR

ITEM 141
GAS TRAP

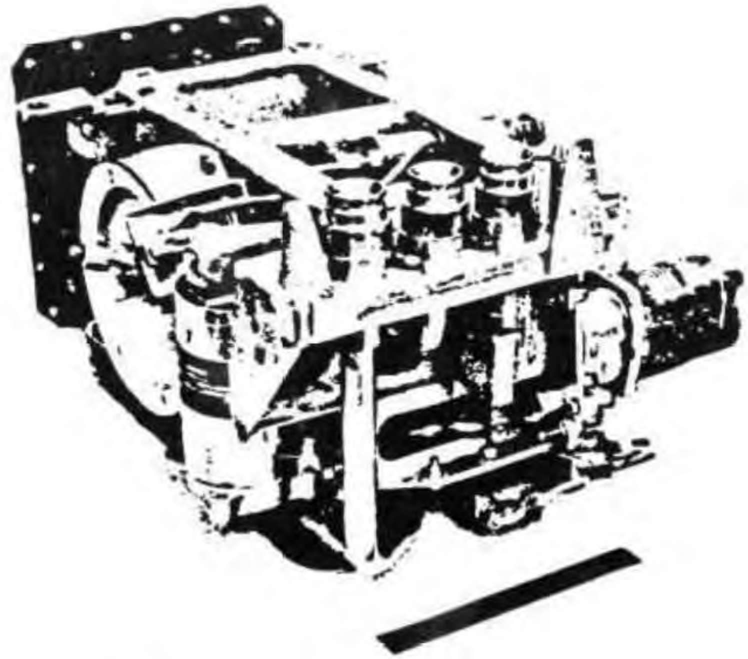
ITEM 143
WATER CHECK VALVE

ITEM 147
NEGATIVE PRESSURE RELIEF VALVE

ITEM 146
POSITIVE PRESSURE RELIEF VALVE

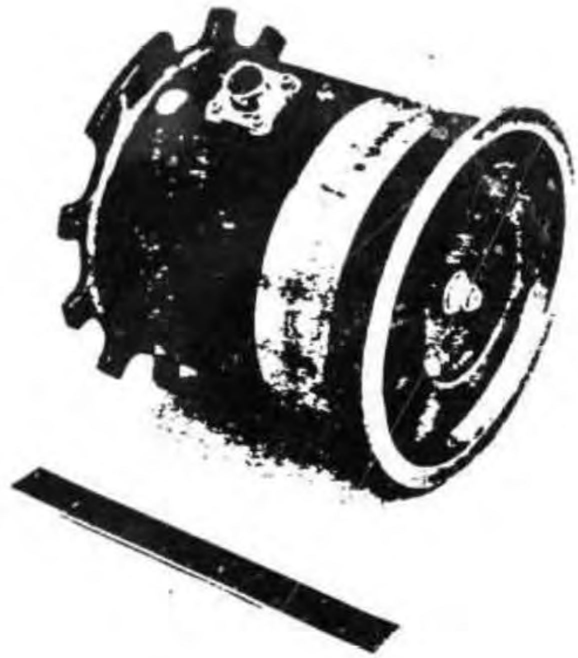
ITEM 321
COOLING CONTROL VALVE

OXYGEN SHUTOFF VALVE



WATER PUMP AND ACCUMULATOR ASSEMBLY

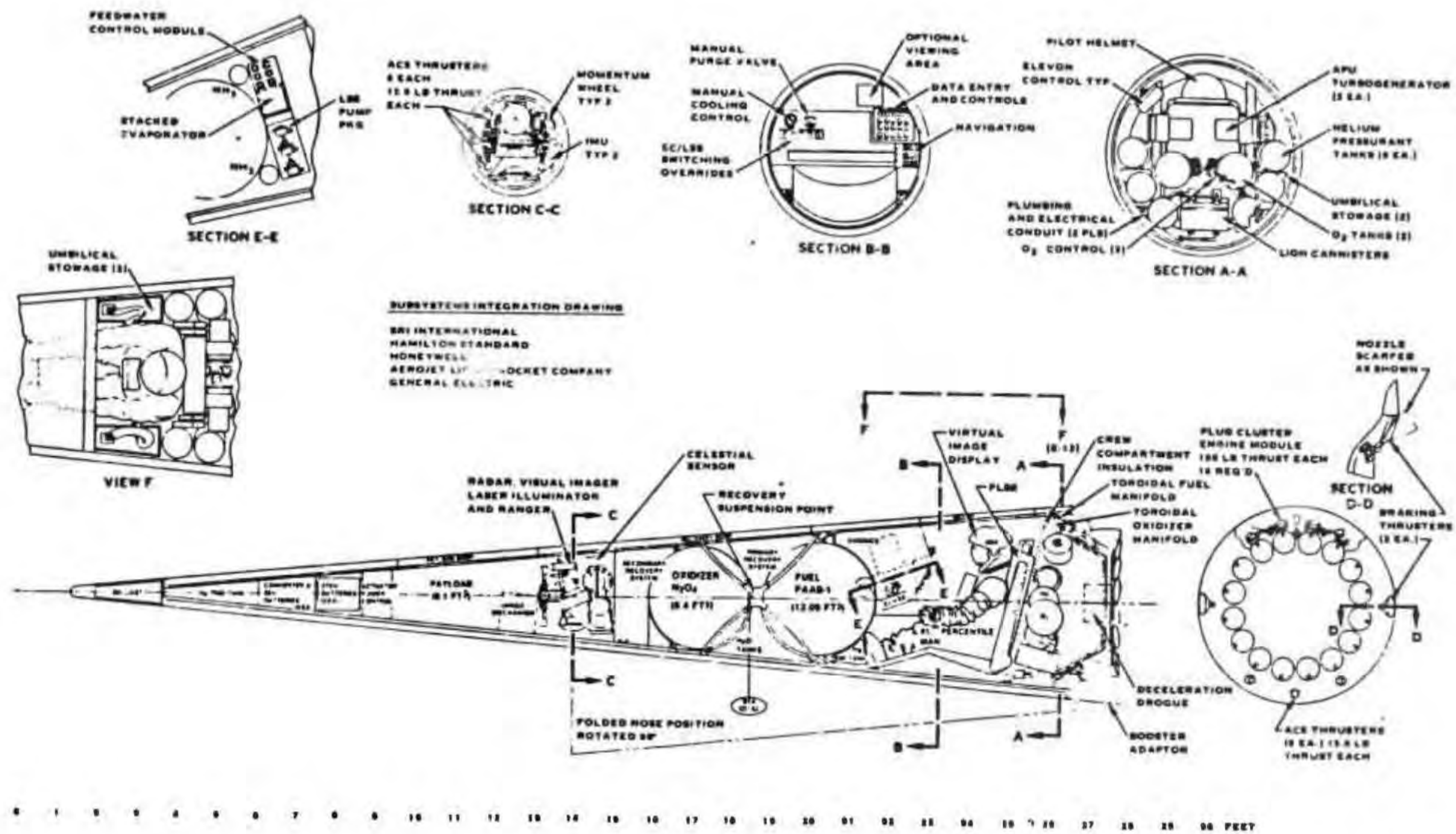
LiOH CANISTER



SHUTTLE AVIONICS FAN

ITEM 113
PRIMARY PRESSURE CONTROL MODULE

APPENDIX D
VEHICLE INTEGRATION LAYOUT



HIGH PERFORMANCE SPACEPLANE

APPENDIX E
SPACEPLANE RECOVERY SYSTEM

(General Electric Re-Entry Systems Division)

'CLASS'

Terminal Descent Control Vehicle Recovery System

Space Plane Application

4-E-2

RE-ENTRY SYSTEMS DIVISION
3198 Chestnut St., Philadelphia, PA 19101

GENERAL ELECTRIC

R.T. MAYER
CHIEF ENGINEER
MILITARY SPACE PROGRAMS
(215) 823-2112

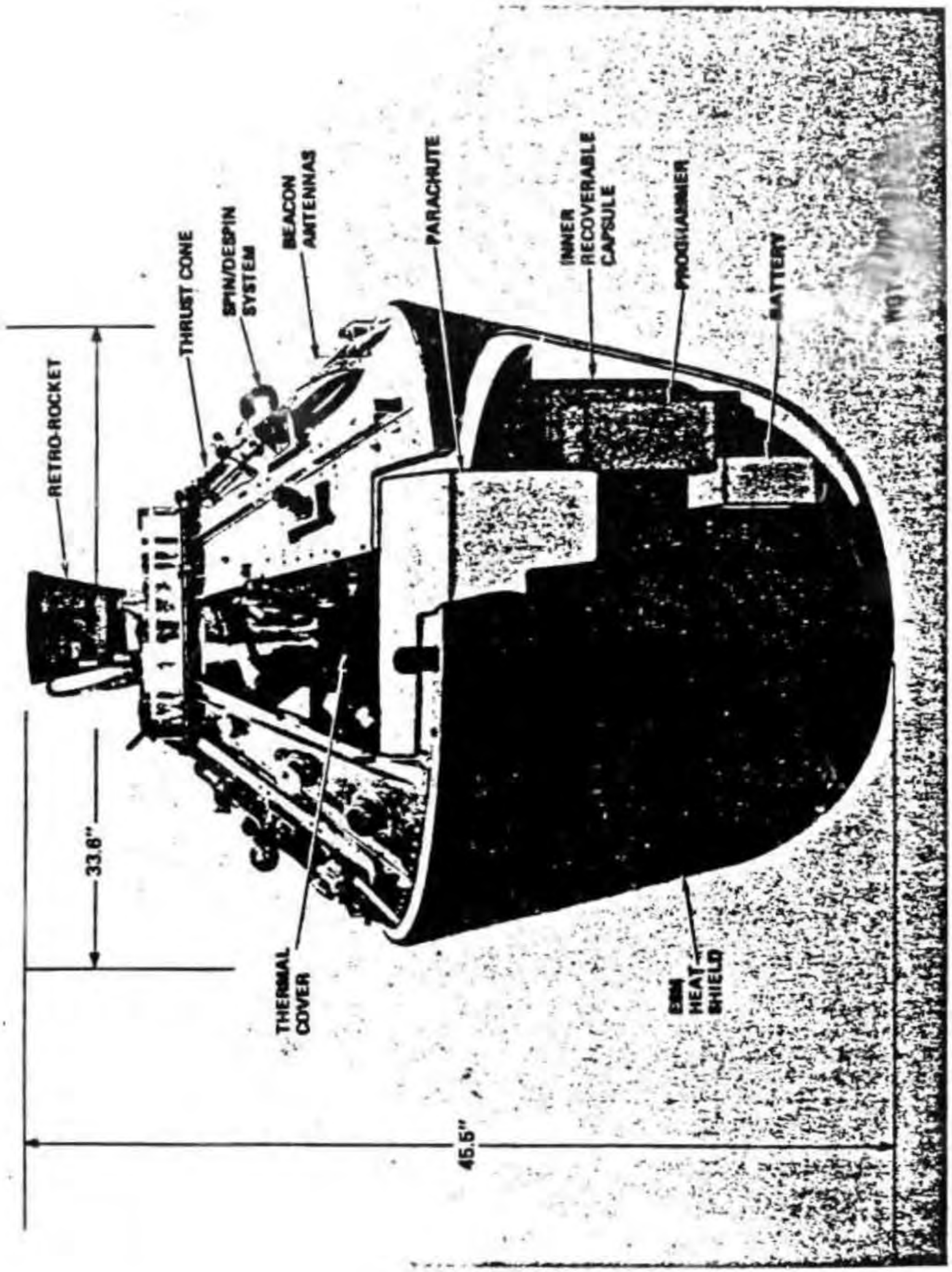
SATELLITE VEHICLE RECOVERY

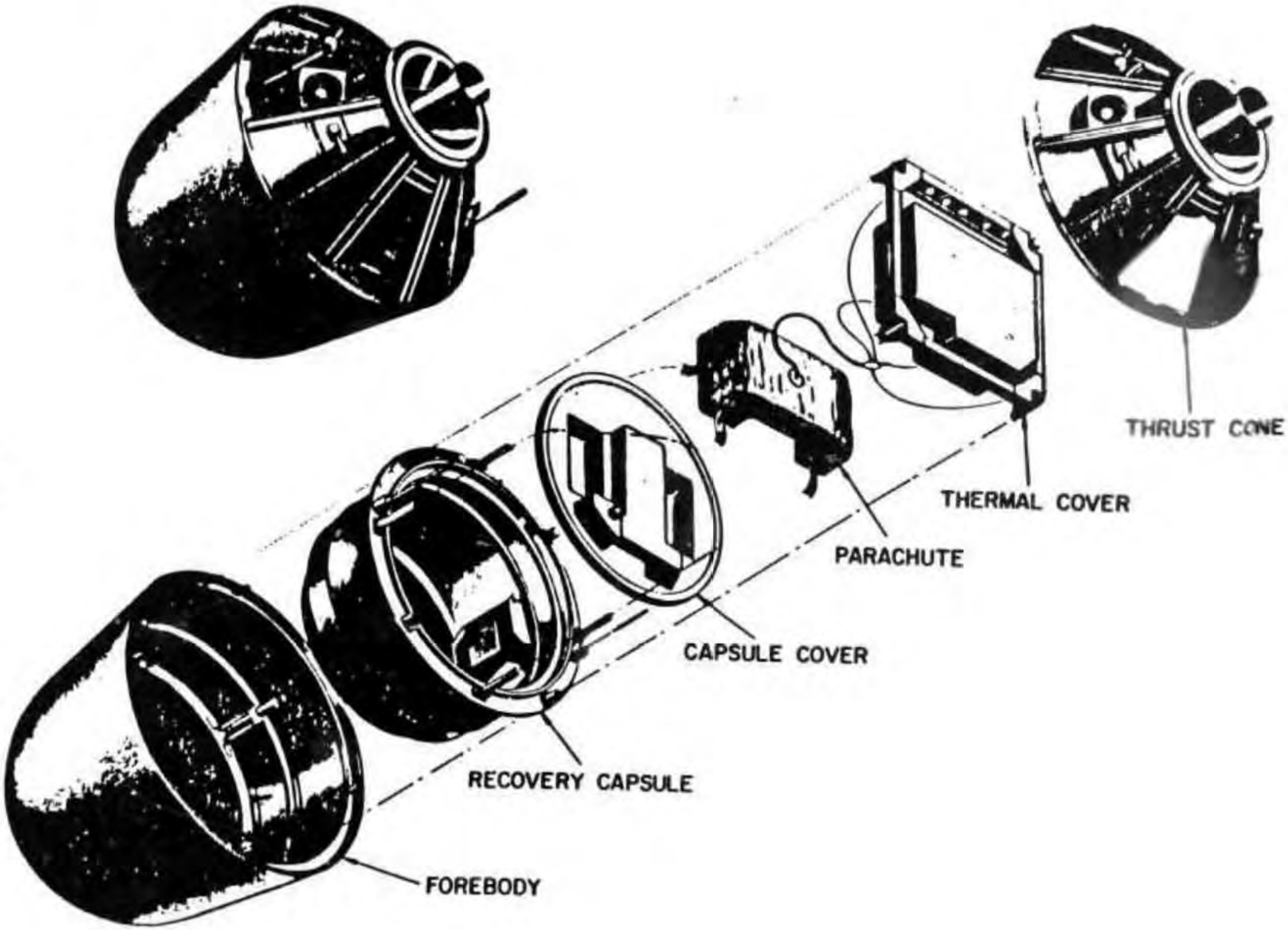
BACKGROUND

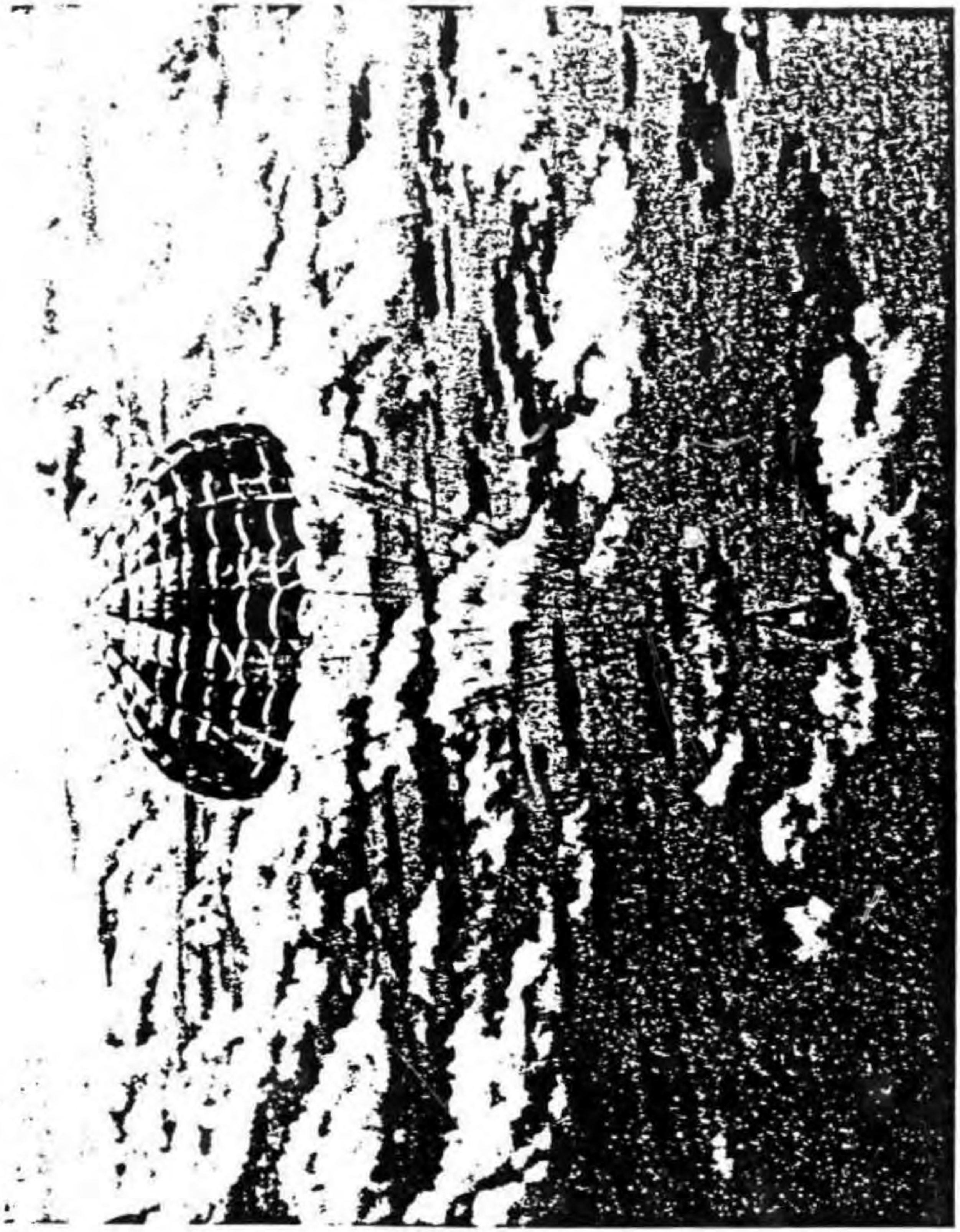
- 21 YEARS OPERATIONAL EXPERIENCE
  (DISCOVERER - AUG. 1960)

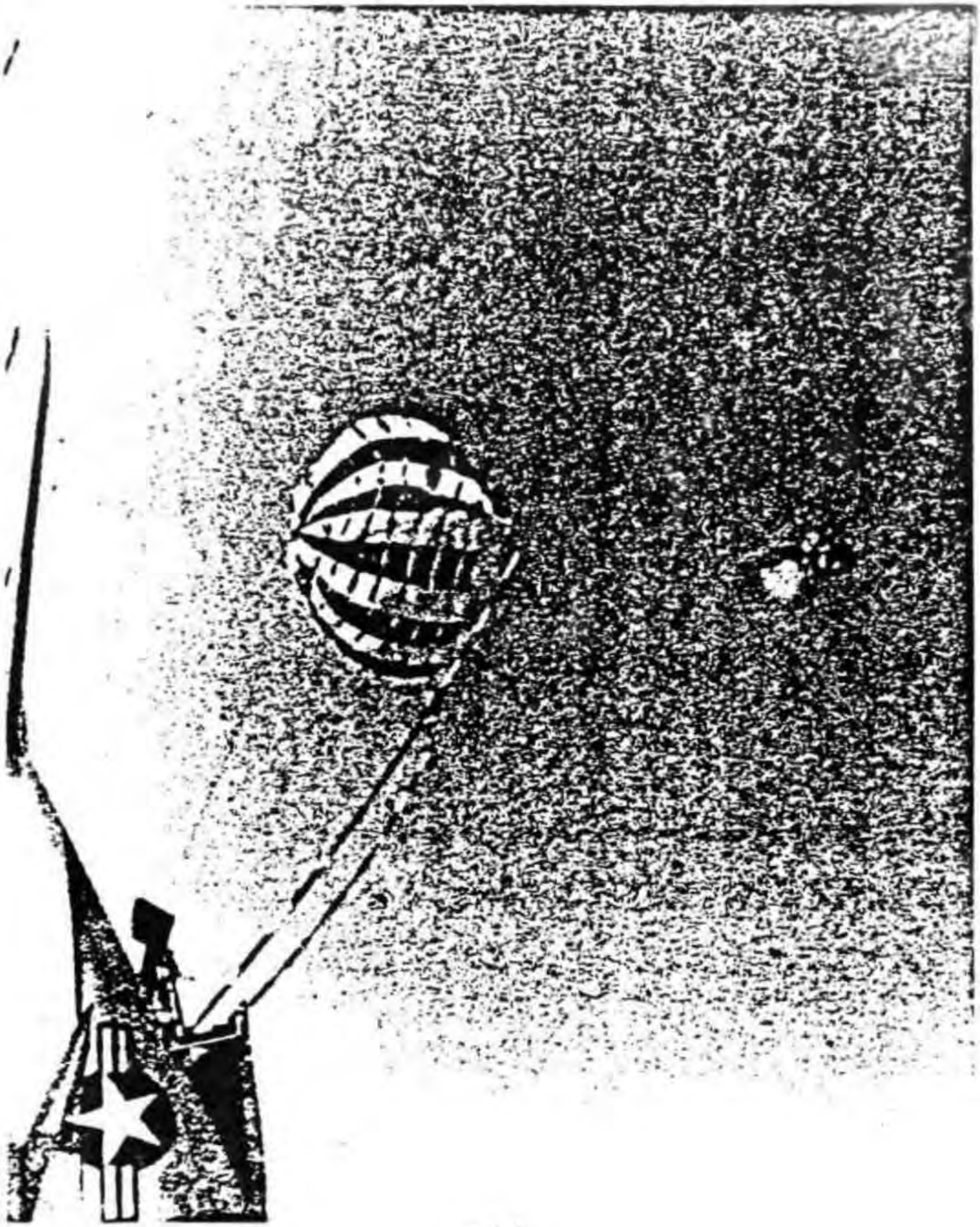
- SEVERAL HUNDRED VEHICLES FLOWN -
  99% SUCCESSFUL RECOVERIES (AIR RETRIEVAL)

- 100% SUCCESS SINCE 1970


RETRO-ROCKET
THRUST CONE
SPIN/DESPIN SYSTEM
BEACON ANTENNAS
PARACHUTE
INNER RECOVERABLE CAPSULE
PROGRAMMER
BATTERY
33.6"
THERMAL COVER
ESM HEAT SHIELD
45.5"


THRUST CONE
THERMAL COVER
PARACHUTE
CAPSULE COVER
RECOVERY CAPSULE
FOREBODY





TERMINAL DESCENT CONTROL VEHICLE RECOVERY SYSTEM

GENERAL

- RADIO CONTROLLED, RAM AIR GLIDING PARACHUTE BASED ON "PARAPOINT" CARGO DELIVERY SYSTEM DEVELOPED BY PARAFLITE INC. PENNSAUKEN, N.J.
- IN SRV RECOVERY APPLICATION IT PROVIDES:
  - CAPABILITY FOR CORRECTION OF REENTRY DISPERSION ERRORS
  - TERMINAL OBSTACLE AVOIDANCE
  - LOW TOUCHDOWN VELOCITY (FLAREOUT MANEUVER)
  - SELF SPILLING (NO TOW) FEATURE IN WATER RECOVERY
- SMALLER AND LIGHTER THAN CONVENTIONAL CHUTE USED FOR AERIAL RECOVERY
- OPERATING MODE OPTIONS
  - AUTO HOMING
  - MANUAL CONTROL
  - "PIED PIPER" MODE
- PERFORMANCE AND RELIABILITY OF BASIC CHUTE CONCEPT DEMONSTRATED BY EXTENSIVE USEAGE IN SPORTS FIELD.

TERMINAL DESCENT CONTROL VEHICLE RECOVERY SYSTEM

TYPICAL PERFORMANCE DATA

- LOW SHOCK - DAMPED OPENING DEPLOYMENT
- 3:1 NOMINAL GLIDE RATIO
- NOMINAL DESCENT VELOCITY
  - VFWD = 60 FPS
  - VDOWN = 20 FPS
- IMPACT VELOCITY (RESULTANT)
  - WITH BRAKING - 10 TO 15 FPS
  - WITH FLAREOUT - 5 FPS (GOAL)
- ACCURACY

  50 FT MISS DISTANCE IN 35 KNOT WIND - MANUAL CONTROL (DEMONSTRATED AT EDWARDS DROP TEST)

LANDING IMPACT

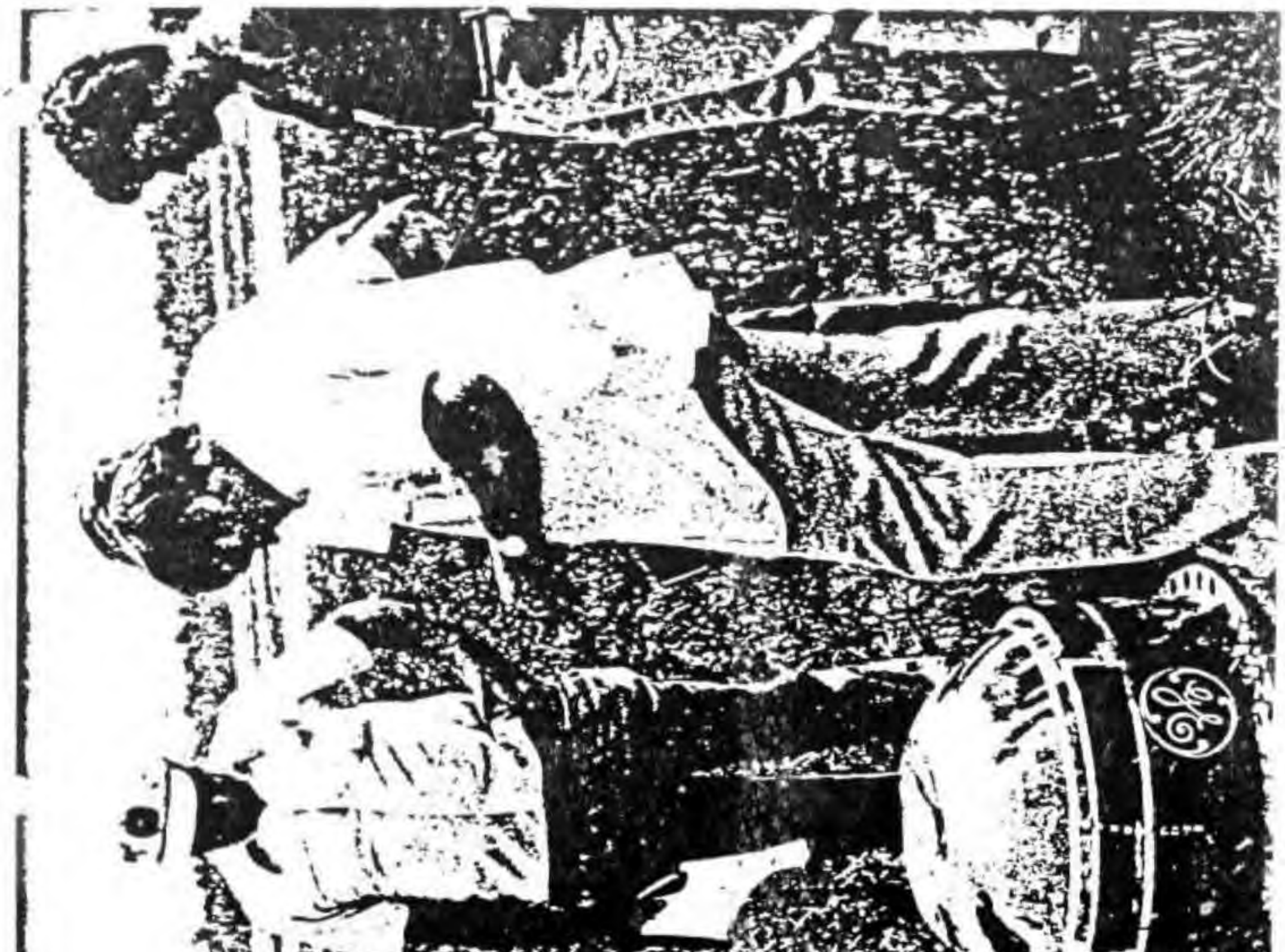
- CURRENT SYSTEM - 688 FT^2 RINGSAIL CHUTE
  - 240 LB PAYLOAD
  - 24 FPS IMPACT VELOCITY
    - SEA LEVEL

- TDC VEHICLE RECOVERY SYSTEM DEMO - 260 FT^2 GLIDING CHUTE
  - 240 LB PAYLOAD
  - 10-15 FPS RESULTANT IMPACT VELOCITY
  - BRAKE BUT NO FLARE OUT

- LIMITED MEASURED DATA ON JUMPERS AND EARLIER PARAGLIDER TESTING INDICATES CONTROLLED FLAREOUT CAN REDUCE DESCENT VELOCITY FROM 40 TO 70%

- AFTER 12 DROPS AT LAKEHURST CUMULATIVE DAMAGE WAS BUCKLING OF UNSUPPORTED .051 ALUM. NOSE - SUPERFICIAL - NO INTERNAL DAMAGE.

- SINGLE DROP AT EDWARDS RESULTED IN SUPERFICIAL NOSE SHELL DAMAGE AS ABOVE.



DEVELOPMENTAL STATUS

CANOPY

- PARAFLITE HAS DEVELOPED NEW CANOPY CONFIGURATION WITH AID OF AEROVIRONMENT INC.
  - LISSIMAN AIRFOIL
  - 3:1 ASPECT RATIO
  - LESS CONTROL MOVEMENT/LOWER CONTROL FORCES
- 800 LB CAPACITY UNIT (12' X 36') BUILT AND UNDER TEST
- 1200 LB CAPACITY UNIT DESIGNED AND STARTING FABRICATION

COMMAND AND CONTROL SYSTEM

- GE IS DEVELOPING A PROTOTYPE WITH FLAREOUT CAPABILITY
- SOME MAJOR ELEMENTS ARE BEING BREAD BOARDED AND TESTED
- DESIGN COMPATIBLE WITH 1500 LB PAYLOAD CHUTE
  - 3000 LB CAPACITY WITH MINOR MODIFICATIONS
  - 5000 LB CAPACITY
    - REWIND MOTORS
    - INCREASE BATTERIES
    - STRENGTHEN ATTACHMENTS

*TEST PROGRAM ON PROTOTYPE RECOVERY SYSTEM COULD BE INITIATED WITH MINIMAL LEAD TIME.

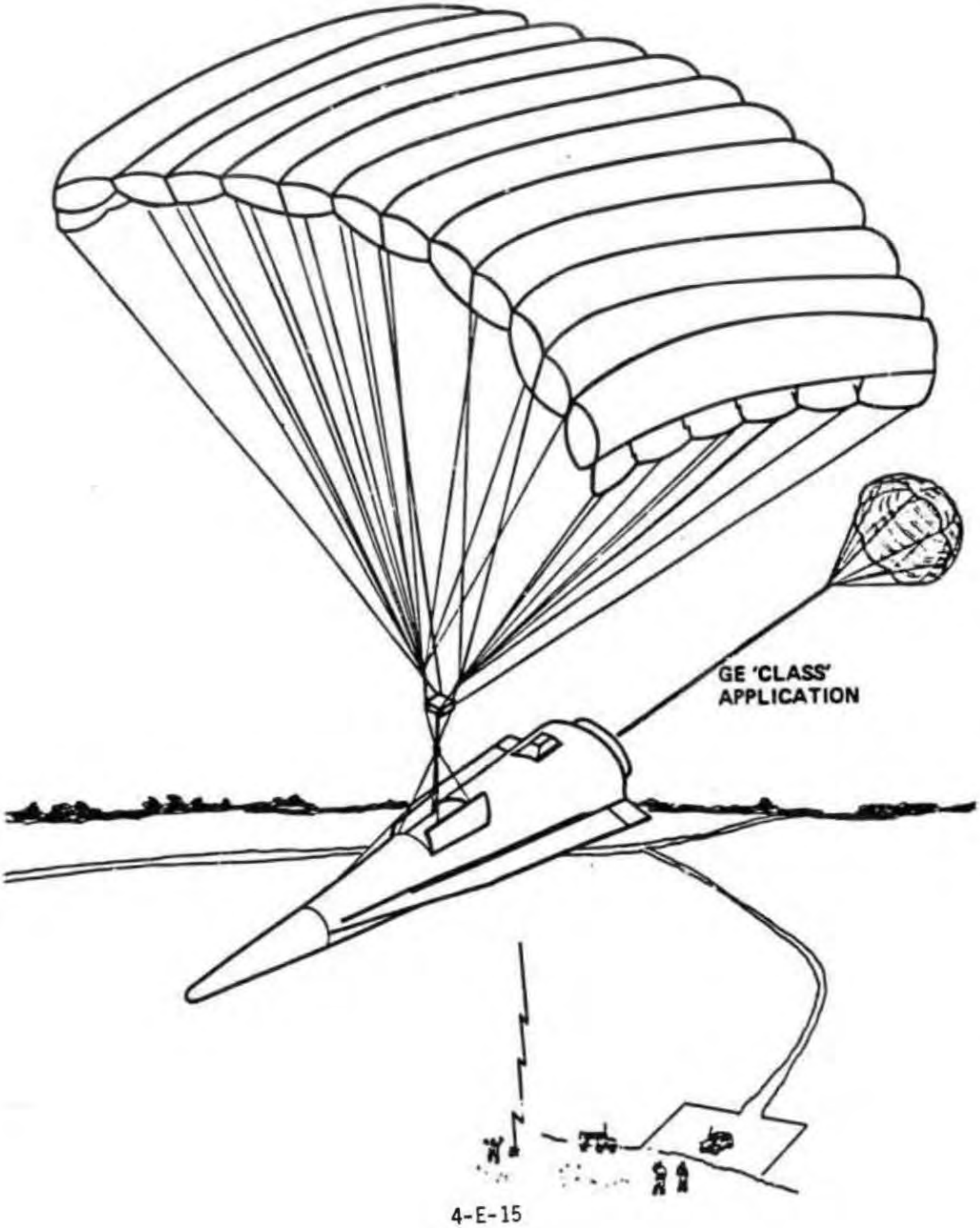
"CLASS"

(CONTROLLED LANDINGS AT SELECTED SITES)

ALTERNATE APPLICATIONS

- SRV RECOVERY (T.D.C.V.R.S.)
- RESEARCH VEHICLE RECOVERY
- MANNED VEHICLE RECOVERY
- PRECISION WEAPON DELIVERY
- ORBITAL HARDWARE RECOVERY

SPACE PLANE
RECOVERY CONCEPT



"CLASS"
ORBITAL HARDWARE RECOVERY

- REQUIREMENTS

  - SOFT LANDING
  - LANDING SITE SELECTIVITY

  $\Big\}$ EQUAL SIGNIFICANCE

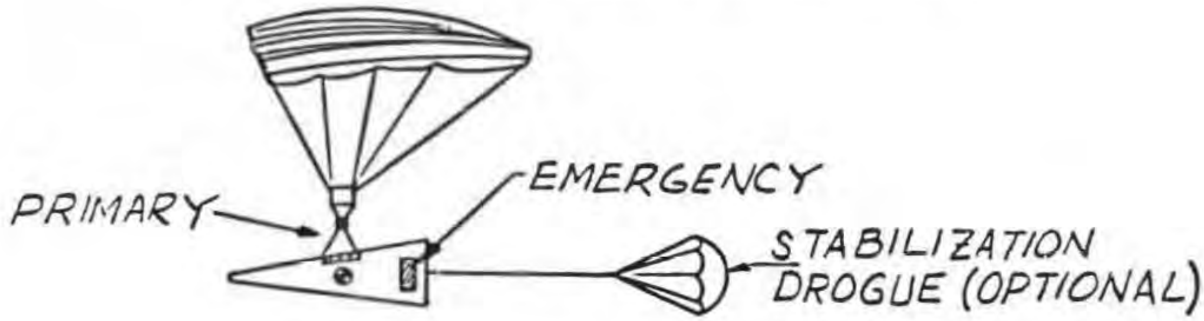
- "CLASS" SYSTEM CAN PROVIDE:

  - SOFT TOUCHDOWN THROUGH AUTOMATED FLARE OUT CONTROL—GROUND SENSING INITIATES FLARE OUT SIGNAL.
  - PRECISE LANDING SITE SELECTIVITY THROUGH AUTO-HOMING AND MANUAL OVERRIDE FROM GROUND OR AIR.
  - FINAL OBSTACLE AVOIDANCE THROUGH TERMINAL MANUAL CONTROL.

CLASS

TERMINAL DECENT CONTROL VEHICLE RECOVERY SYSTEM

SPACE PLANE APPLICATION



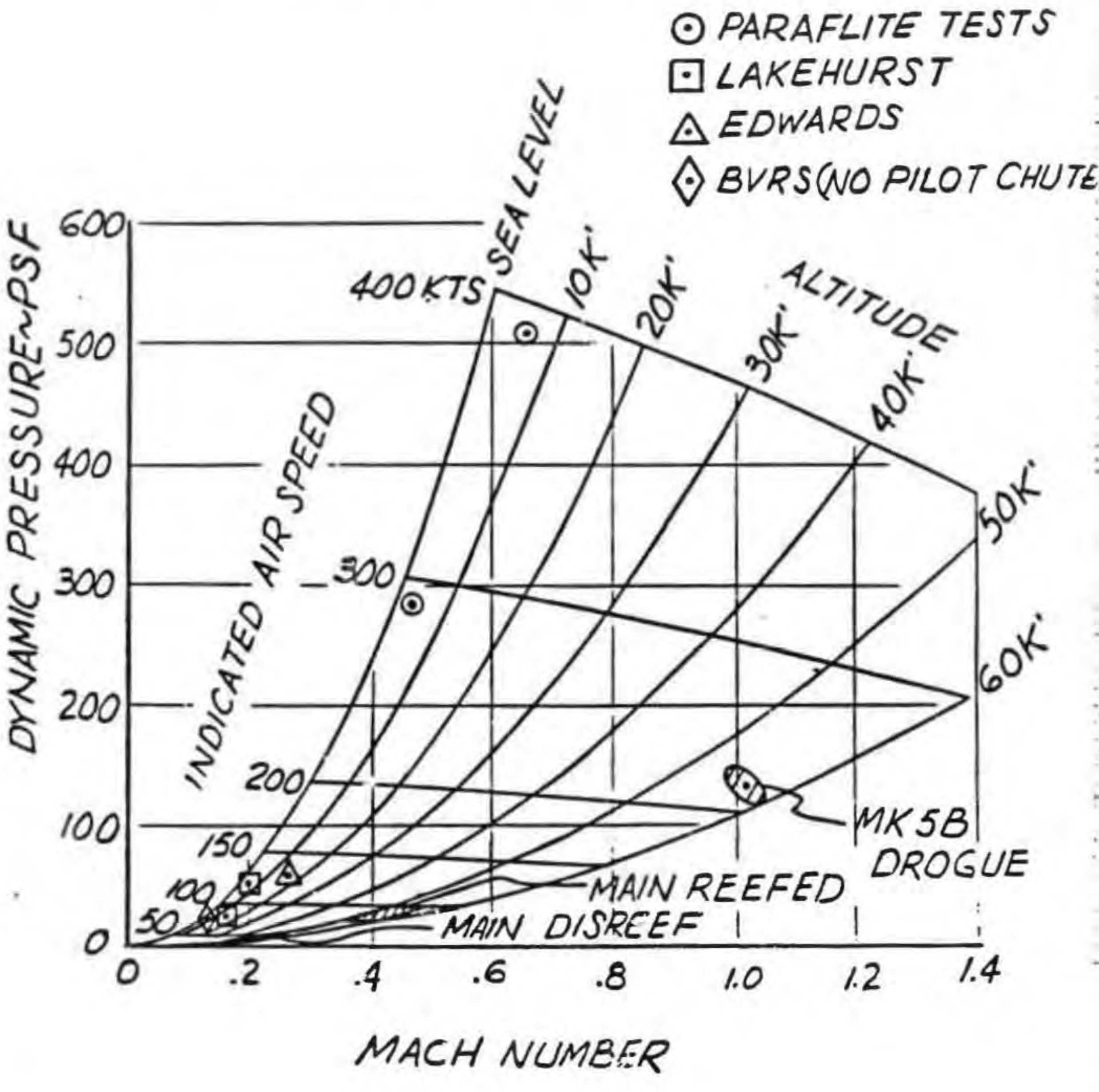
NOTE:-STABILIZATION DROGUE DEPLOYED BY HIGH SPEED DECELERATOR (NOT SHOWN)
-EMERGENCY CHUTE DEPLOYED BY STABILIZATION DROGUE

SUSPENDED WT.	4000 LBS			5000 LBS		
CHUTE SIZE	66' x 33'			74' x 37'		
PRIMARY SYSTEM	WT (LBS)	VOL* (FT^3)	TYPICAL SIZE(IN)	WT (LBS)	VOL* (FT^3)	TYPICAL SIZE(IN)
CHUTE, DROGUE & HARNESS	130	3.3	48×12×10	165	4.1	60×12×10
CONTROL BOX	55	1.0	16×11×10	65	1.2	16×12×11
DEPLOY MORTOR	1	<0.01	1.2d×5	1	<0.01	1.2d×5
EMER. SEP DEVICE	1	**	**	1	**	**
STAB. DROGUE (6'DIA)	2	0.1	12×10×1	2	0.1	12×10×1
TOTALS	189	4.41		234	5.41	
EMERGENCY SYSTEM						
CHUTE & HARNESS	128	3.2	30d×8	163	4.0	30d×10

* PRESSURE PACK AT 40 PCF

** INCLUDED IN CHUTE VOLUME

DEPLOYMENT EXPERIENCE



"CLASS"

GROWTH ITEMS

- INTEGRAL HARD WIRE CONTROL
  - FOR ONBOARD PILOTED LANDING
- BIASED AUTO HOMING
  - RANGE IMPROVEMENT IN AUTOMATIC MODE
- AUTO FLAREOUT
  - PRECISE SENSOR CONTROLLED FLAREOUT FOR MINIMUM VELOCITY TOUCHDOWN

SPACE PLANE

SUGGESTED DEVELOPMENTAL DROP TEST PROGRAM

- PROTOTYPE "CLASS" RECOVERY SYSTEM
  - NEW 1200# CAPACITY CANOPY UNDER DEVELOPMENT BY PARAFLITE
  - NEW PROTOTYPE COMMAND AND CONTROL SYSTEM WITH BUILT IN FLAREOUT CAPABILITY UNDER DEVELOPMENT BY G.E.-RSD.
  - MODIFIED PARAFLITE COMMAND TRANSMITTER INCORPORATING FLAREOUT COMMAND
- 1/5 SCALE SPACE PLANE DROP TEST SIMULATOR
  - APPROX. 5 FT. LENGTH
  - 1000 LBS.
  - REPRESENTIATIVE MASS PROPERTIES
  - VARIABLE ATTACHMENTS
  - INSTRUMENTATION
- DEMONSTRATION DROP TESTS AT LAKEHURST NAS (OR SIMILAR)
  - HELICOPTER DEPLOYMENT
  - 6K TO 10K FT. DROP ALTITUDE
  - STATIC LINE INITIATION
- TEST OBJECTIVES
  - EVALUATE SYSTEM CONTROLLABILITY
  - FLAREOUT EFFECTIVITY/LANDING IMPACT
  - SPACE PLANE SUSPENSION
    - . LOCATION/C.G. TRAVEL SENSITIVITY
    - . TRIM ANGLE
    - . POINTING STABILITY-NEED FOR DIRECTIONAL DROGUE
  - DEPLOYMENT BEHAVIOR

TERMINAL DESCENT CONTROL VEHICLE RECOVERY SYSTEM

DROP TEST DEMONSTRATION

- FUNDED DEMONSTRATION DROP UNDER REPRESENTATIVE RECOVERY CONDITIONS
- TEST PLANNING, DIRECTION AND INTEGRATION - GE
- FLIGHT TEST UNIT
  - OFF THE SHELF 'PARAPOINT' SYSTEM
  - DUMMY PAYLOADS AND INSTRUMENTATION - GE
- 12 DROPS LAKEHURST N.A.S. – SEPTEMBER 1980
- 6 DROPS EDWARDS AFB – DECEMBER 1980
- FILMED SUMMARY (LAKEHURST SEGMENT)

5.0 STORABLE PROPULSION SYSTEM

5.1 SUMMARY

In support of the Spaceplane Examination, Aerojet Liquid Rocket Company (ALRC) served as the storable propellant liquid rocket propulsion subcontractor. ALRC completed a preliminary design and evaluation of an onboard propulsion systems as reported herein. Sandia National Laboratories and Hamilton Standard supplied vehicle profile and vehicle internal layout integration data respectively.

5.1.1 Study Requirements

The primary requirement of this study was to conceptually design and evaluate a storable, pressure fed, liquid rocket propulsion system for the Spaceplane vehicle. The initial requirements for this propulsion system are listed below:

- o Storable propellants (N_2O_4/MMH)
- o Pressure fed system
- o Plug cluster engine (PCE) configuration
- o Thrust level: 4500 to 6500 lbF
- o Chamber pressure: TBD
- o PCE diameter: 30 to 60 inches
- o Propellant weight: approx. 1200 lbM

On the basis of these general requirements, a study plan was formulated wherein the contracted program was performed in three phases:

- o Phase I - Concept Design Synthesis
- o Phase II - Parametric Analysis/Preliminary Design
- o Phase III - Propulsion System Integration and Interface Definition

These phases and the subtasks contained therein are depicted in Figure 1, Spaceplane Study Program, as revised on 30 April 1982 by agreement with SRI International. This agreement extended the technical period of performance from 30 April to 31 May 1982. The final report was prepared in June and submitted to SRI in July 1982. The month of July was reserved to respond to questions during the final report review by SRI.

5.1.2 Study Results

The main features of the onboard propulsion system, including the resulting plug cluster engine (PCE) and reaction control system (RCS), for application to the Spaceplane vehicle are as follows. All tasks were completed within the contract period of technical performance.

Plug Cluster Engine (PCE)

- o 16 modules (thrusters)
- o Module thrust: 188 lbF
- o Total thrust: 3058 lbF
- o Module chamber pressure: 100 psia

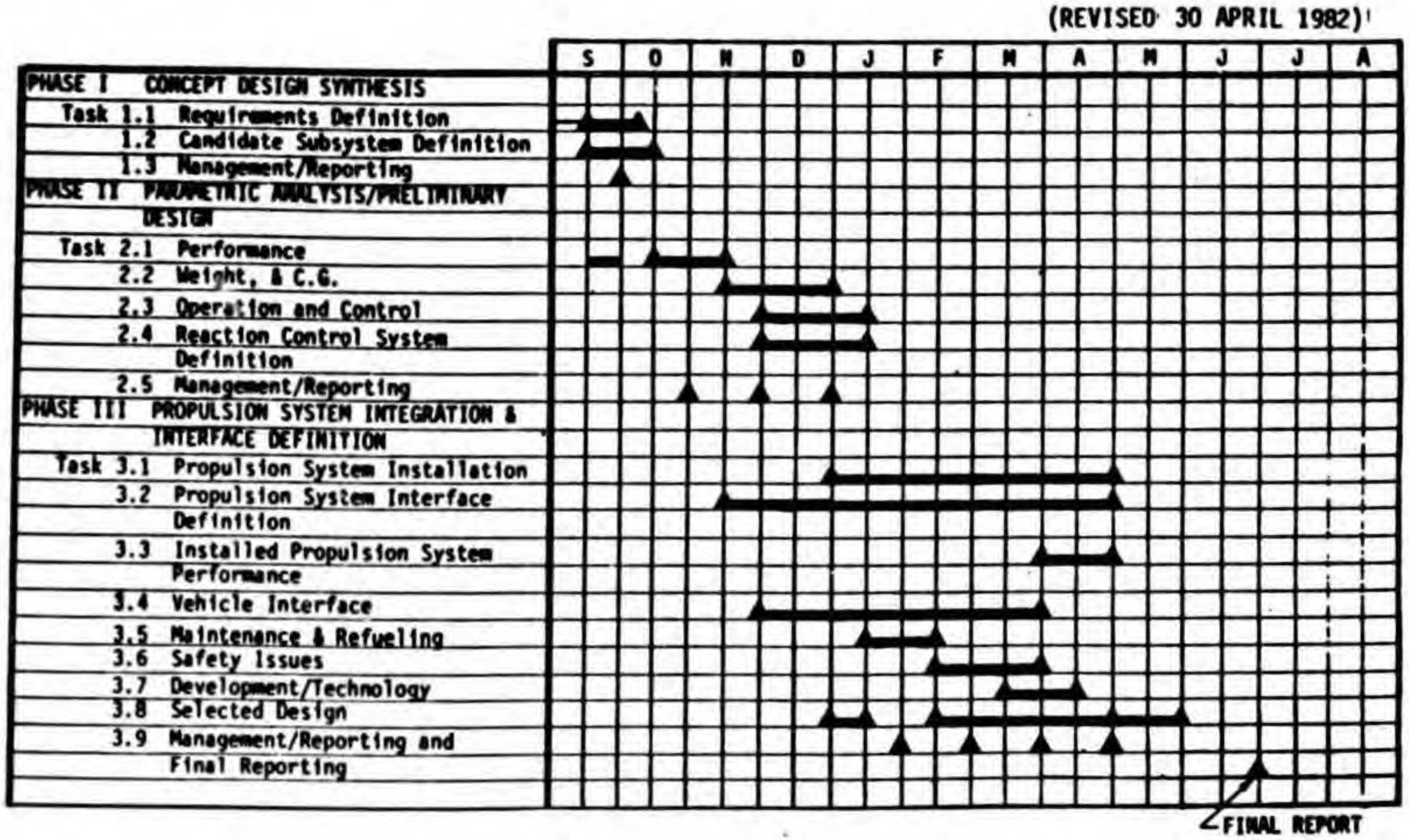


Figure 5-1 Spaceplane Study Program Schedule

- Propellants: N_2O_4/PAAB-1*
- Mixture ratio: 1.2

FIG 1

- Total weight: 85.1 lbM
- Module nozzle area ratio: 37:1
- PCE area ratio: 85.7
- PCE diameter: 43.4 inches
- PCE length: 13.9 inches
- Module tilt angle: 14.6^{o}
- Module delivered Isp: 311.7 seconds
- PCE delivered Isp: 316.9 seconds

*A propulsion system using N_2O_4/MMH is described the Task 3.8 (Selected Design) discussion in this report.

RCS

- RCS thrust: 15.0 lbF/module
- RCS configuration:
  8 Foreward thrusters
  6 Aft thrusters
  2 Aft mounted retro thrusters (188 lbF each)
- RCS weight: 45.0 lbM

Propellant Tanks

- Total propellant load: 1400 lbM
- Spherical oxidizer tank: R = 15.1 in, t = .047 in.
- Spherical fuel tank: R = 16.2 in, t = .050 in.
- Oxidizer tank weight: 21.4 lbM
- Fuel tank weight: 26.5 lbM
- Propellant tank (ox and fuel) pressure: 200 psia

Pressurization Subsystem

- The pressurization subsystem weight: 78.0 lbM

Figure 2 shows a Spaceplane vehicle configuration utilizing the propulsion system described above. Figure 3 shows additional details of the foreward RCS thrusters. Figures 4 and 5 display rear and side views, respectively, of the PCE including the 6 aft mounted RCS thrusters. Figure 6 shows two aft mounted retro thrusters. The two aft thrusters represent two different valve location/nozzle configurations. Obviously, only one configuration would actually be implemented on the Spaceplane vehicle.

The major conclusions and recommendations resulting from this study are:

- The Spaceplane concept is viable from a propulsion viewpoint

- The 3058 lb_f thrust level selected results in acceptable vehicle acceleration. Selection of the optimum thrust will be done after further study.

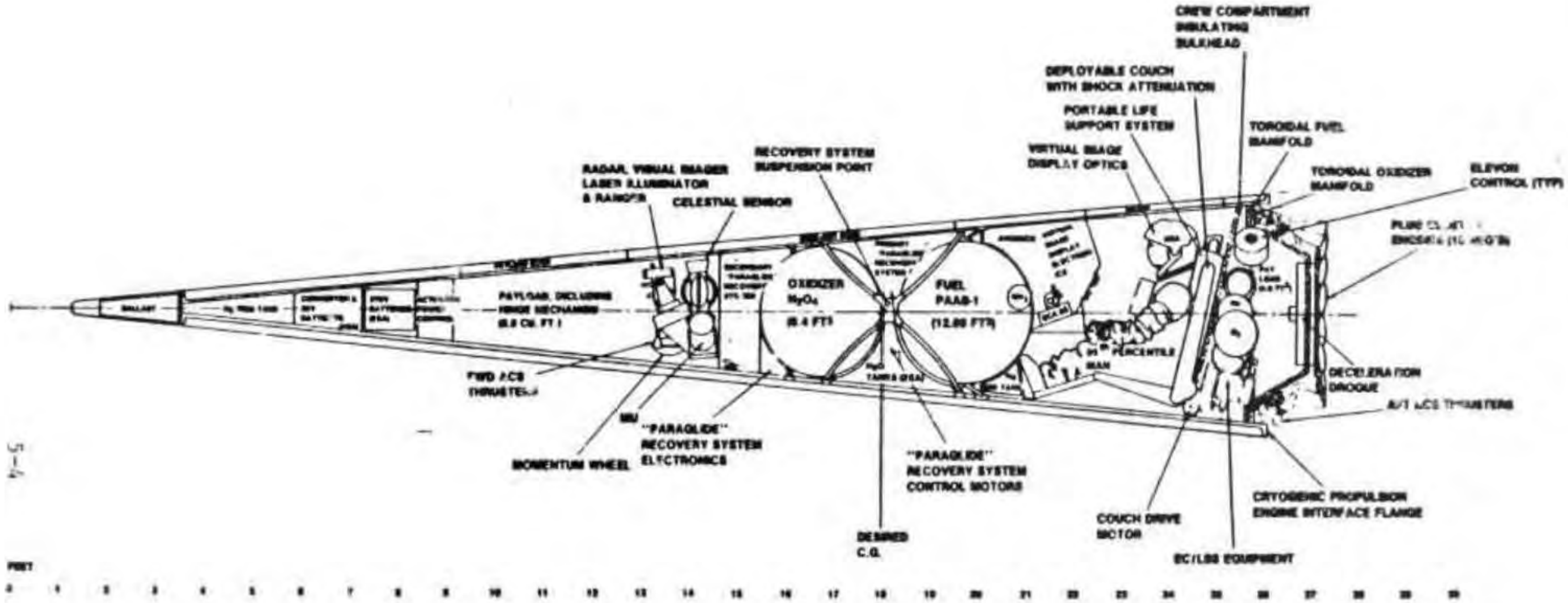


Figure 5-2 Hamilton Standard Integrated Spaceplane Vehicle

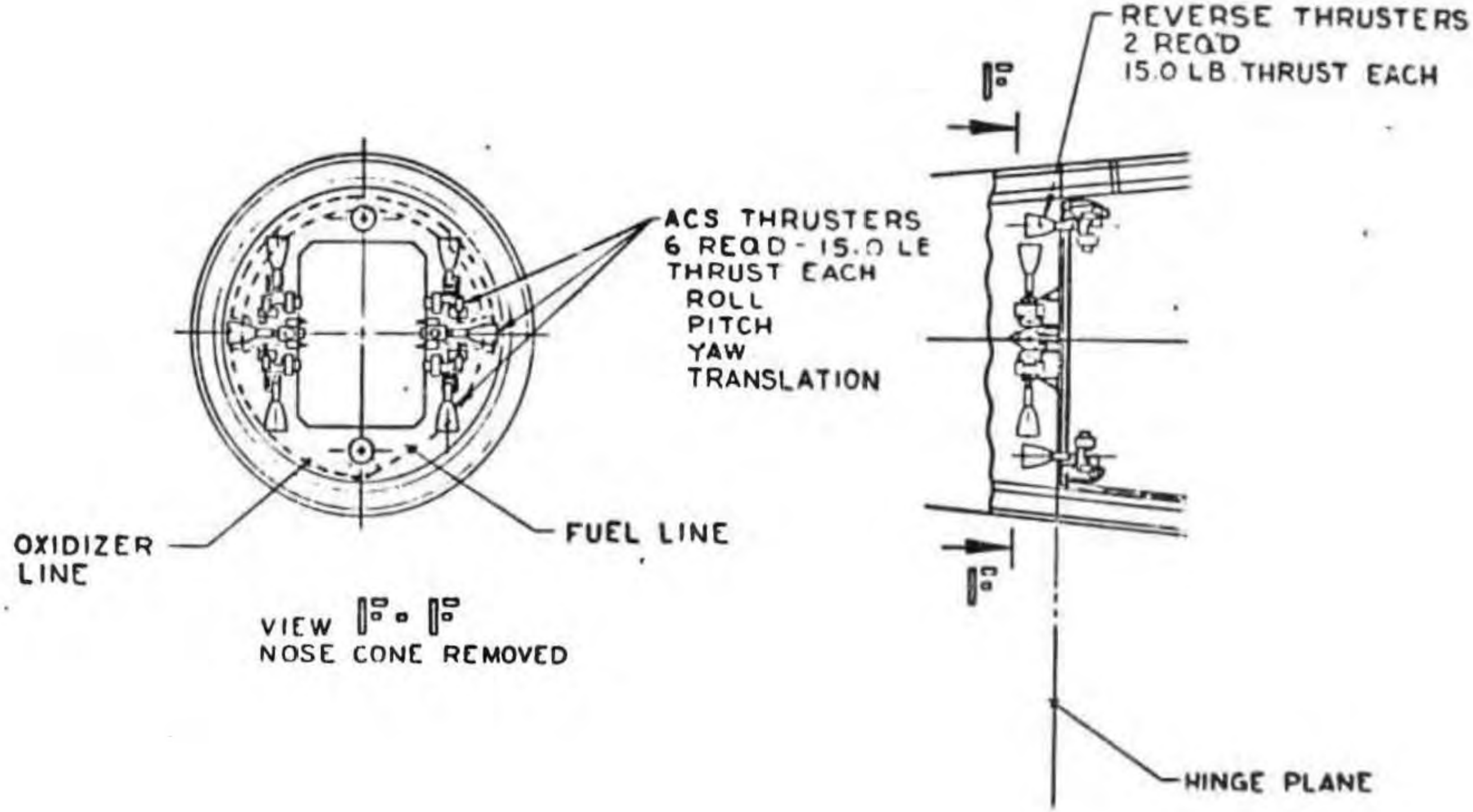


Figure 5-3 N_2O_4/PAAB-1 Foreward RCS Thrusters

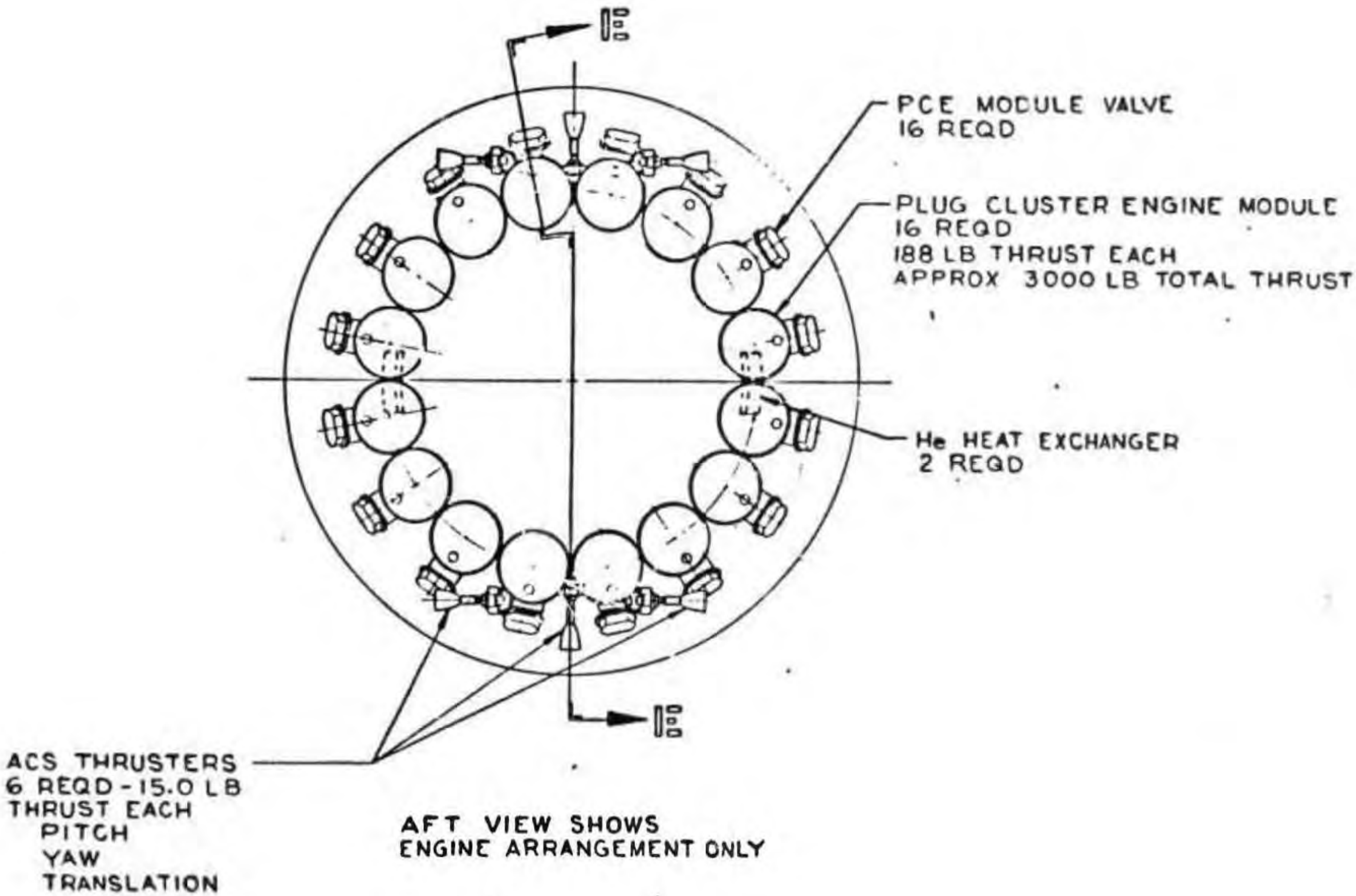


Figure 5-4 **N_2O_4/PAAB-1 Plug Cluster Engine (Rear View)**

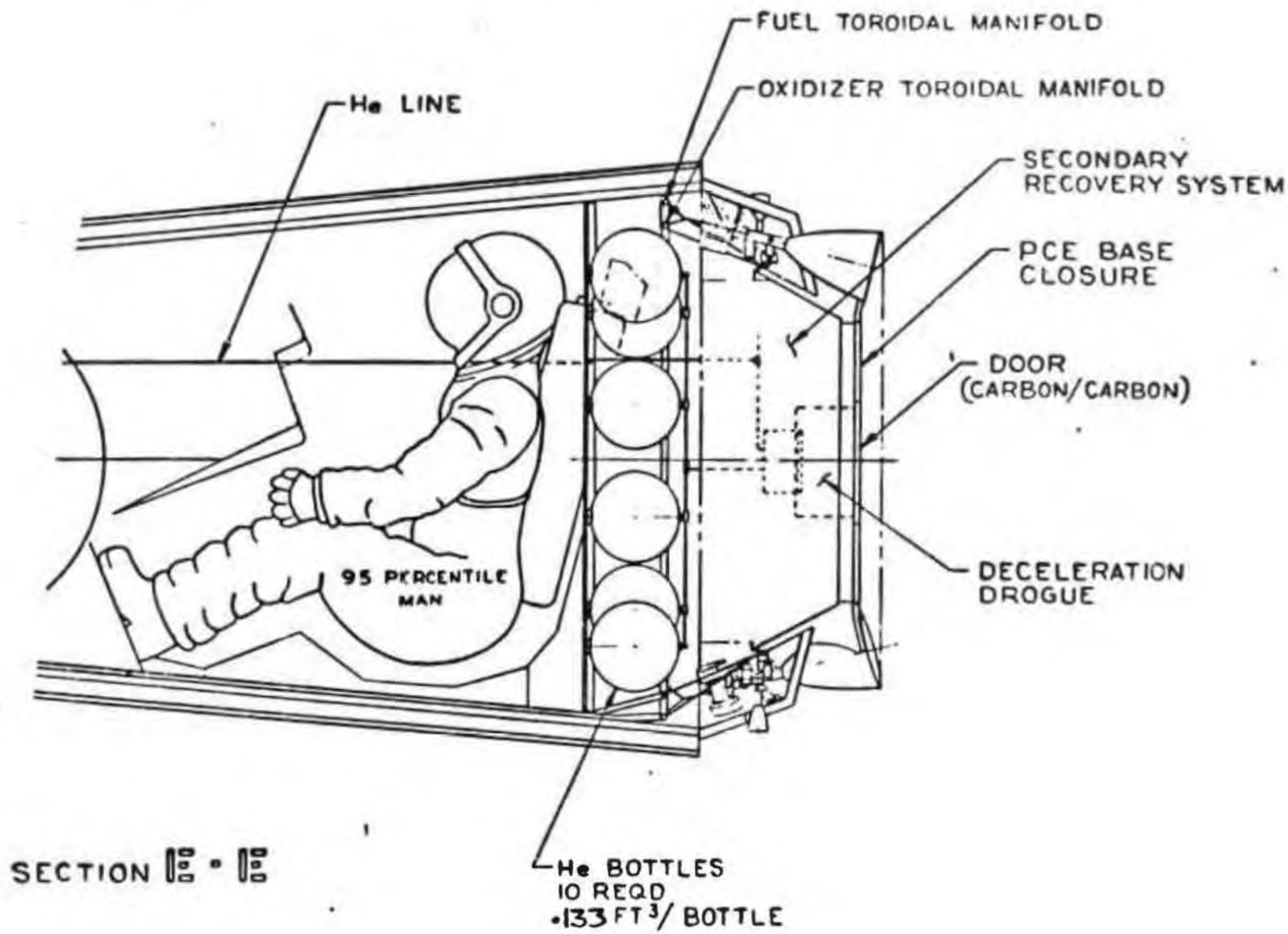


Figure 5-5 N_2O_4/PAAB-1 Plug Cluster Engine (Side View)

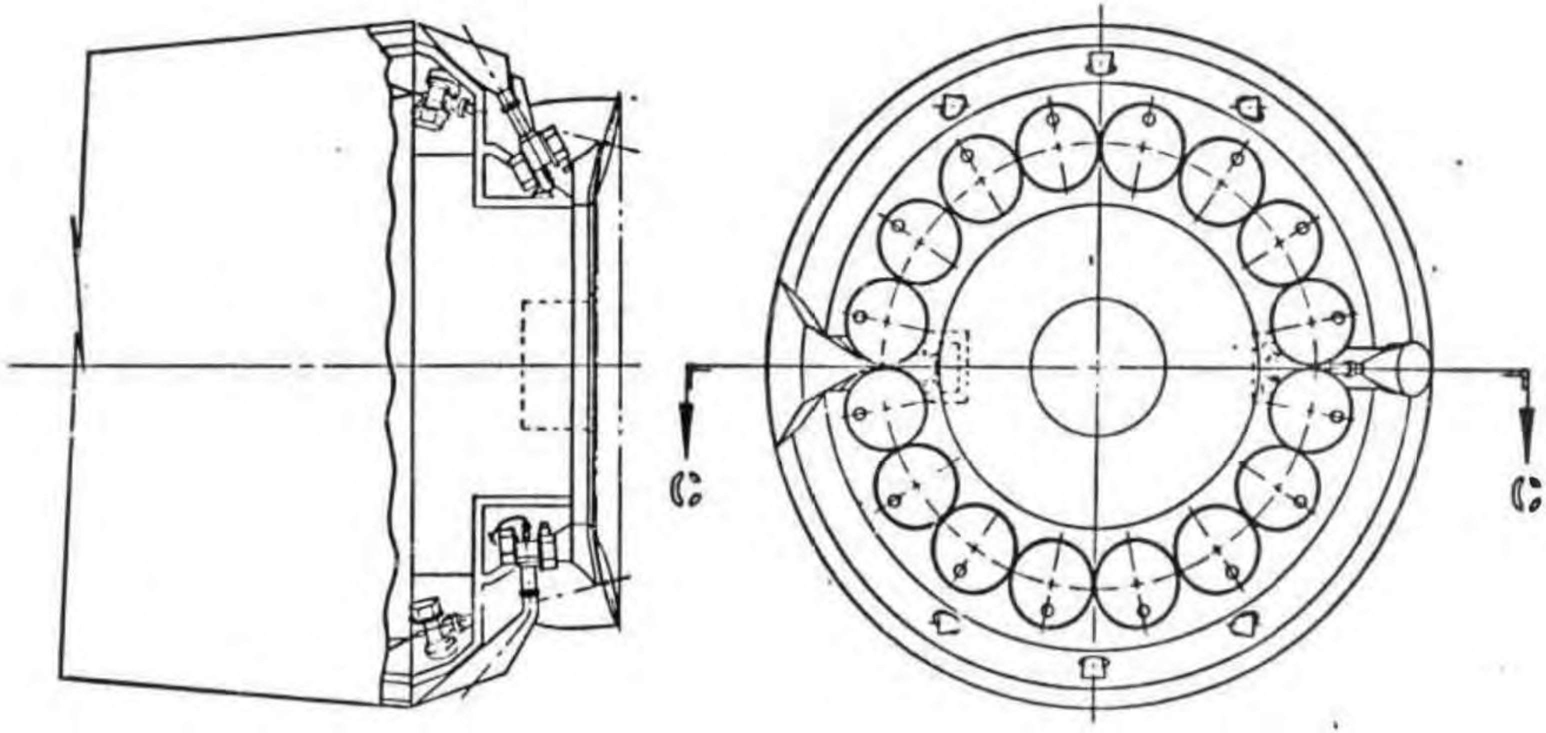


Figure 5-6 **N_2O_4/PAAB-1 Retro Thrusters**

- Demonstrated ALRC 100 lbF and 5 lbF bipropellant engines, with minor modifications, meet the Spaceplane internal primary and secondary propulsion requirements
- The selected Spaceplane internal propulsion system is a flexible concept which can be optimized for many conditions, (eg. various launch modes)
- Extensive Spaceplane internal propulsion system modeling capability (prediction of propulsion system weight, c.g, envelope, performance, etc.) exists at ALRC
- Recommendations
- Establish firm vehicle and propulsion system design requirements
- Prepare Spaceplane preliminary propulsion system design, based on above requirements
- Begin development of a long-life, N_2O_4 - compatible, elastomeric tank diaphragm
- Begin design modifications to ALRC 5 and 100 lbF thrusters for Spaceplane PCE and RCS application

5.2 INTRODUCTION

5.2.1 Study Background

ALRC submitted its original proposal (LR801085) to SRI in August 1980 in which the use of ALRC's Low Cost Axial Engine (LCAE) was recommended for use on the Spaceplane (then called Space Cruiser) vehicle. The Low Cost Axial Engine, designated AJ10-203 by ALRC, was built in the mid-1970's under Air Force Contract F04611-76-C-0066 and demonstrated satisfactory operation over the thrust range required for the space cruiser.

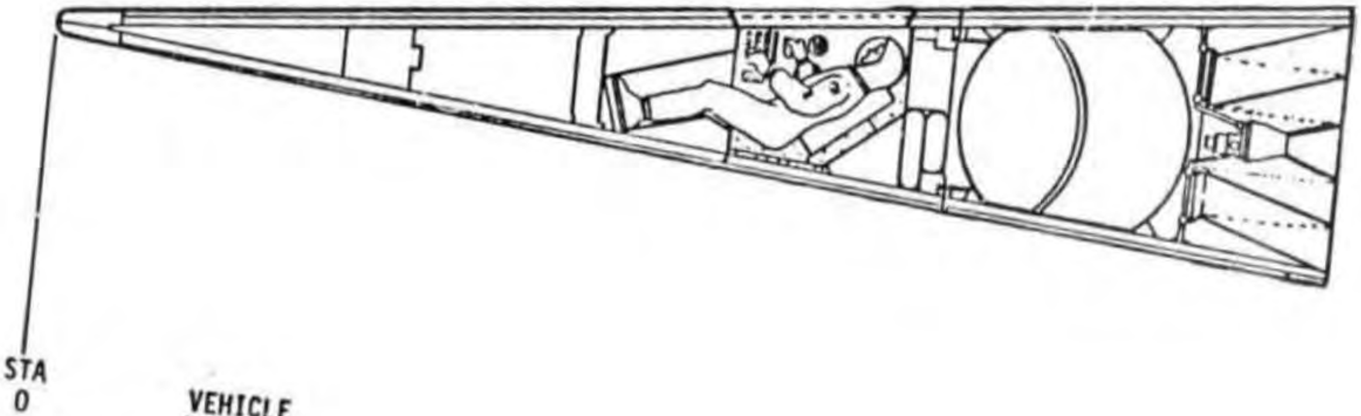
ALRC proposed to combine the Low Cost Axial Engine with many of the pressurization and feed system components which had been qualified for the Thor Delta Upper Stage and Japanese Delta or N-2 program. Propellant tanks, two each, fuel and oxidizer were to be of a design similar to those produced for the Space Shuttle OMS, Concord SST and MX Fourth Stage by Aerojet Manufacturing Company (AMCO). All of the components considered were in production or were commercial items. The resulting main propulsion system assembled in the Spaceplane was as shown in Figure 7.

It was also proposed that the ALRC 5 lbF thruster AJ10-181-2 be used to maintain vehicle stability and perform docking and maneuvers in space. This engine was developed under AFRPL Contract F04611-73-C-0061 and demonstrated high performance both steady-state and pulse mode in a small bipropellant thrust class engine.

The components selected for the pressurization system were based on cost, procurement, and technical criteria. The cost aspect related to the flight certification. Using flight qualified and flight demonstrated components would minimize the certification testing for this program as well as the design activity. The procurement criteria relates to the fact that all of the pressurization components were (1) currently in production for other space programs, (2) would be fabricated for planned space programs by the time the space cruiser program is in development, or (3) were commercial off the shelf hardware. The fact these components are "in production" provides relatively short lead times. The selection criteria for the propellant and feed system hardware were the same as the pressurization components.

The proposed propellant tanks were a design developed by Aerojet Manufacturing Company. These all titanium tanks were compatible with both fuel and oxidizer. This tank design had been flight qualified on the PEP experimental program, MX validation, Space Shuttle OMS pod and on the Concord. They had also demonstrated excellent operating characteristics during long dynamic loading conditions.

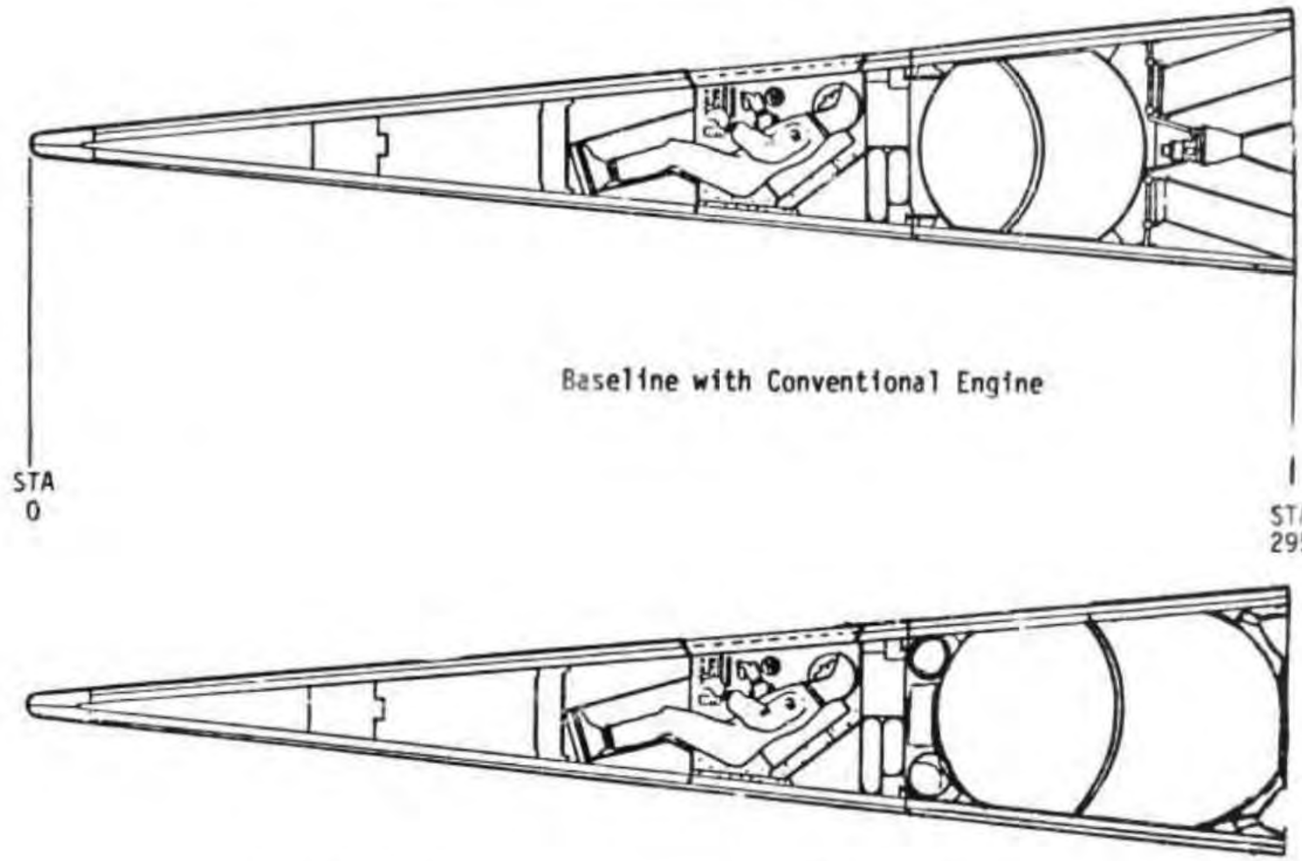
Subsequently, (i.e. after Aug 1980), it became apparent that one way of increasing the Spaceplane propellant volume and therefore payload, was through utilization of the plug cluster engine (PCE) concept. A Spaceplane based comparison of the LCAE and PCE was made. The important results of this comparison are shown in Figure 8. Although the vehicle dry weights and resulting values in Figure 8 were subsequently shown to be optimistic, the relative values demonstrated the benefit of the PCE. It was on this basis that SRI requested an evaluation of both the LCAE and PCE



VEHICLE

IDEAL VAC ΔV	7097
GROSS WT	4000
VAC DEL Isp	318
BURNOUT WT	2000
PROPELLANT WT	2000

Figure 5-7 Spaceplane Vehicle Description (Original Proposal)



	Baseline Vehicle	Clustered Engine Vehicle
Ideal Vac ΔV	7,097	11,320
Gross Wt	4,000	6,450
Vac Del Isp	318	323
Burnout Wt	2,000	2,180
Propellant Wt	2,000	4,280

Figure 5-8 **Comparison of Vehicle/Engine Concepts**

for Spaceplane application. The analysis resulted in a decision to eliminate the LCAE for Spaceplane application.

5.2.2 Study Requirements

The study was divided into 3 phases with subtasks. The subtask requirements are outlined below by phase.

1. Phase I - Conceptual Design Synthesis

Task 1.1 - Requirements Definition

The Spaceplane onboard propulsion system requirements (thrust, chamber pressure, etc.) were to be established with the concurrence of both SRI and Sandia National Laboratories at Albuquerque (SNLA).

Task 1.2 - Candidate Subsystem Definition

The study was to consider, as a minimum:
1 engine geometry (plug cluster configuration)
4 tank designs and materials
3 propellant control subsystems
3 tank pressurization subsystems
2 methods of thrust vector control (TCV)
2 methods of thrust magnitude control (TMC)

2. Phase II - Parametric Analysis/Preliminary Design

Task 2.1 - Performance

Both propulsion system delivered specific impulse (Isp) and Spaceplane vehicle (SP) were to be presented as functions of nozzle area ratio, chamber pressure, propellant mixture ratio and plug cluster engine configuration.

Task 2.2 - Weight and Center of Gravity

Propellant weights and propulsion system dry weights for all of the combinations resulting from Task 1.2 were to be provided, plus the center of gravity for one Spaceplane vehicle.

Task 2.3 - Operation and Control

The operation and control of a conceptual propulsion system was to be described.

Task 2.4 - Reaction Control System (RCS) Definition

The thrust level and location of RCS thrusters to provide vehicle roll, yaw and pitch were to be defined on the basis of moment requirements to be supplied by SNLA.

3. Phase III - Propulsion System Integration & Interface Definition

Task 3.1 - Propulsion System Installation

A selected propulsion system design was to be updated per vehicle requirements supplied by SNLA, RCS nozzle configurations were to be specified and propulsion system thrust load paths were to be defined.

Task 3.2 - Propulsion System Interface Definition

Fluid, power, command and control interfaces were to be established.

Task 3.3 - Installed Propulsion System Performance

Total impulse (specific impulse multiplied by total propulsion system burntime), SP and potential improvements to either parameter were to be defined.

Task 3.4 - Vehicle Interfaces

Mechanical, electrical, control and fluid interfaces were to be described.

Task 3.5 - Maintenance and Refueling

In-situ refurbishment and/or replacement (R&R) of four major propulsion subsystems were to be discussed. Refueling provisions and auxiliary drop tanks were to be described. Any vehicle requirements thus affected were to be identified.

Task 3.6 - Safety Issues

A preliminary failure mode analysis of each propulsion subsystem was to be performed.

Task 3.7 - Development/Technology Issues

Any technology requirements and justifications were to be identified.

Task 3.8 - Selected Design

The optimum propulsion system design was to be selected. A layout drawing of this design was to be made. A written description (preliminary engine specification) of this design was to be provided, including performance and operations information. The justification for the design selection was to be described.

There were, in addition to the tasks described above, three reporting tasks (1.3, 2.5 and 3.9). These reporting tasks together required a written monthly progress report at the end of each month during the technical period of performance, except for the last month when a final report was to be written. Therefore, there were eight monthly reports and one final report.

Additionally, two technical presentations were made at TI/TD (Technical Information/Technical Direction) meetings held at Aerospace Corporation in January and May of 1982. A final presentation may be made in August or September of 1982.

5.3 TASK 1.1 REQUIREMENTS DEFINITION

The essential requirement of Task 1.1 was to define the onboard propulsion system requirements and other Spaceplane vehicle parameters impacting the onboard propulsion with the concurrence of both SRI and SNLA. These propulsion system requirements or goals, and vehicle parameters, as understood at ALRC at contract initiation, are listed below.

- o Technology readiness: 1985
- o Spaceplane vehicle length: 22 ft (or less)
- o Propulsion system dry weight (not including tanks and pressurant gas): 200 lb
- o Propulsion system dry weight (including tanks and pressurant gas): 550 lb
- o Main engine propellant weight (N_2O_4 and MMH): 1200 lb
- o Maximize Isp (minimum will be Isp of single 100 lb engine with SS chamber and e = 150:1 nozzle: 310 seconds)
- o Start/Stop response: TBD
- o Multiple restart capability: 3-4 in space restarts
- o Reusability: (50 to 100 missions)
- o Non-shifting c.g. location during main engine burn
- o Man-rated reliability
- o Minimum vehicle : 2500 ft/sec
- o Main engine heat transfer to vehicle structure; Maximum is TBD
- o Main engine propellants: N_2O_4/MMH
- o Thrust magnitude control is required; throttling ratio: TBD
- o Thrust vector control is required; total vector movement: TBD

During the course of the study, it became apparent that there were additional requirements to be defined. Several TBD (To Be Determined) values were defined while other values were revised. The resulting requirements list is shown below.

- o Technology readiness: 1985
- o Spaceplane vehicle length: 25 ft (or less)
- o Propulsion system dry weight (excluding tanks and pressurant gas): 220 lb
- o Propulsion system dry weight (including tanks and pressurant gas): 550 lb
- o Main engine propellant weight (N2O4/MMH or N_2O_4/PAAB-1): 1200 lb or more
- o Isp which maximizes Spaceplane
- o Start/Stop response: TBD
- o Multiple restart capability: Minimum: 3 to 4 restarts in space; Maximum: TBD
- o Reusability: 50 to 100 missions
- o Minimum shifting of c.g. location during main engine burn
- o Man-rated reliability
- o Minimum vehicle Wy: 2500 ft/sec
- o Main engine heat transfer to vehicle structure: maximum: TBD
- o Main engine propellants: N_2O_4/PAAB-1 or N_2O_4/MMH
- o Thrust magnitude control: 8:1
- o Thrust vector control: required; total vector movement: 14.2^{o} round pattern

- o Vehicle c.g. location: 69% of vehicle length (measured from front of vehicle), Station 221.6
- o RCS:

Function	Max Acceleration Angular ($^\circ/sec^2$)	Max Acceleration Linear (ft/sec^2)	Max Time Required to give Vehicle = 0.02 ft/sec (sec)
Pitch	4.0	--	--
Yaw	4.0	--	--
Roll	5.0	--	--
Retro Thrust (-X Axis)	--	0.40	0.10
Forward Thrust (+X Axis)	--	0.40	0.10
Translation (±Y Axes)	--	0.40	0.10
Translation (±Z Axes)	--	0.40	0.10

- o Main engine thrust level: 3000 lbF

The justification for revisions and additions to the initial requirements is discussed here. Initially, these additional requirements were identified:

- o Vehicle c.g. location: approximately 66% of vehicle length (measured from front of vehicle)
- o RCS (pitch, yaw and roll capability) rates: TBD
- o Vehicle 3-axis translation capability (in addition to main engine and RCS) to provide g levels of: TBD

PAAB-1 (Proprietary Aerojet Amine Blend No. 1) was added as an internal propulsion fuel. There were two major reasons for making this recommendation:

(1) PAAB-1 can serve as both a bipropellant and monopropellant. This means that the Spaceplane APU's will not require separate monopropellant tanks. This represents a savings in weight and complexity and, hence, an improvement in reliability.

(2) The N_2O_4/PAAB-1 propellant combination is also higher-performing than the N_2O_4/MMH combination - from an Isp as well as an Isp-bulk propellant density standpoint.

Some of the available data on PAAB-1 are listed in Table I.

TABLE I
PAAB-1 CHARACTERISTICS

Freezing Point, $^\circ$F	+24
Specific Gravity (at 60°F)	0.99
Vapor Pressure, psia (at 60°F)	0.2
Maximum Isp Mixture Ratio (N_2O_4 Oxidizer)	1.48
Mixture Ratio of Equal Tank Volumes (N_2O_4 Oxidizer)	1.48

The reaction control subsystem (RCS) requirements were formulated on the basis of discussions at the January TI/TD meeting. A more in-depth discussion of the RCS is found in the Task 2.4 (RCS Definition) description in this report.

A thrust level of 3,000 lbF was also selected on the basis of discussions at the January TI/TD meeting. The optimization of the plug cluster engine (PCE) was performed on the basis of this thrust level.

The thrust magnitude control (TMC) capability shown is inherent in a 16-module plug cluster engine (PCE) configuration. 2:1 throttling of individual PCE modules would increase this total PCE throttling capability to 16:1. Presently, the 8:1 throttling capability appears to be adequate, with the possible exception of rendezvous and docking maneuvers where the RCS is available.

The TVC capability shown (14.2^{o} round pattern) is also a result of the baseline (16 module) PCE configuration.

The format of the RCS requirements was revised to allow easier evaluation of a given RCS. The new format is the same as that used to describe the capabilities of the baseline RCS.

The Spaceplane vehicle length was changed to 25 ft (was 22 ft) based on discussions with both H/S and SNLA. The consensus was that 25 ft would be required to accommodate the pilot in an upright position and to provide the propellant volume necessary to achieve a total Spaceplane of 2500 ft/sec.

The main propellant weight was changed from "1200 lbM" to "1200 lbM or more." Analysis showed that 1200 lbs of propellant would only provide a of approximately 2000 to 2200 ft/sec as opposed to the required 2500 ft/sec.

The optimum Isp was defined as the delivered Isp which would result in the maximum SP . This value will vary considerably depending on the available diameter for the internal propulsion system or PCE and the required PCE thrust level. For the Spaceplane application, Isp increases with either a thrust decrease or diameter increase. The resulting Isp could range from approximately 280 to 320 seconds. Ultimately, the Isp of the N_2O_4/PAAB-1 system was 316.9 seconds. The corresponding value for the N_2O_4/MMH system was 312.9 seconds. In the case of the Spaceplane, the maximum possible Isp did not result in the highest SP .

5.4 TASK 1.2 CANDIDATE SUBSYSTEM DEFINITION

The primary objective of this task was to evaluate, and recommend on a priority basis, the following Spaceplane onboard propulsion subsystems:

- 1 engine geometry (plug cluster configuration)
- 4 tank designs and materials
- 3 propellant control subsystems
- 3 tank pressurization subsystems
- 2 methods of thrust vector control (TVC)
- 2 methods of thrust magnitude control (TMC)

The numbers of subsystems were considered to be minimum values.

The bases for comparing these propulsion subsystems and parameters include considerations derived from the propulsion system requirements (see Task 1.1 Requirements Definition). The evaluation of candidate subsystems and selection of recommended baseline subsystems for each of the propulsion subsystems categories listed above is considered here in the same order as listed.

5.4.1 Engine System Geometry

The plug cluster engine (PCE) configuration was the customer (SRI) designated baseline concept. The major reason for selecting the PCE concept for the Spaceplane application is that the PCE is much shorter than a conventional bell nozzle engine of the same thrust level and chamber pressure. The requirement for a minimum length vehicle was a major Spaceplane system requirement. Another advantage of the PCE, over a single engine installation, is its inherent reliability due to its multi-chamber configuration.

5.4.2 Propellant Tanks

5.4.2.1 Tank Configuration (Shape)

The propellant tank configurations, in order of preference, are:

- o Separate spherical tanks
- o Combination of spherical and conformal tanks
- o Separate conformal tanks
- o Integral bulkhead, conformal fuel/ox tank
- o Separate toroidal tanks

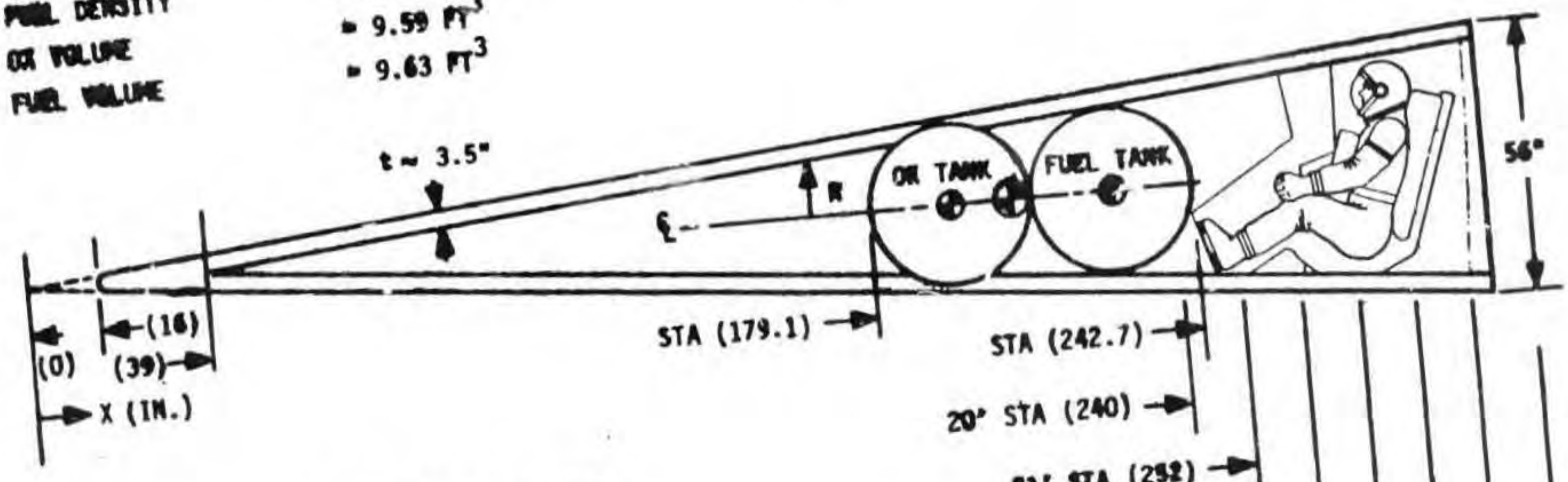
Multiple tanks offer potential benefits such as increased reliability and packaging advantages plus some disadvantages including increased weight.

Four of these tank configurations are shown in Figures 9 through 12. These figures show the volume, weight and shape of these configurations.

The use of a single conformal tank with elliptical bulkheads, with an internal elliptical bulkhead separating the propellants (see Figure 11),

MIXTURE RATIO = 1.65 (N_2O_4/MMH)
TOTAL PROPELLANT WEIGHT = 1400 LB
OX WEIGHT = 871.7 LB
FUEL WEIGHT = 528.3 LB
OX DENSITY = 90.9 LB/FT^3
FUEL DENSITY = 54.85 LB/FT^3
OX VOLUME = 9.59 FT^3
FUEL VOLUME = 9.63 FT^3

WEIGHT OF BOTH TANKS = 98.16 LB
TANK SAFETY FACTORS = 4.0
TANK MATERIAL DENSITY = 0.16 LB/IN^3 (Ti)
TANK MATERIAL σ = 130,000 PSI (Ti)
PROPELLANT CG, IN. = 207.0

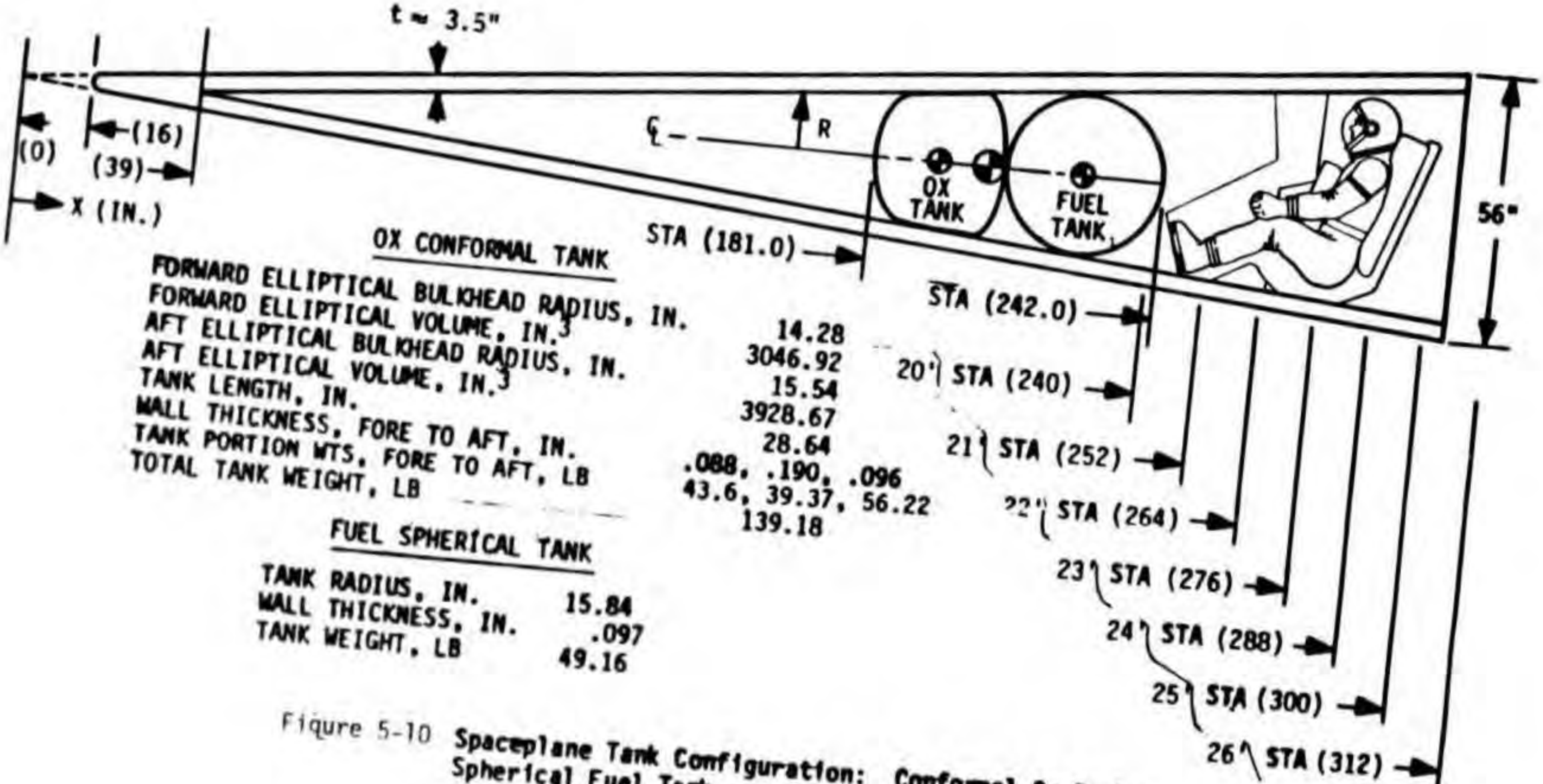


	OX TANK	FUEL TANK
TANK RADIUS, IN.	15.82	15.84
WALL THICKNESS, IN.	.097	.097
TANK WEIGHT, LB	49.0	49.16
PROPELLANT CG STATION, IN.	196.	226.8

Figure 5-9 Spaceplane Tank Configuration: 2 Spherical Tanks

MIXTURE RATIO	= 1.65 (N_2O_4/MMH)
TOTAL PROPELLANT WEIGHT	= 1400 LB
OX WEIGHT	= 871.7 LB
FUEL WEIGHT	= 528.3 LB
OX DENSITY	= 90.9 LB/FT^3
FUEL DENSITY	= 54.85 LB/FT^3
OX VOLUME	= 9.59 FT^3
FUEL VOLUME	= 9.63 FT^3

WEIGHT OF BOTH TANKS	= 188.34 LB
TANK SAFETY FACTORS	= 4.0
TANK MATERIAL DENSITY	= 0.16 LB/IN.^{3} (Ti)
TANK MATERIAL σ	= 130,000 PSI (Ti)
PROPELLANT CG, IN.	= 207.0



OX CONFORMAL TANK

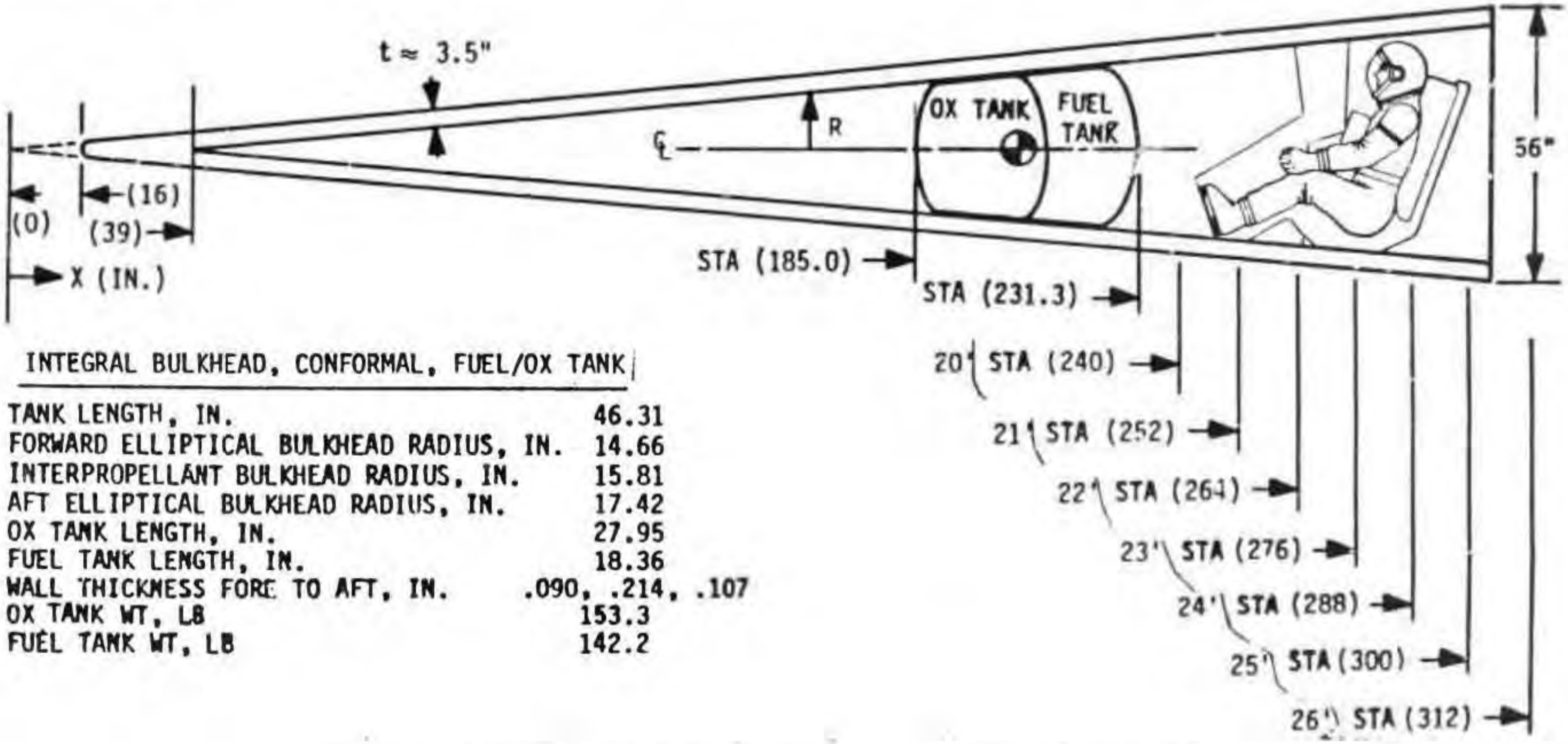
FORWARD ELLIPTICAL BULKHEAD RADIUS, IN.	14.28
FORWARD ELLIPTICAL VOLUME, IN.^{3}	3046.92
AFT ELLIPTICAL BULKHEAD RADIUS, IN.	15.54
AFT ELLIPTICAL VOLUME, IN.^{3}	3928.67
TANK LENGTH, IN.	28.64
WALL THICKNESS, FORE TO AFT, IN.	.088, .190, .096
TANK PORTION WTS, FORE TO AFT, LB	43.6, 39.37, 56.22
TOTAL TANK WEIGHT, LB	139.18

FUEL SPHERICAL TANK

TANK RADIUS, IN.	15.84
WALL THICKNESS, IN.	.097
TANK WEIGHT, LB	49.16

Figure 5-10 Spaceplane Tank Configuration: Conformal Ox Tank + Spherical Fuel Tank

MIXTURE RATIO	= 1.65 (N_2O_4/MMH)	WEIGHT OF BOTH TANKS	= 295.5 LB
TOTAL PROPELLANT WEIGHT	= 1400 LB	TANK SAFETY FACTORS	= 4.0
OX WEIGHT	= 871.7 LB	TANK MATERIAL DENSITY	= 0.16 LB/IN.^{3} (Ti)
FUEL WEIGHT	= 528.3 LB	TANK MATERIAL σ	= 130,000 PSI (Ti)
OX DENSITY	= 90.9 LB/FT^3	PROPELLANT CG, IN.	= 207.0
FUEL DENSITY	= 54.85 LB/FT^3		
OX VOLUME	= 9.59 FT^3		
FUEL VOLUME	= 9.63 FT^3		

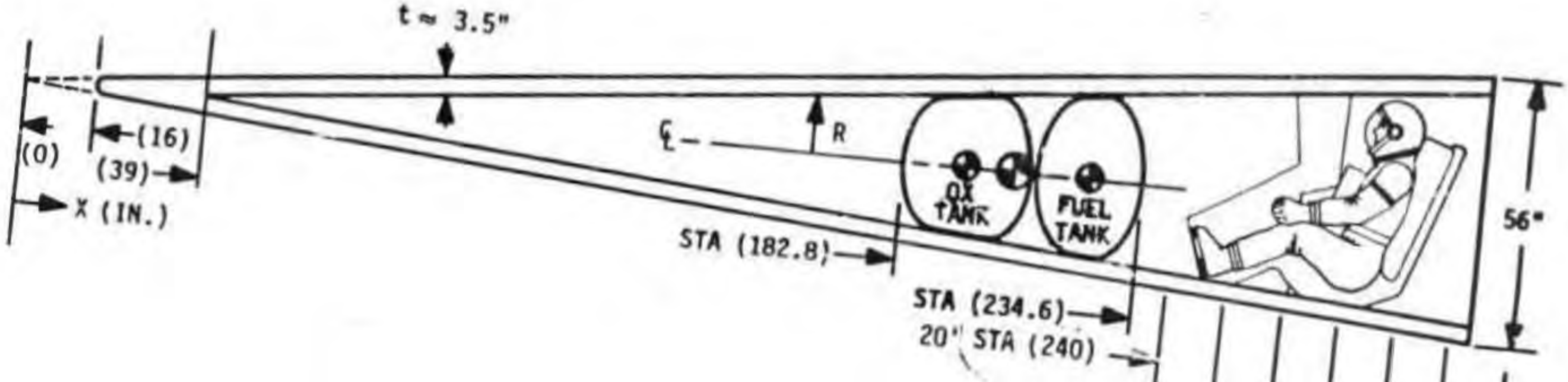


INTEGRAL BULKHEAD, CONFORMAL, FUEL/OX TANK	
TANK LENGTH, IN.	46.31
FORWARD ELLIPTICAL BULKHEAD RADIUS, IN.	14.66
INTERPROPELLANT BULKHEAD RADIUS, IN.	15.81
AFT ELLIPTICAL BULKHEAD RADIUS, IN.	17.42
OX TANK LENGTH, IN.	27.95
FUEL TANK LENGTH, IN.	18.36
WALL THICKNESS FORE TO AFT, IN.	.090, .214, .107
OX TANK WT, LB	153.3
FUEL TANK WT, LB	142.2

Figure 5-11 **Spaceplane Tank Configuration: Integral Bulkhead, Conformal Fuel/Ox Tank**

MIXTURE RATIO	= 1.65 (N_2O_4/MMH)
TOTAL PROPELLANT WEIGHT	= 1400 LB
OX WEIGHT	= 871.7 LB
FUEL WEIGHT	= 528.3 LB
OX DENSITY	= 90.9 LB/FT^3
FUEL DENSITY	= 54.85 LB/FT^3
OX VOLUME	= 9.59 FT^3
FUEL VOLUME	= 9.63 FT^3

WEIGHT OF BOTH TANKS	= 322.6 LBS
TANK SAFETY FACTORS	= 4.0
TANK MATERIAL DENSITY	= 0.16 LB/IN^3 (Ti)
TANK MATERIAL σ	= 130,000 PSI (Ti)
PROPELLANT CG, IN.	= 207.0



	OX TANK	FUEL TANK
FORWARD ELLIPTICAL BULKHEAD RADIUS, IN.	14.45	17.19
FORWARD ELLIPTICAL VOL, $IN.^3$	3159.3	5314.68
AFT ELLIPTICAL BULKHEAD RADIUS, IN.	15.66	17.71
AFT ELLIPTICAL VOL. $IN.^3$	4022.12	5820.9
TOTAL TANK LENGTH, IN.	28.23	23.21
WALL THICKNESS, FORWARD, CONICAL, AFT, IN.	.089, .192, .096	.106, .217, .109
TANK PORTION WTS, FORWARD, CONICAL, AFT, LB	45.21, 38.45, 57.55	76.05, 22.02, 83.29
TOTAL TANK WEIGHT	141.21	181.36
PROPELLANT CG STATION, IN.	197.25	222.90

Figure 5-12 Spaceplane Tank Configuration: 2 Conformal Tanks

although compact, is inherently unsafe in the event of an interpropellant bulkhead leak. With the hypergolic propellants involved, such a leak would likely be catastrophic to the Spaceplane. It could also destroy the Shuttle orbiter if the Spaceplane had not been deployed. Consequently, there is a question of safety relative to this tank configuration.

The use of an integral bulkhead fuel/ox tank has another serious problem which complicates the interpropellant leak danger. If H2O is used in an interpropellant barrier, then the barrier cavity pressure should be approximately equal to the propellant tank pressure to keep the barrier wall thicknesses small. Maintaining the barrier pressure nearly constant if water is depleted represents an added complexity.

Another consideration regarding the integral bulkhead fuel/ox tank is the fact that although an elliptical barrier is preferred over a flat barrier, to save weight, an elliptical barrier can, under a relatively small pressure differential, buckle. Again, a heavy barrier wall may be required to eliminate this possibility.

A promising solution to these problems is the use of radial ribs between the two elliptical, integral bulkhead surfaces. This would result in a modest weight penalty but would also eliminate the potential bulking problem. The cavity would still be available to hold water if required and could even sustain leaks in both propellant tanks if vented through different segments enclosed by the radial ribs.

Based on discussions with SNLA and H/S, tank shapes will ultimately be determined by the vehicle c.g. location requirement (i.e., originally 66% of the vehicle length, later 69% of the vehicle length). At the original vehicle c.g. location (i.e. 66% vehicle length), only conformal tanks would fit.

The selection of two spherical tanks (Figure 9) as the recommended configuration was made on the basis of discussions held at the January TI/TD meeting with SRI, H/S and Lincom. The important reasons for this recommendation are listed below.

- Spherical tanks are the lightest weight.
- The internal vehicle volume/length penalty of two spherical tanks compared to the conformal tank configuration (see Figure 11) is not considered excessive.
- Spherical tanks are ideal for placement of a toroidal He bottle located between them.
- The use of PAAB-1 as the main fuel, instead of MMH will, because of the required mixture ratio, result in a smaller ox tank and larger fuel tank. This would result in a more efficient use of the tapering conical envelope of the Spaceplane interior volume.

- Since the desired vehicle c.g.location moved aft (from Station 207 shown in Figures 9 through 12) to Station 221.6 (69% vehicle length), adequate spacing between the oxidizer tank wall and vehicle structure was provided for placement of propellant and other lines.

5.4.2.2 Tank Materials

The four recommended tank material options, listed in order of preference, are:

- Titanium (Ti 6Al-4V alloy)
- Aluminum (Al 2219)
- Stainless Steel (Cryoformed 301 SS)
- Nickel Alloy (Inconel 718)

This selection was made on the basis of these primary considerations:

- Cost
- Technology
- Propellant compatibility
- Weight

The most expensive of these materials, titanium, also has the most extensive use as an actual storable propellant tank material. The only major disadvantage of titanium is that, although fully compatible with green (NO inhibited) N_2O_4, it is not compatible with red (uninhibited) N_2O_4.

Aluminum is much cheaper and relatively easy to fabricate. However, if wall thicknesses are large (over 0.5 inches) reliable welding can become difficult.

Cryoformed 301 stainless steel is a viable option whose only real disadvantage is the question of fabrication of tank shapes or sizes required by the Spaceplane.

The nickel alloy, Inconel 718, provides a fourth, but not recommended, tank material option. Although compatible with N2O4, MMH and PAAB-1, Inconel 718 tanks would require a lengthy and intricate heat treatment. This consideration alone also makes Inconel 718 tanks expensive.

Also noted here, in order of preference, are the recommended tank materials for fabrication of the storage bottles.

- Titanium (Ti 6Al-4V alloy)
- Stainless Steel (Cryoformed 301 SS)

The basis for this selection was the same as outlined for the main propellant tanks except that propellant compatibility was not a consideration.

5.4.3 Propellant Control Subsystem

The initial propellant control (PC) candidate subsystems included:

- o Non-metallic bladders
- o Separate start tanks
- o Hoppers
- o Screen assemblies (surface tension barrier)
- o Tank pistons

Before defining the selection criteria, several important considerations were identified:

- o Butyl rubber bladders are compatible with MMH but not with N_2O_4, and they are reusable. It should also be pointed out that elastomeric bladders means elastomeric bladders or diaphragms. The basic difference between a bladder and diaphragm for conical tanks which are typical of the Spaceplane is illustrated in Figure 13. The choice of a bladder or diaphragm is a design detail that will be resolved in a later phase of the Spaceplane program.

- o Based on a literature survey and industrial contacts, Carboxy Nitroso Rubber (CNR) bladders appear to be a viable, but probably expensive, option for use in the N_2O_4 propellant tank. Two formal literature searches were then initiated to verify (or negate) this initial assumption. The two formal literature searches conducted produced the following documents:

  1. NASA Literature Search, "Elastomers that are Compatible with Nitrogen Tetroxide." 2 Volumes ((1) Open and (2) Limited Distribution references), NASA Search No. 47004, dated 16 November 1981.

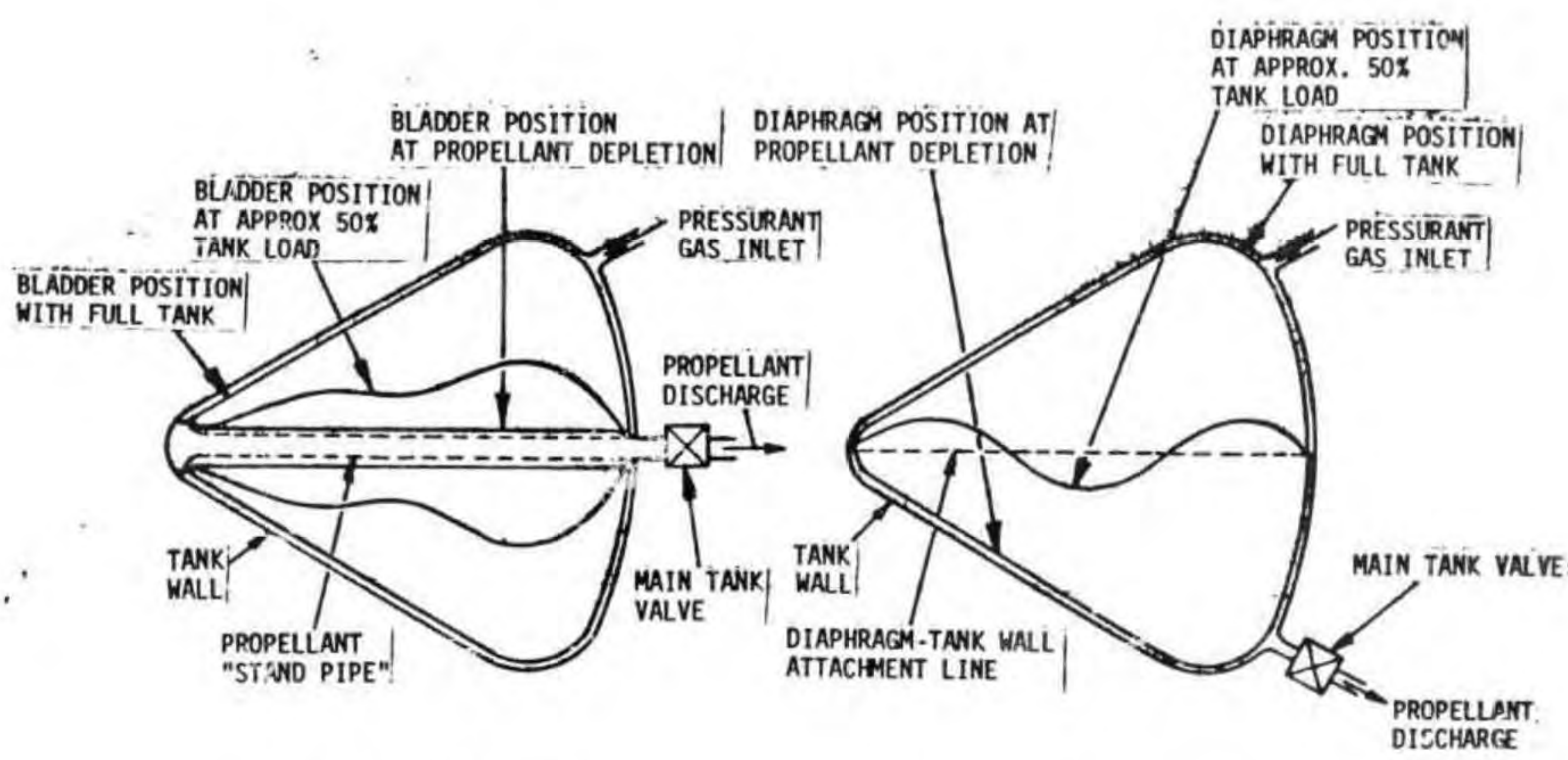
  2. Defense Technical Information Center (DTIC), "Elastomers." 1 Volume, DTIC Search Control No. 008546, dated 13 November 1981.

- o Fixed metallic propellant control systems (such as screens, hoppers and start tanks) are preferred over bladders. Figure 14 illustrates the basic operation of start tanks and hoppers with some of their relative advantages and disadvantages.

- o Different metallic bladders are compatible with both propellants, but they are not reusable.

- o A preliminary analysis, based on available data, indicated that screens are suitable for zero g engine starts, but not the 5 g loads anticipated for the Spaceplane.

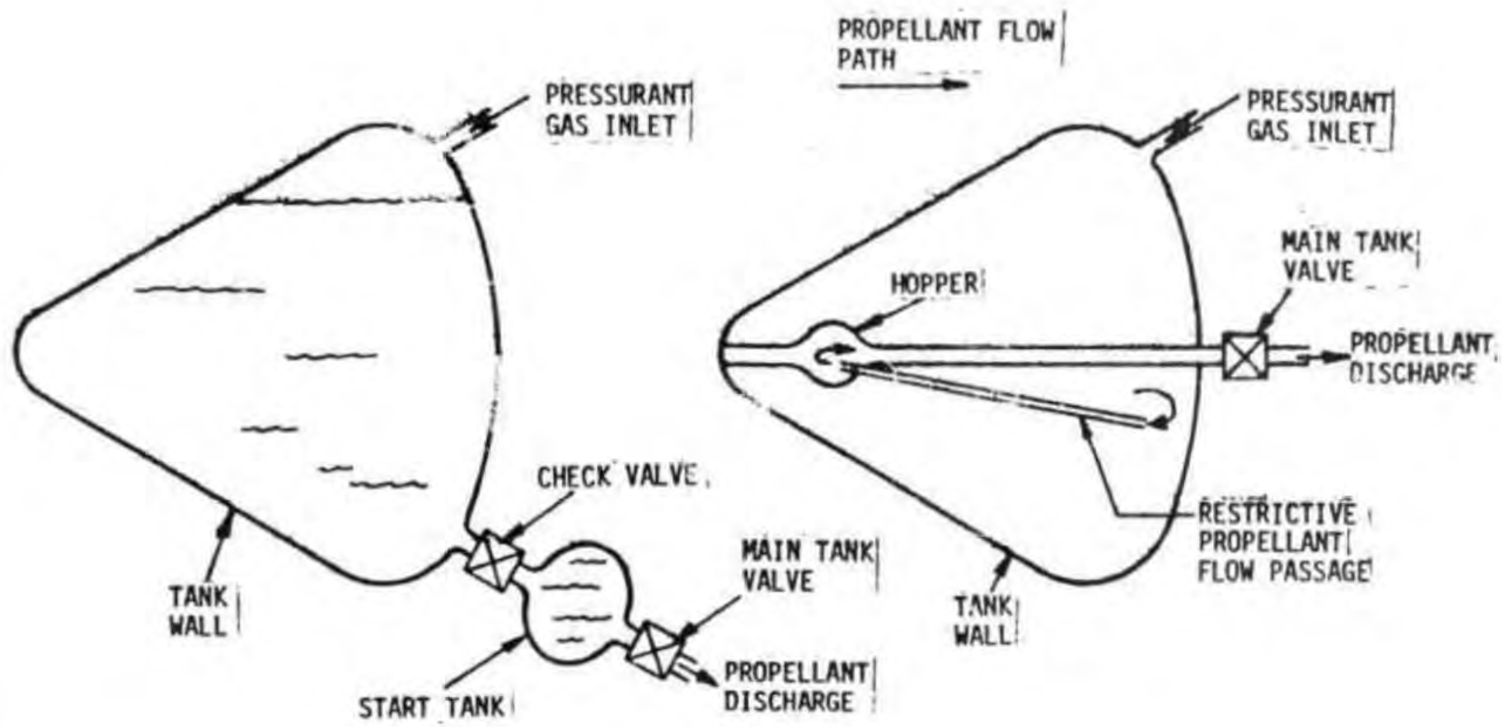
- o Screen assemblies are potentially heavy.



a) ELASTOMERIC BLADDER

b) ELASTOMERIC DIAPHRAGM

Figure 5-13 **Elastomeric Bladder and Diaphragm Propellant Control Subsystem**



a) START TANK

FEATURES: EXTERNAL COMPONENT (START TANK)
SIMPLE
EASY TO MAINTAIN, REFURBISH, ETC.

b) HOPPER

FEATURES: POTENTIALLY COMPLEX TANK-INTERNAL COMPONENTS
HOPPER CAN PROVIDE TANK STRENGTHENING STRUCTURE

Figure 5-14 Start Tank and Hopper Propellant Control Subsystems

- o Tank pistons are considered the least desirable option as they are heavy and require cylindrical tanks which do not package efficiently in the Spaceplane vehicle envelope.

- o Any propellant control system must be compatible with the tank pressurization system.

Eventually, screen assemblies (surface tension devices) were eliminated as a candidate PC subsystem based on these considerations:

- o A fine mesh metallic screen can have a catalytic effect on N_2O_4 over long term (over 2 months) storage.

- o Screen assemblies for the Spaceplane would be the most complex of any PC subsystem because of the propellant orientation during the synergistic plane change maneuvers. In addition to the screen channels, one or more internal tank barriers would be required. Internal tank barriers are also heavy.

- o Screen assemblies must be clean, otherwise contamination can result in corrosion of the screen mesh.

- o Fine mesh screen shows susceptability to direct oxidation by N_2O_4.

On the other hand, elastomeric bladders were retained as a viable PC subsystem for both propellant tanks. This is based on the results of the DTIC literature search. Basically two important developments were noted in this literature search:

- o TRW reported, in 1971, on an improved CNR (Carboxy Nitroso Rubber) material that "... has mechanical properties which compare favorably to previous CNR compounds and, most significantly, has a thousandfold decrease in the nitrogen tetroxide permeability rate." (Ref. 1)

- o More recently (1978-80), TRW has reported on "AF-E-124T, a perflouro polymer manufactured by DuPont and modified by TRW, ...was evaluated for nitrogen tetroxide expulsion bladders, ...these evaluations demonstrated the feasibility of using AF-E-124T expulsion bladders for nitrogen tetroxide service." (Ref. 2)

Also, ALRC became aware of another DuPont elastomer, Kalrez R , which was advertised as being compatible with N_2O_4. Unfortunately, this material appears to be suitable only for use in seals or gaskets rather than bladders or diaphragms. This observation is based on the known historical aerospace uses of Kalrez R . The optimum bladder material still needs to be determined.

Based upon these preliminary considerations, the recommended propellant control subsystems for separate and conformal tank configurations were:

1. Separate propellant tanks

- o Start tanks for both propellant tanks
- o Combination of start tank and bladder
- o Elastomeric bladders in both tanks
- o Combination of start tank and hopper

2. Integral bulkhead, conformal fuel/ox propellant tank

- o Start tanks for both propellant tanks
- o Ox tank elastomeric bladder and fuel tank start tank
- o Combination hopper and start tank (one to each propellant tank)
- o Ox tank elastomeric bladder and fuel tank hopper

This preliminary selection was specifically made on the basis of these weighted criteria:

- o Complexity
- o Reliability
- o Material (propellant) compatibility
- o Weight
- o Expulsion efficiency
- o Size
- o Tank configuration

Subsequent evaluation showed that, with the high g maneuvers anticipated to be performed by the Spaceplane, both start tanks and hoppers would be inadequate to ensure liquid propellants (instead of vapors) to the PCE. Based on preliminary calculations, elastomeric diaphragms appear to have excellant propellant control capability even under high adverse g loads (f10 g's). The requirement of Spaceplane retro thrust capability would necessitate at least two start tanks or hoppers per propellant tank.

It also appears that with the relatively small mass of the Spaceplane, that propellant sloshing in orbit may require extensive use of the RCS to maintain the required vehicle attitude. This in turn will require more propellant and hence bigger tanks and ultimately a reduced Spaceplane . If propellant slosh is to be suppressed, this will eliminate both hoppers and start tanks as propellant control subsystem options. The next choice would be elastomeric diaphragms in the spherical propellant tanks. Other, heavier options would be bellows or piston tanks. Both are heavy, have low expulsion efficiency and are not usable in spherical tanks.

Based on these considerations, the only remaining propellant control subsystem for any tankage configuration was elastomeric bladders or diaphragms. This resulted in the requirement for a long life, N_2O_4 compatible elastomeric bladder, or diaphragm, material although not currently state-of-the-art.

The use of elastomeric materials in either tank also eliminates any hot gas pressurization systems which were evaluated in the pressurization subsystem comparison described next.

5.4.4 Tank Pressurization

The tank pressurization candidate subsystems as first defined included:

- Autogenous pressurization using heat exchanger(s) and electrically-driven pump(s)
- Cold gas pressurization using He, N_2, etc.
- Warm gas pressurization using He, N_2, etc., and heat exchanger(s)

Some of the first observations made included the following:

- He and N_2 are both compatible with MMH and N_2O_4; however, autogenous pressurization is superior from a performance standpoint. The basic tradeoff to be made is the simpler, but heavier pressurant gas system using He or N_2 versus the lighter but more complex autogenous system. A further consideration is the propellant control method utilized.
- A cold gas (He or N_2) pressurization system evaluation resulted in a He requirement of approximately 10 to 15 lbs in addition to a pressurant tank weight of 75 to 200 lbs (depending on the safety factor assumed). This is a simple but heavy system.
- Warm gas (He or N_2) would be a relatively light system, especially compared to the cold gas system. The possibility of heating the pressurant gas with heat from the main propulsion system or from the cockpit was also considered. It should be emphasized that the use of a warm pressurant gas in place of a cold pressurant gas can have a significant impact on the total pressurization system (pressurant and tank) weight.
- The autogenous pressurization system (requiring a pump assembly) is the third and least desirable candidate pressurization system. Its inherent complexity makes it the least reliable of the three candidate pressurization systems.

Subsequent to these initial conclusions, three additional hot gas (i.e. gas generator) pressurization subsystem concepts were identified. These are described briefly below.

The first was an autogenous pressurization system resulting from a 1971 AFRPL-sponsored program conducted by Rocket Research Corporation (RRC). In this program, a motor/pump-fed monopropellant hydrazine bootstrap pressurization system was designed, built, and tested. The final report for this program is Reference 3. Some of the major features of this program and the resultant system are described below.

- o The major objective of the program was to demonstrate the capability of a pump-fed monopropellant hydrazine bootstrap pressurization subsystem to meet the performance and environmental requirements of advanced post-boost propulsion systems. A schematic of the subsystem indicating its relationship to an overall propulsion system is shown in Figure 15. A photograph of this system is shown in Figure 16. Table II lists the system requirements as specified in the stu' contract.

- o The total weight of the assembled subsystem was 8.61 lbM.

- o The system is also compatible with the N_2O_4/PAAB-1 propellant combination. In addition to serving as a bipropellant fuel, PAAB-1 will also serve as the monopropellant used to drive the system reactor (see Figure 15). This motor/pump system will receive serious consideration when the candidate propellant pressurization subsystems are ranked in order of preference.

The second concept was a monopropellant gas generator using a He bottle pressure source. This subsystem is illustrated schematically in Figure 17. The basic advantage of this concept over a cold He pressurization subsystem is that the pressurant gas is actually stored as a liquid (i.e., the monopropellant or PAAB-1 if used as the Spaceplane fuel) instead of a gas. The large difference in density between the liquid and gas results in a much smaller volume, and hence weight, required for the liquid as compared to the gas. This reduction in size and weight comes at the cost of increased complexity (i.e., gas generatory, valves, warm gas lines, etc.).

The third concept was a monopropellant gas generator using air intensifier (bootstrap) tank pressure source. This concept is illustrated schematically in Figure 18. This bootstrap pressurization subsystem is potentially the lightest of all pressurization subsystems because it stores all of the required pressurant as a liquid, but does not require a separate He bottle for monopropellant pressurization as does the first subsystem. The subsystem as shown in Figure 18 is for a one-use application, but it can be configured as a reusable, demand subsystem. This concept requires the use of a "bootstrap" or "intensifier" tank which, depending on the volume of monopropellant to be expelled, can be large and heavy. Additionally, this system requires a separate monopropellant (bootstrap) tank and thus would not particularly benefit from the use of PAAB-1 as the Spaceplane fuel.

Both of these subsystems were included in the PS selection process. The results of this selection showed that the preferred pressurization subsystems in order, are:

1. PS redundancy not required

- o Cold He
- o Warmed He pressurization
- o Monopropellant gas generator using an electric motor driven pump
- o Monopropellant gas generator using He pressurization

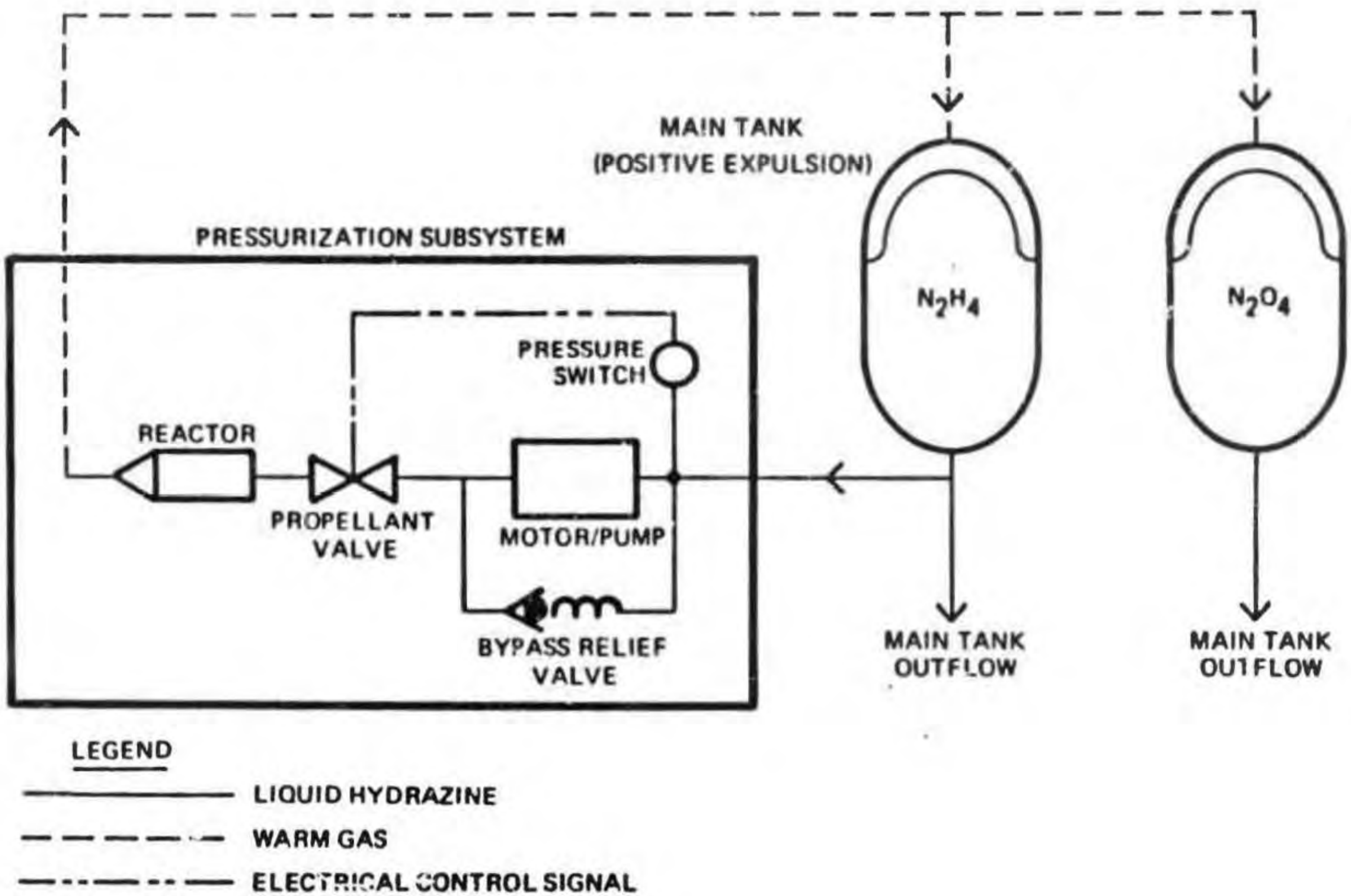


Figure 5-15 Pump-Fed Bootstrap Pressurization System Schematic

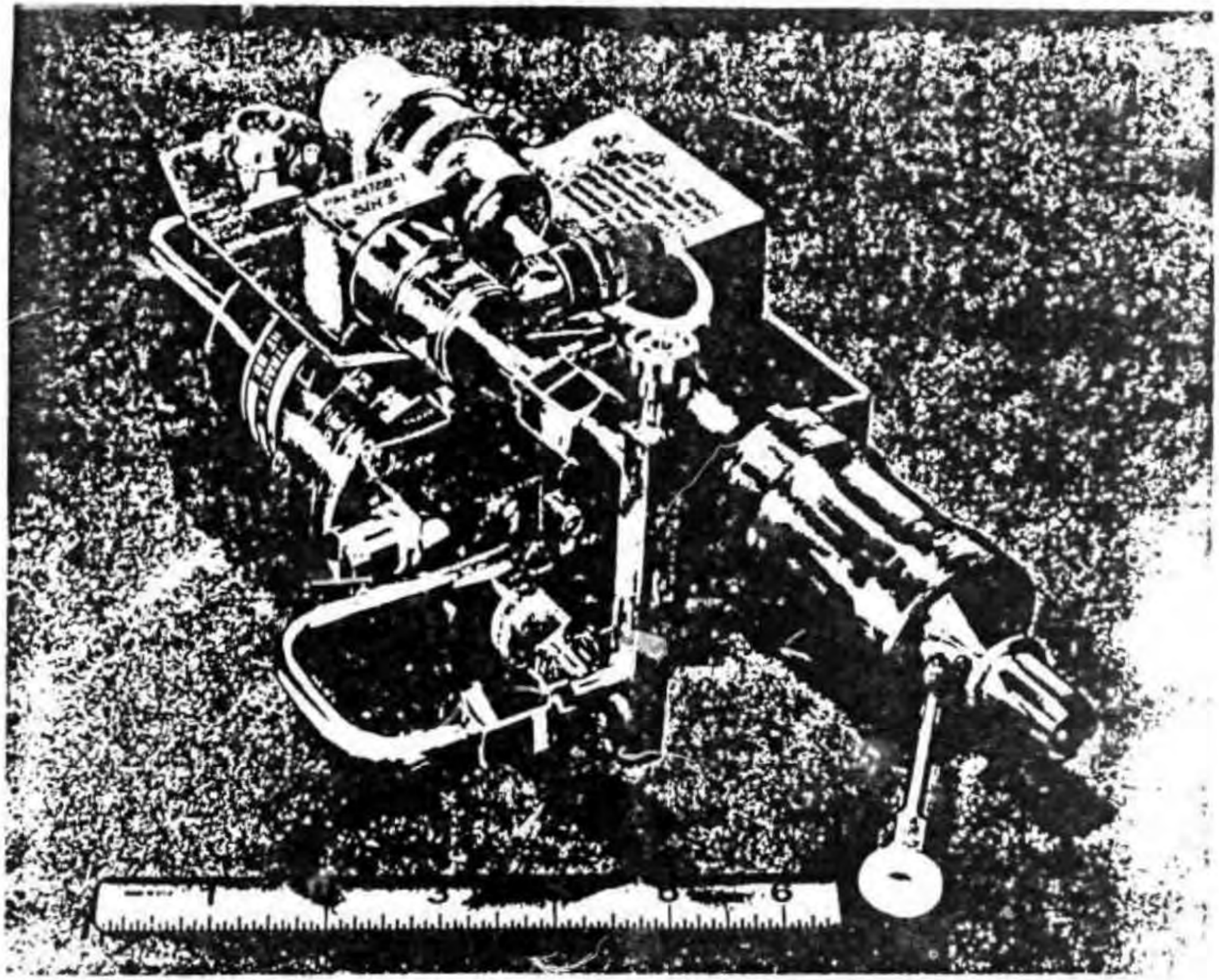


Figure 5-16 Pump-Fed Monopropellant Hydrazine Bootstrap Pressurization Subsystem

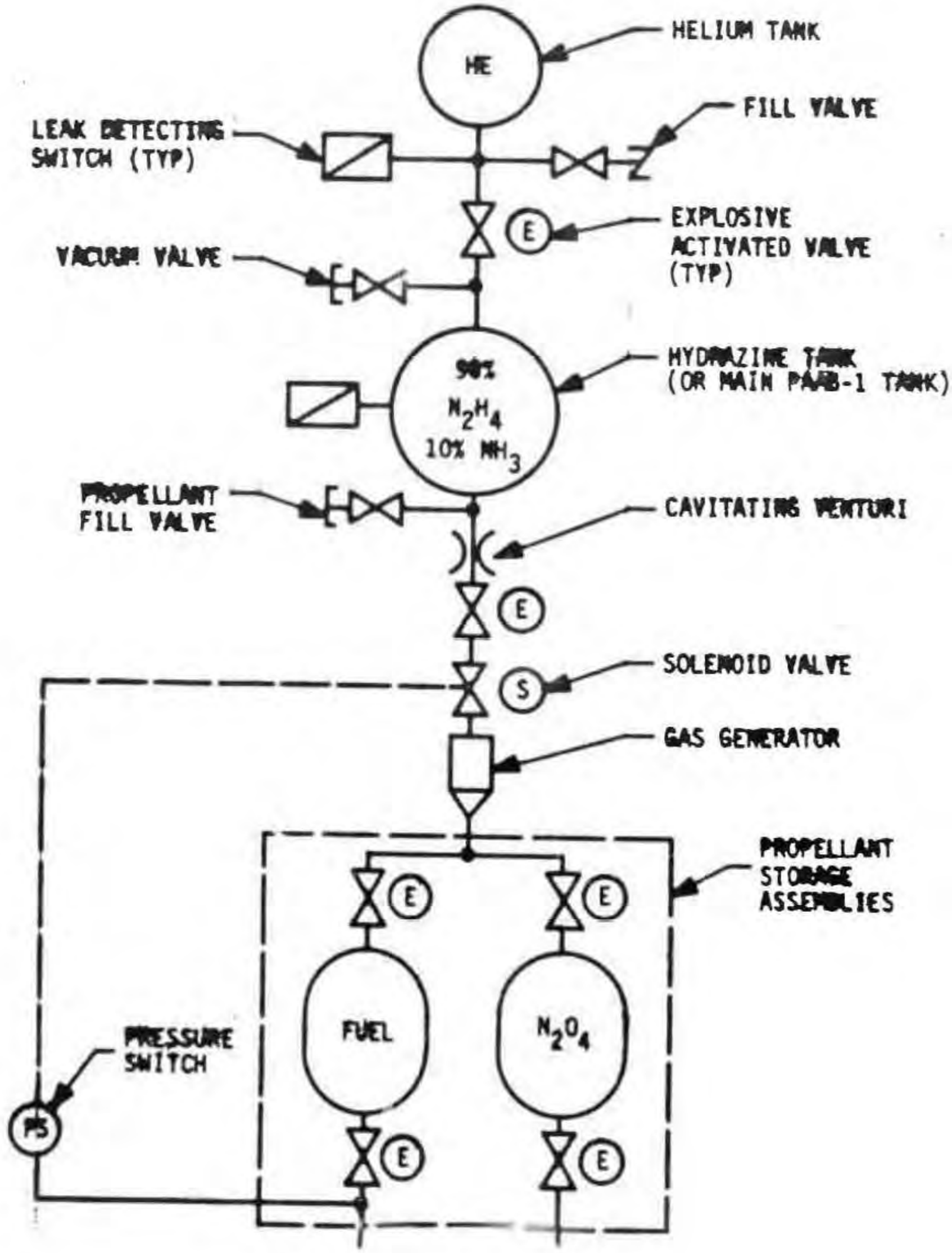


Figure 5-17 Monopropellant Gas Generator Using He Bottle

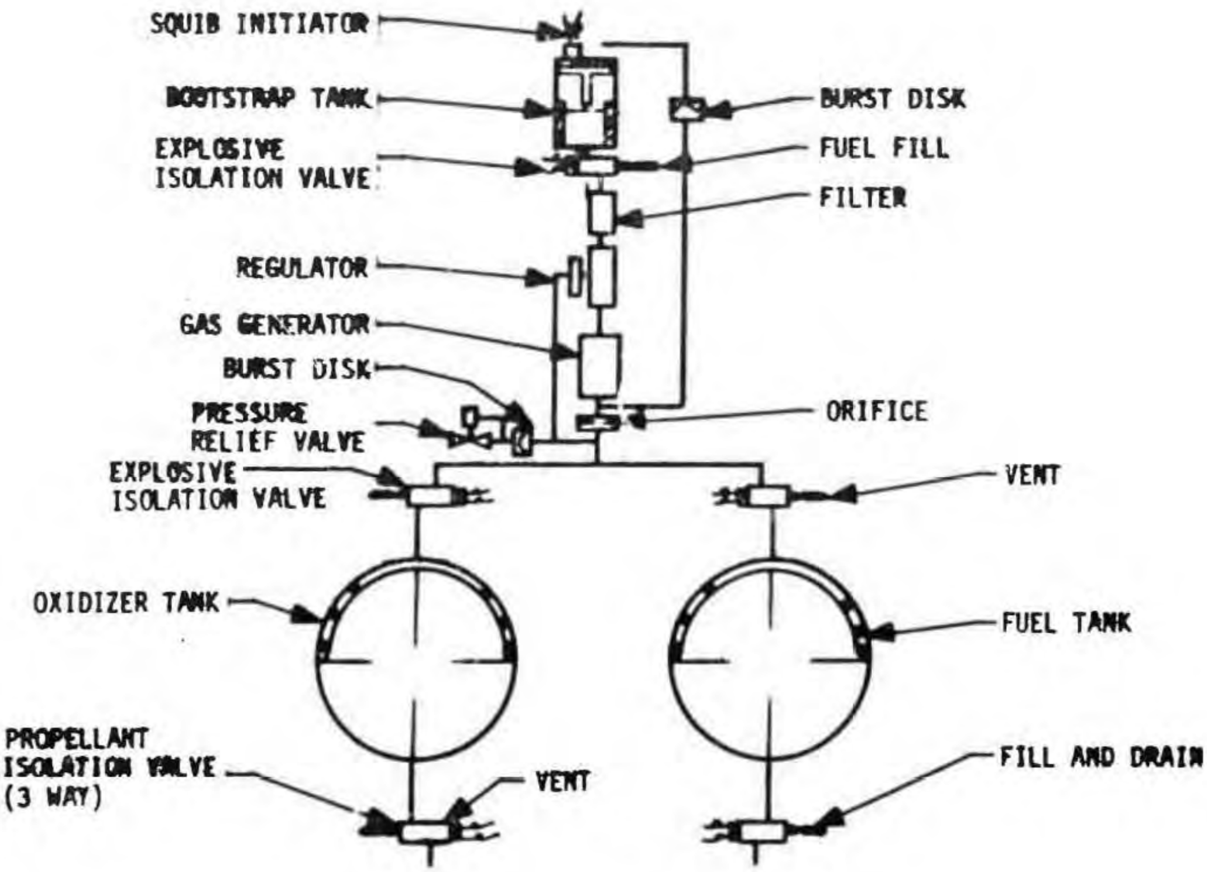


Figure 5-18 Monopropellant Gas Generator Using Intensifier (Bootstrap) Tank

TABLE II

CANDIDATE PRESSURIZATION SUBSYSTEM REQUIREMENTS

Total propellant tank volume to be pressurized	30 ft^3
Propellants	N_2O_4 Amine fuel
Gas generator output pressure	500 psi
Gas generator output flowrate	0 - 0.05 lb/sec
Maximum gas temperature	1,500°F
Gas generator propellant	Hydrazine (N_2H_4)
Mission time	900 seconds maximum
Tank pressure control band	±2% maximum, ±1% desired
Storage temperature	80°F ± 40°F
Operating temperature	70°F ± 30°F
Storage life goal	10 years
Ambient pressure	0 - 15 psia
Acceleration	23 g's maximum 2 g's operating
Shock	27-g peak. 18 msec, half time

2. Redundant PS

- o 2 cold He bottles
- o 2 warmed He bottles
- o Monopropellant gas generator with a motor/pump for fuel tank pressurization with direct, warmed He pressurization of ox tank
- o Monopropellant gas generator with an He bottle for fuel tank pressurization tank with direct, warmed He pressurization of ox tank

A redundant PS for the Spaceplane is one in which either of two independent PS's (one for each propellant tank) can provide at least partial pressurization for both tanks in the event of a failure of one of the two independent PS's.

This selection was based on these weighted criteria:

- o Complexity
- o Weight
- o Presence of combustion devices (gas generator)
- o Capability of providing redundancy
- o Hot or warm gas circuits
- o Size
- o Reliability

The results of this screening can be combined with the propellant control (PC) candidate subsystems previously described, to formulate several compatible combinations. All of the resulting pressurization/propellant control subsystems provide PS redundancy.

The propellant control/pressurization subsystems compatibility criteria were:

- o No hot, fuel-rich gas in direct contact with the oxidizer or any elastomeric bladder in eighter the primary mode or backup mode.

- o No elastomeric bladders or diaphragms in fuel portion of an integral bulkhead, fuel/ox tank.

1. For use with an integral bulkhead, fuel/ox propellant tank

- o Start tanks in both propellant tanks
- o Ox tank elastomeric bladder and fuel tank start tank

Both of these PC subsystems are compatible with only the first two of the redundant pressurization subsystems listed above.

2. For use with separate propellant tanks

- o Start tanks in both propellant tanks
- o Combination of elastomeric bladder and start tank
- o Elastomeric bladders in both propellant tanks

All three of these PC subsystems are compatible with only the first two redundant pressurization subsystems listed above.

Finally, as noted in the discussion of propellant control subsystems, the selection of elastomeric diaphrams (or bladders) eliminates all hot gas PS concepts. For this reason, only warm or cold He pressurization, for either redundant or non-redundant pressurization subsystems, is recommended for application in the Spaceplane onboard propulsion system.

A more rigorous weight and envelope comparison of cold and warm He pressurization subsystems is described below.

Figures 19 through 21 are plots of pressurization subsystem, (PS) weight versus He temperature in the propellant tanks at propellant depletion. This temperature value is really an indicator of the relative use (or absence) of heating the He before it enters the propellant tanks. A value of approximately $233^{o}R$ (or $-227^{o}F$) on these plots corresponds to an unheated He PS. A temperature value of $530^{o}F$ ($70^{o}F$) is representative of a PS in which the He is raised to room temperature ($70^{o}F$) before it enters the propellant tanks. The three figures correspond to the storage pressures of 2000, 3000 and 4000 psia, respectively. As the plots show, the PS weight decreases with increasing He storage pressure. They also show the major PS weight reduction possible with the use of He heating.

It should also be noted that the redundant systems identified are completely redundant. That is, the failure of either He storage bottle will not degrade the mission capability of the Spaceplane. On the other hand, the non-redundant PS's are those in which the loss of a single bottle will result in the complete loss of the PS. A two-bottle, non-redundant PS (such as a torus and sphere) means that the loss of both bottles is required for a complete PS loss.

This last observation indicates the potential benefit of a multiple bottle PS. Theoretically, several spherical bottles are equal in weight to one large spherical bottle of the same total volume. In practice, there would be some weight penalty to the multiple bottle system due to additional lines, mounts, and pressure regulators. The advantage of the multiple bottle system is that the loss of one or more bottles will not necessarily result in a mission abort since, depending on how many bottles are used, the PS will have lost only a portion of its total capability. The use of multiple spherical bottles also offers attractive packaging flexibility.

The optimum configuration of a multiple-bottle PS (i.e., number, size, and placement of the several He bottles) is a Spaceplane subsystem integration problem which H/S addressed. From a propulsion system performance standpoint, number, size, and placement of He bottles is relatively unimportant. The propulsion system reliability, on the other hand, is quite dependent on the number of He bottles in the PS.

Figures 22, 23 and 24 are plots of important dimensions associated with the PS weights plotted in Figures 19, 20 and 21, respectively. Again, each figure corresponds to He storage pressures of 2000, 3000 and 4000 psia, respectively. In all cases the torus tank not only has a larger minimum diameter, but is also heavier than an equal volume spherical tank. This is

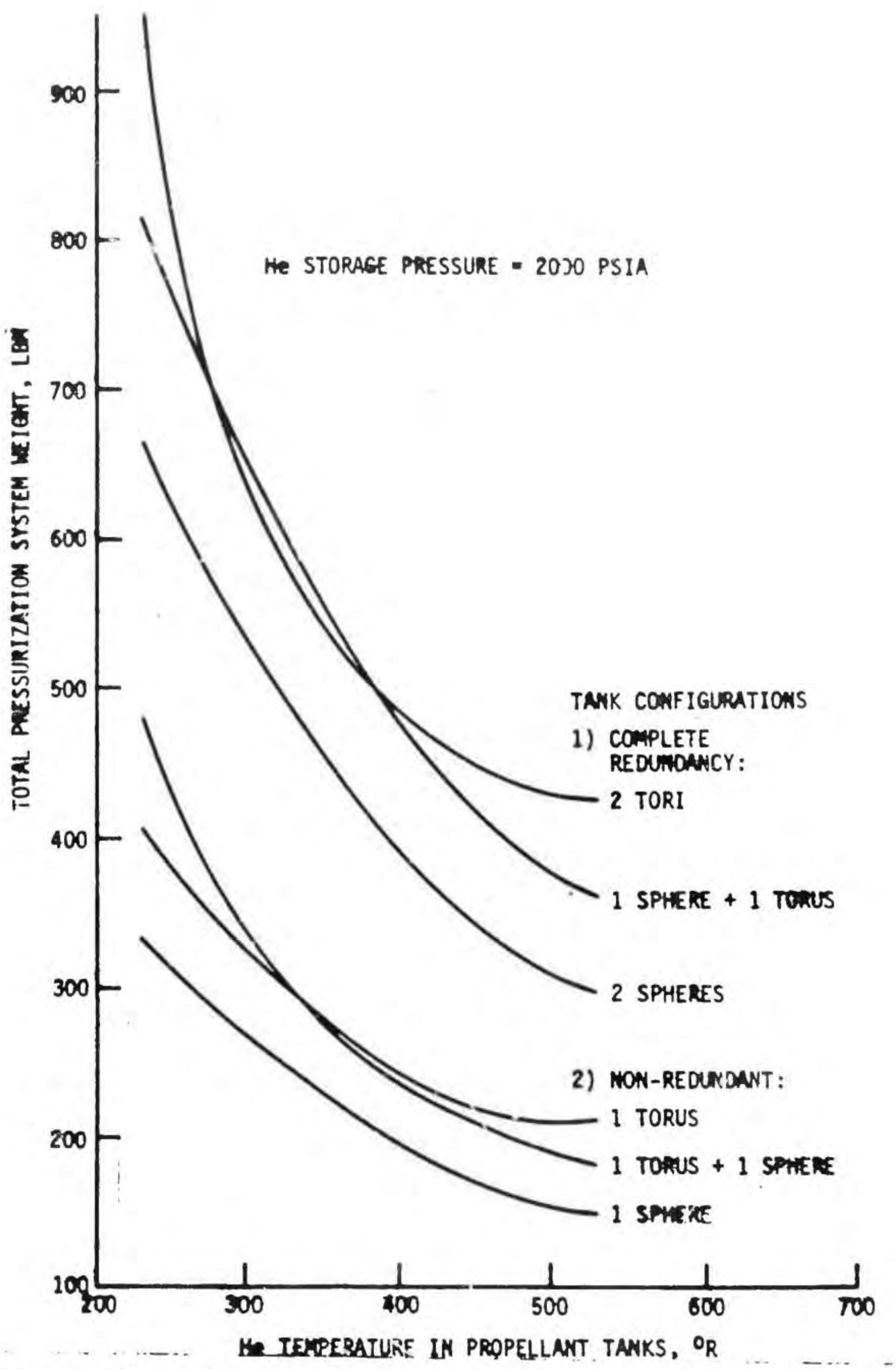


Figure 5-19 Total Pressurization Subsystem Weight vs He Temperature in Propellant Tanks (He Storage Pressure = 2000 psia)

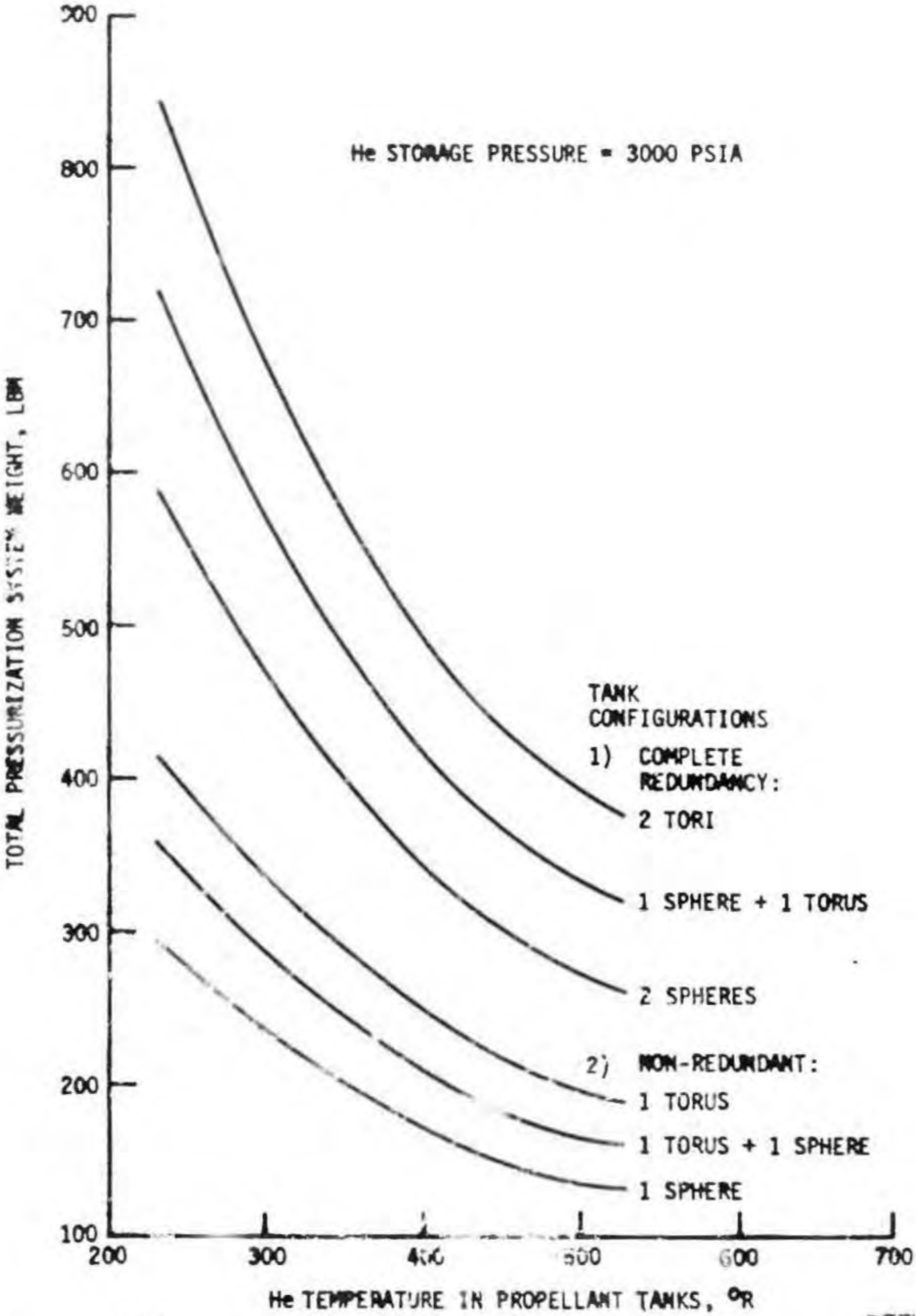


Figure 5-20 Total Pressurization Subsystem Weight vs He Temperature in Propellant Tanks (He Storage Pressure = 3000 psia)

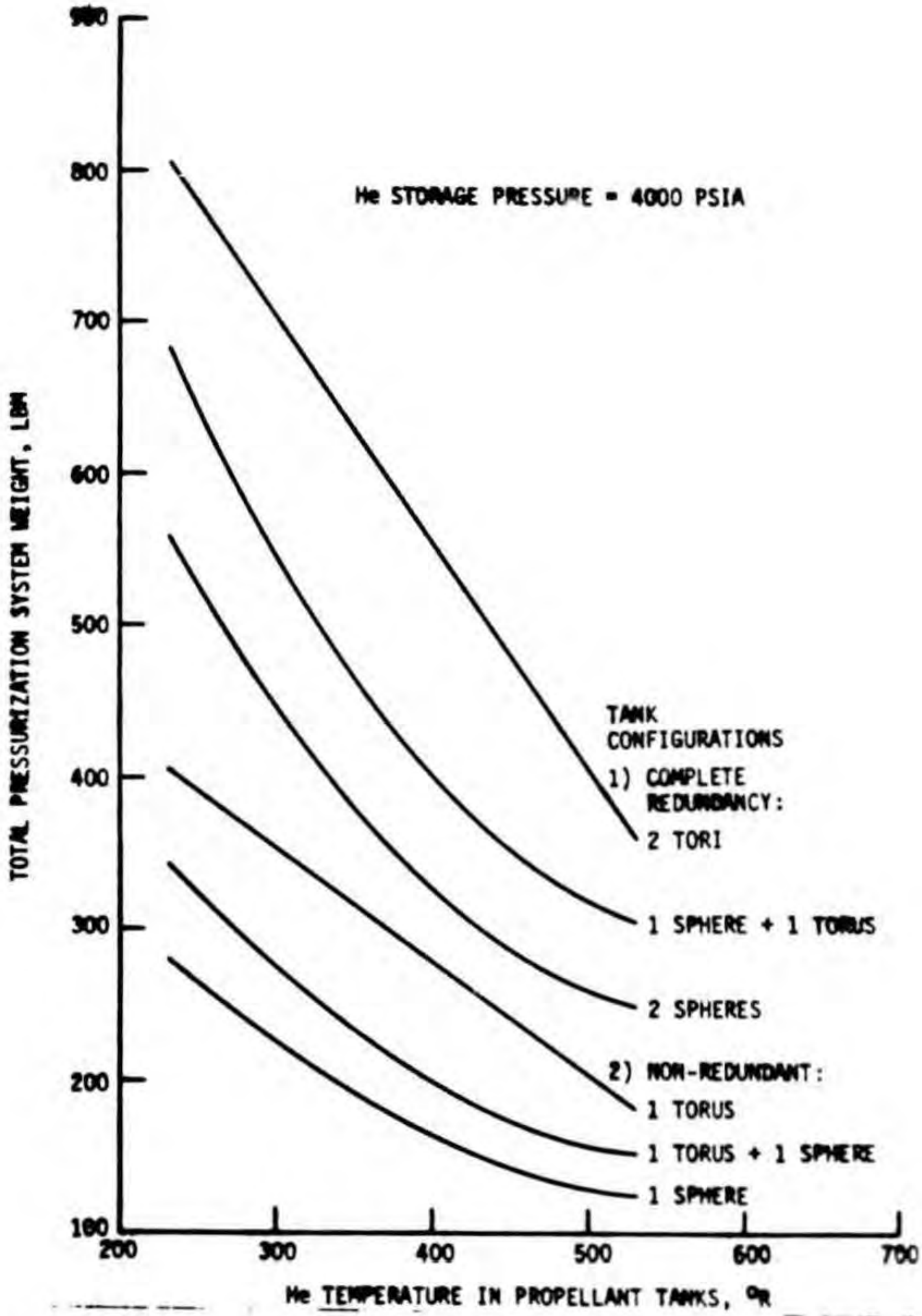


Figure 5-21 Total Pressurization Subsystem Weight vs He Temperature in Propellant Tanks (He Storage Pressure = 4000 psia)

the same conclusion reached from the comparison of toroidal and spherical propellant tanks.

On the basis of the codd vs. warm He pressurization subsystem weights for a 4000 psia He storage pressure system as shown in Figure 21, the warm He pressurization subsystem was recommended as the baseline Spaceplane onboard propulsion propellant pressurization concept.

Initially it was assumed that the PCE itself would provide the heat source, via plug mounted heat exchanges, to warm the He prior to propellant tank injection. There are at least two other methods of warming the He that were also considered:

- Using the low power level (approx. 1 HP), continuously running Spaceplane APU(s) as the He heat source
- Heating the He electrically, using the Spaceplane onboard electrical power system (APU's and batteries)

These two methods are discussed below:

APU's as He Heat Source

First of all, there was some question as to whether or not a hydrazine powered APU could run continuously for 24 hours. It is possible however, that such a low power device (approximately 1 HP for the Spaceplace) may be able to employ an energy conversion device other than a turbine, which is a life limiting element of a conventional APU. An APU manufacturer should be consulted to resolve this issue.

Electrical He Heating

A heat transfer analysis was completed to evaluate the possibility of electrically (resistance) heating the He pressurant gas instead of using PCE mounted heat exchanges. Some of the advantages and disadvantages of the electrical heating as opposed to PCE heating are listed below:

- Electrical (Resistance He Heater)

  Advantages:

  - Flexible packaging (easy to locate He line heater inside of Spaceplane).
  - Lightweight (basically wire and possibly insulation).

  Disadvantage:

  - Results in requirement for additional Spaceplane electrical power (increased battery weight and/or additional APU monopropellant required).

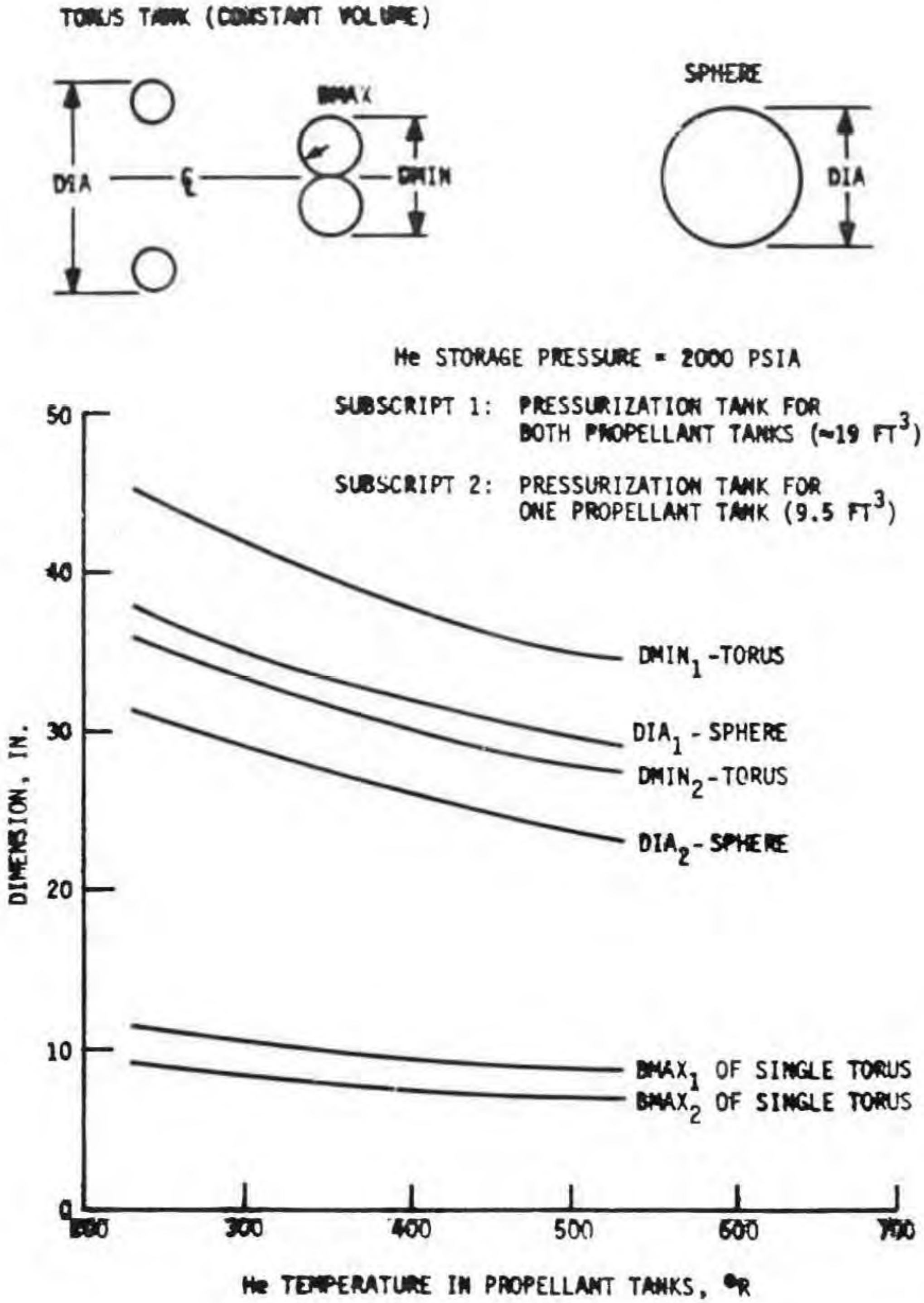


Figure 5-22 Pressurization Subsystem Dimensions vs He Temperature in Propellant Tanks (He Storage Pressure = 2000 psia)

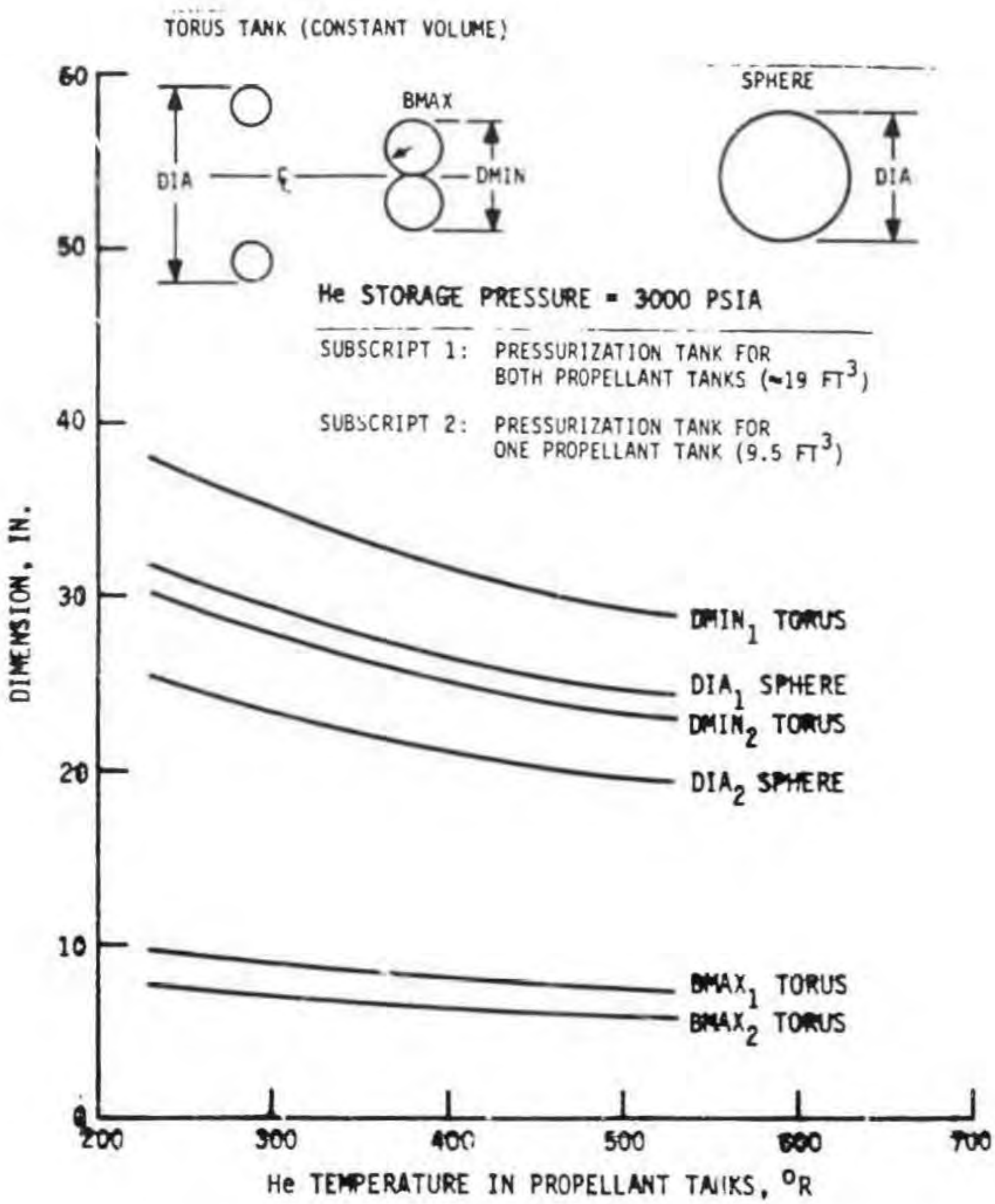


Figure 5-23 Pressurization Subsystem Dimensions vs He Temperature in Propellant Tanks (He Storage Pressure = 3000 psia)

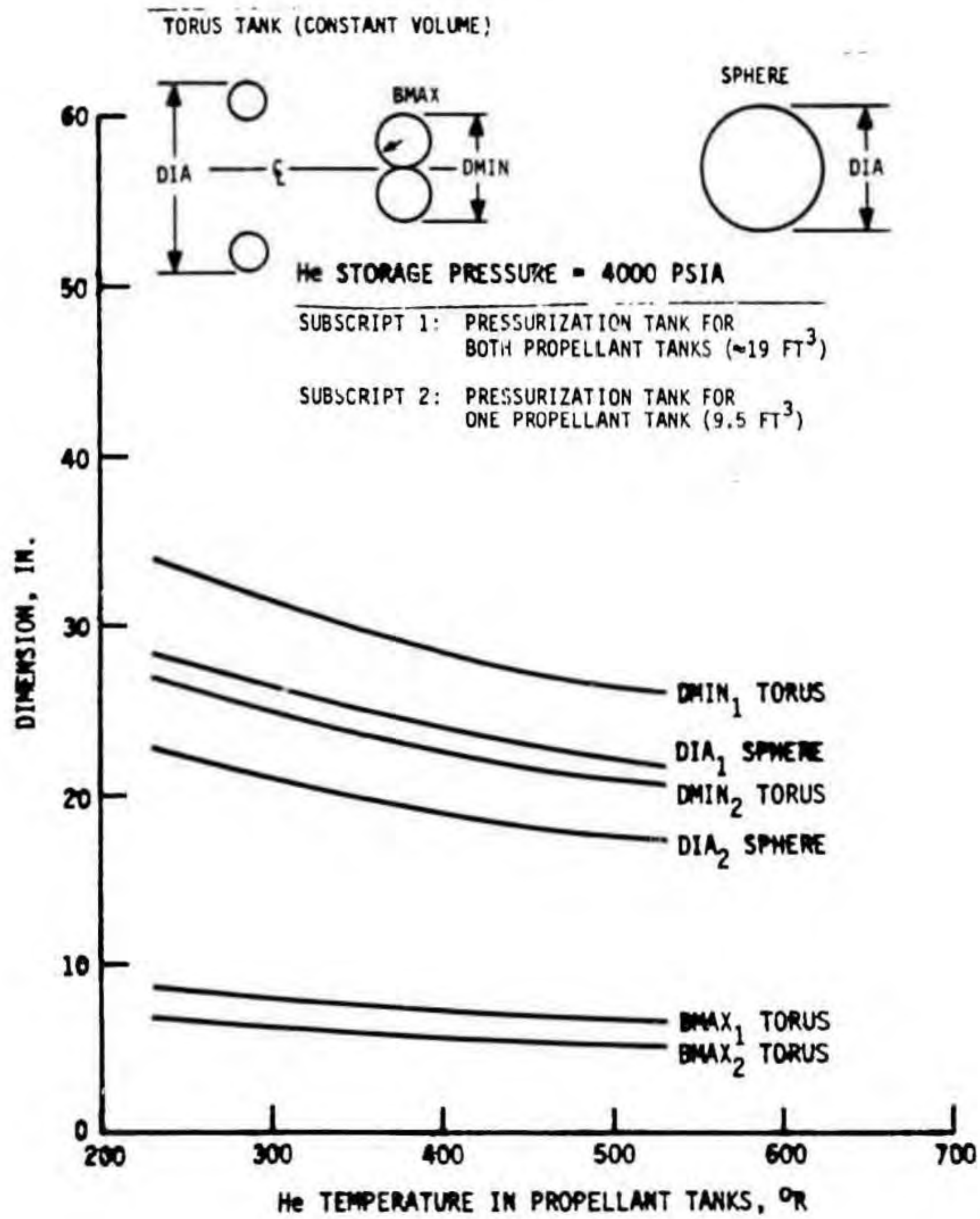


Figure 5-24 Pressurization Subsystem Dimensions vs He Temperature in Propellant Tanks (He Storage Pressure = 4000 psia)

o PCE Mounted Gas/Gas He Heat Exchanger

Advantages:

- o No impact to Spaceplane electrical power requirements.

Disadvantages:

- o Some weight penalty for one or more heat exchangers and He lines.
- o Potential leak path for hot PCE combustion products to pellant tanks.
- o Inflexible packaging (i.e., heat exchangers must be on Spaceplane aft end)
- o Loss of heat source if heat exchangers not located near firing PCE module (heat exchanger near every PCE module is to be avoided because of potentially severe aft end weight penalty with adverse vehicle c.g. effects and He "plumbing" complexity).

The primary purpose of this analysis was to determine how much electrical power would be required to heat the He since, as indicated above, the requirement for additional electrical power is the only, or at least major, drawback to this concept.

The analysis done was based on the use of a constant area duct (or tube) with an external heat source (electrical) resulting in a simple T_o - change process for the He. The results of this analysis showed that the electrical heater would have to maintain, for a one-foot tube, an average tube wall temperature of approximately 191^{o}F to result in a He exit temperature of approximately 70^{o}F. Based on a heat transfer coefficient correlation for turbulent gas flow through a constant area duct, this results in a power requirement of approximately 0.9 KW. It should be remembered that this is the power level required only while the PCE is firing, which is approximately two minutes total. This value may be compared to the power requirements for the other Spaceplane subsystems, as presented by Honeywell, at the Jan TI/TD meeting. These values are presented in Table III. The approximate requirement for the PCE valves is also shown for comparison.

TABLE III
SPACEPLANE POWER REQUIREMENTS

	Voltage, Volts	Average Power, KW	Peak Power, KW	Energy KW-HRS
o Avionics	28	0.25 for 24 hrs	.80	6.0
o Life Support	28	0.30 for 24 hrs	.30	7.2
o Aero-Surfaces	280	2.10 for 0.5 hrs	42.0	1.05
o Propulsion	28	0.65 for .033 hrs	.65	0.02
o He Heating		0.90 for .033 hrs	.90	0.03
			TOTAL	14.30

The table also shows that, although the peak power requirement for He heating is significant compared to the other Spaceplane subsystems, except for the Aero-Surfaces, the resulting energy requirement (.03 KW-HRS) is negligible because the He heating time is so short (.033 hrs or approximately two minutes). The appropriate voltage level (28 v. or 280 v.) should be selected on the basis of impact to the total Spaceplane power system.

The He pressure drop associated with this one-foot long heated tube is relatively small (approximately 12 psia). And, although the He velocity in the tube is relatively high (100 to 300 ft/second), the resulting mach numbers are still low.

It is noted that no effort was made to design, much less optimize, an electrical He heater for application to a constant area duct (or tube). The primary reason for this present preliminary analysis was to determine a reasonable electrical power requirement. A more rigorous analysis, and possible design, of an electrical heater is recommended.

5.4.5 Thrust Vector Control (TVC)

These two TVC methods considered were:

- o Selectable thruster on/off capability
- o Mechanical gimbaling of individual thrusters and/or engine system

It was also observed that:

- o The PCE configuration is capable (with appropriate thrusters shut down) of providing pitch and yaw with axial . It will not provide, without gimbaling of individual thrusters, pure pitch or yaw moments to the vehicle.

- o A Reaction Control System (RCS) will perform the roll maneuver.

Finally:

- o TVC by selective on/off operation, rather than throttling of individual thrusters, is the simplest and preferred concept.

- o A second, and less desirable, TVC system requires the gimbaling of individual thrusters. This system carries a weight penalty (due to gimbal assemblies) and power requirement penalty (due to gimbal actuation power requirements).

5.4.6 Thrust Magnitude Control (TMC)

These three TMC methods were considered:

- o On/off operation of individual thrusters
- o Throttling of individual thrusters and/or engine system
- o Both of the above.

The following observations were also made:

- o Throttling of individual thrusters is both technically difficult because of resulting low flowrates and also appears to be expensive.
- o The remaining two options (individual thruster on/off operation and engine system throttling) are preferred. The tank safety valves could be throttleable valves, or the tank pressure could be varied, for example.

On the basis of these considerations:

- o TMC accomplished by selective on/off operation of individual pairs of thrusters is the preferred concept.
- o A second, or backup, concept would be throttling of individual thruster pairs.
- o As a typical design, a 20 engine cluster of 300 lbF modules (6000 lb total axial thrust) will permit 10:1 throttling capability.

5.5 TASK 2.1 PERFORMANCE

The primary requirement of this task was to present both propulsion system delivered specific impulse (Isp) and Spaceplane vehicle (SP) as a function of nozzle area ratio, chamber pressure, propellant mixture ratio and plug cluster engine configuration.

5.5.1 Performance Prediction Methodology

First of all, the methodology for calculating plug cluster engine (PCE) performance had already been developed at ALRC under Contract NAS 3-20109 (Ref. 4). It was based on the use of the JANNAF Simplified Performance Prediction Methodology (Ref. 5) described here. This methodology may be summarized by this expression:

$$Isp_d = Isp_{ode} \frac{DIV \cdot ERE \cdot KIN}{\left(1 + \frac{F_{BL}}{F}\right)}$$

where:

Isp_d	=	Isp delivered (seconds)
Isp_{ode}	=	Isp ode (seconds)
DIV	=	Nozzle efficiency
ERE	=	Energy release efficiency
KIN	=	Kinetic efficiency
F_{BL}	=	Boundary layer loss (lbF)
F	=	Delivered thrust (lbF)

This simplified methodology provides a very cost-effective procedure, with results comparable in accuracy to more rigorous methods when properly utilized. This procedure is described in Section 3 of Reference 5. As depicted schematically in Figure 25, it consists of starting with one-dimensional equilibrium specific impulse (Isp_{ode}) and correcting the performance downward for contributing component performance losses. Subtracting the kinetic, divergence, and boundary layer losses from Isp_{ode} in Figure 25 provides a hypothetical "perfect injector performance" that is degraded for the real nozzle losses. This performance is further degraded by the energy release loss which accounts for the injector related performance inefficiencies due to incomplete propellant vaporization and/or non-uniform gas-phase mixing. When chamber pressure becomes too high, it may become necessary to provide auxiliary cooling from the combustion gas-side through such means as barrier or zone mixture ratio cooling, film cooling, or transpiration cooling.

Once the performance of the individual thruster is calculated by way of the JANNAF simplified methodology, the performance of the PCE is determined by the following steps:

1. Determine axial thrust loss of modules due to tilting.

2. Determine the average pressure of the recirculating gases on the plug base (this pressure is an assumed function of the PCE total area ratio and PCE module chamber pressure).

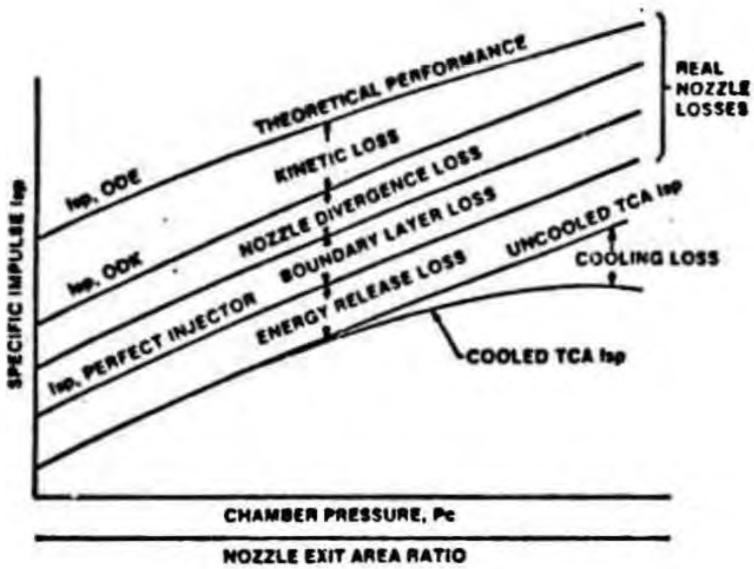


Figure 5-25 **Simplified JANNAF Performance Methodology Predicts Plug Cluster Engine Module Performance**

3. The thrust of the plug base is then the base area multiplied by the average base pressure.

4. With the total thrust (due to tilted modules and plug base) known, the PCE performance (Isp) is found by dividing the total PCE thrust by the total flowrate going to all of the individual thrusters.

A method for defining the performance loss due to engine out operation, which lets the recirculating gases "trapped" on the plug base escape, has yet to be defined. The effect of scarfing the module nozzles, if scarfing is done, also needs to be defined. The work done under the present contract did not consider either of these effects.

This performance methodology was then used with engine system and component weight scaling relationships, described in the next section, to generate PCE parametric performance, weight and envelope data.

5.5.1 Weight and Envelope Prediction Methodology

Both the weight and envelope scaling relationships include PCE parameters such as module exit diameter, module tilt angle, engine system diameter, plug base area and engine thrust level as functions of number of modules, chamber pressure, and module thrust level.

These scaling equations are formulated around the existing ALRC 5 lbF and 100 lbF RCS bipropellant thrusters as the baseline case. The total propulsion system weight is determined by calculating and adding together the weights of all the major components included in the propulsion system. These included:

- Plug base
- Injectors
- Combustion chambers
- Nozzles
- Valves and actuators
- Propellant lines
- He bottle(s)
- He required
- Tank pressurant heat exchangers and associated equipment
- Propellant tanks
- Miscellaneous (electrical harness, instrumentation, brackets, auxiliary line and controls)

A contingency was not included because this is normally accounted for in the total vehicle weight statement.

Most of the conceptual design engine baseline component weights were established initially from the historical data base on components of similar nature from existing engines of comparable thrust and chamber pressure (such as the ALRC 5 lbF and 100 lbF RCS thrusters). If a component did not exist at a particular design point, baseline weights were established through the use of scaling equations (i.e., scale what does exist to the desired condition).

Scaling equations are formulated around these baselines from geometric considerations (i.e., dimensions and surface areas) along with known component weight trends with the design variables of interest. For engine weight scaling, this is generally thrust chamber pressure, thrust and nozzle area ratio or parameters which can be related to one of these (i.e., thrust and flow or chamber pressure and pump discharge pressure).

This procedure has been utilized effectively in several past and continuing ALRC contracted and in-house studies such as the Advanced HiPc Program (NAS 3-19727), the Mixed-Mode OTV Study (NAS 3-21049), the Unconventional Nozzle Tradeoff Study (NAS 3-20109), the Orbit-to-Orbit Shuttle Engine Design Study (F04611-71-C-0040), the Storable Space Tug Study (NAS 8-29806), OTV Engine Phase "A" Study (NAS 8-32999), Dual-Fuel, Dual-Throat Engine Preliminary Analysis (NAS 8-32967), and the parametric analyses conducted for the early Phase B Shuttle Main Engine Definition Study (Contract NAS 8-26188). Existing pump-fed engines such as the Agena, RL-10 and Titan II second stage and study engines such as the OOS, Storable Tug and RL-10 derivatives provided the original basis for component weight data. For this study, the ALRC 5 lbF and 100 lbF RCS thruster data will be the primary checks on the weight data obtained from the scaling equations.

Engine length is the summation of injector, chamber and nozzle lengths. As with the baseline component weights, the baseline injector and chamber lengths were determined on the basis of the ALRC 5 lbF and 100 lbF RCS thrusters. The nozzle exit diameter is calculated as a function of nozzle area ratio, thrust and chamber pressure.

Nozzle length is determined from the thrust, area ratio and chamber pressure. The percent bell nozzle (85%) is based upon the ALRC 100 lbF RCS thruster nozzle. The injector and chamber lengths are based upon geometrical scaling of the baseline values.

5.5.3 Computer Program SPV1

A computer program, SPV1 (for Spaceplane Version 1) was developed to generate onboard propulsion system parametric performance (including Spaceplane), weight and envelope data. SPV1 incorporates fully both the performance prediction and weight and envelope prediction methodologies discussed in the two previous sections.

Some of the programs more important features are outlined below:

- Terminal operated
- Completely self-tutorial and interactive
- 24 required inputs (input default value always displayed)
- Performance calculation for both N_2O_4/PAAB-1 and N_2O_4/MMH based on JANNAF simplified methodology described previously
- Utilizes propulsion system weight and envelope prediction methodology described previously
- Generates weights, envelopes and c.g. locations of (1) spherical, (2) conformal, (3) toroidal, and (4) integral bulkhead fuel/ox conformal propellant tanks
- Other propulsion subsystems and components modeled (i.e., weight and envelope determination) include:

- plug base
- propellant lines
- valves and actuators
- injectors
- combustion chambers
- nozzles
- He pressurization subsystem
- RCS

- o Calculates c.g. attributable to propellants only
- o Calculates c.g. of PCE
- o Calculates total predicted ideal Spaceplane vehicle

SPV1 was used to generate the pressurization subsystem weight and envelope data presented in Task 1.2 (Candidate Subsystems Definition). SPV1 was also used to generate the propulsion system parametric performance, weight and envelope data presented in the next report section (Preliminary Parametric Data). SPV1 was also used extensively in Task 3.8 (Selected Design) to optimize the onboard propulsion system using either propellant combination.

5.5.4 Preliminary Propulsion System Parametric Data

In support of the SOW Task 2.1 requirement, SPV1 was used to generate preliminary propulsion system parametric performance (SP), weight and envelope data. Specifically, these data was generated in three categories:

- o Mixture ratio optimization
- o SPΔV vs. propellant volume
- o Propulsion system parametric performance, weight and envelope data.

These data are presented below.

5.5.4.1. Propellant Mixture Ratio Optimization

The optimum mixture ratio (MR) is defined as the MR which results in the maximum vehicle ΔV. This optimization was done for both the N_2O_4/PAAB-1 and N_2O_4/MMH propellant combinations. From a vehicle ΔV standpoint, assuming a fixed volume for onboard propellants, a mixture ratio of 1.481 for N_2O_4/PAAB-1 is optimum. It also results, coincidentally, in equal volume fuel and oxidizer tanks. An equivalent analysis for N_2O_4/MMH showed that a mixture ratio of 2.4 is optimum. Neither of these analyses, however, took into account the influence of film cooling.

The effect of fuel film cooling on the optimum N_2O_4/MMH mixture ratio was evaluated first. In general, the effect of fuel film cooling is to decrease both the delivered specific impulse and the MR at which the maximum delivered specific impulse occurs. This principle is conceptually illustrated in Figure 26.

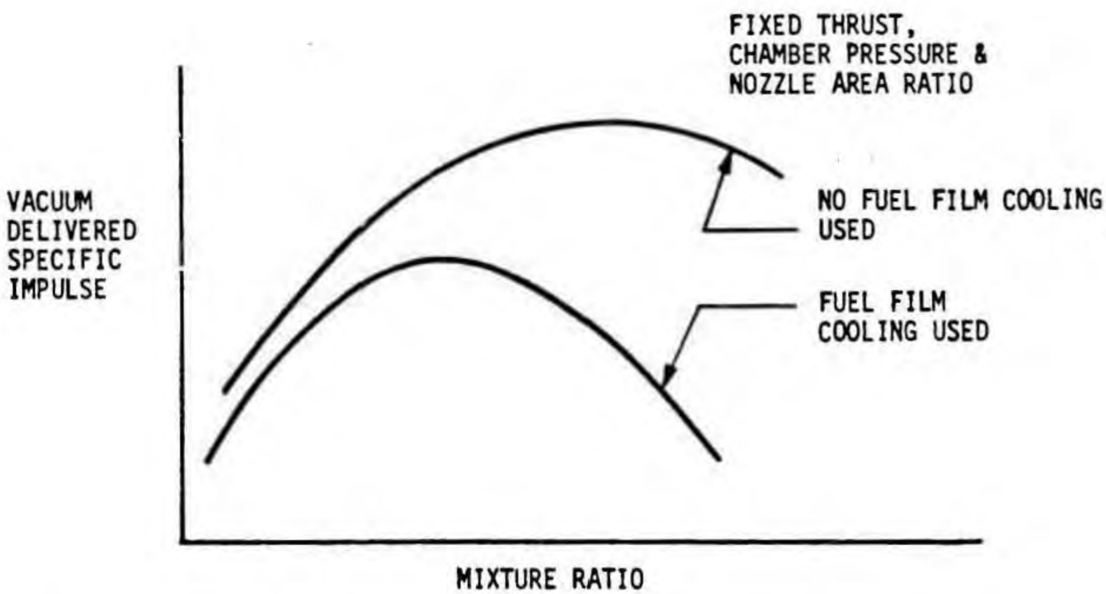


Figure 5-26 **Effect of Fuel Film Cooling on Vacuum Delivered Specific Impulse**

The analysis takes into account the effect of propellant bulk density as well as specific impulse on delivered vehicle ΔV.

The results of this optimization showed that, for a Spaceplane class vehicle, an MR of 1.65 with N_2O_4/MMH is very close to the optimum. The fact that the curve of ΔV vs. MR is very flat, indicates that the PCE MR could be anywhere between 1.5 and 2.0 with only minor ΔV penalties (less than 50 ft/sec). Also, since the existing ALRC engines were designed to operate at an MR of 1.65, 1.65 was assumed for the purpose of performing the PCE optimization. With both propellant combinations (N_2O_4/MMH and N_2O_4/PAAB-1), a lower MR is desirable because it reduces the oxidizer volume and hence tank size, allowing easier line routing around the periphery of the tank.

A similar analysis for N_2O_4/PAAB-1 yielded an optimum propellant mixture ratio of 1.2. All of the data required in this mixture ratio optimization was generated by SPV1. These values were used as constants in the propulsion system optimization performed in Task 3.8 (Selected Design).

5.5.4.2. Spaceplane ΔV vs. Propellant Volume

The generation of plots of Spaceplane vehicle ΔV versus propellant volume for both (N_2O_4/MMH and N_2O_4/PAAB-1) propellant combinations was completed primarily as an aid to the Spaceplane vehicle integrator (H/S). They are reproduced here in Figures 27 and 28. These plots are based on the ideal vacuum equation:

$$\Delta V = g_c \; Isp \; \ln (M_I/M_{BO})$$

where:

ΔV	=	total vehicle velocity increment, ft/sec
g_c	=	gravitational constant, 32.174 ft/sec^2
Isp	=	delivered specific impulse of propulsion system, seconds
M_I	=	initial (before propulsion system operation) vehicle mass, lbM
M_{BO}	=	final, or burnout (depletion of propellants) vehicle mass, lbM

5.5.4.3. Preliminary Parametric Data

SPV1 was used to generate the parametric data plotted in Figures 29, 30, and 31. These data are preliminary, since the performance of the N_2O_4/PAAB-1 propellant combination require some refinement which was accomplished before the propulsion system optimization of Task 3.8 (Selected Design). The performance (Isp) values shown are approximately 2.5% (7 to 8 sec) conservative. This deficiency was later corrected. In the interim, hand calculations showing the influence of this Isp bias on the total Spaceplane performance were made.

The resulting preliminary estimates of the Spaceplane performance vs. propellant characteristics are shown in Table IV. The assumptions made for the calculations are that the initial vehicle weight is 6275 lbM, that the baseline propellants will deliver 320 sec. specific impulse, and that the PAAB-1 propellant performance is 1.2% higher.

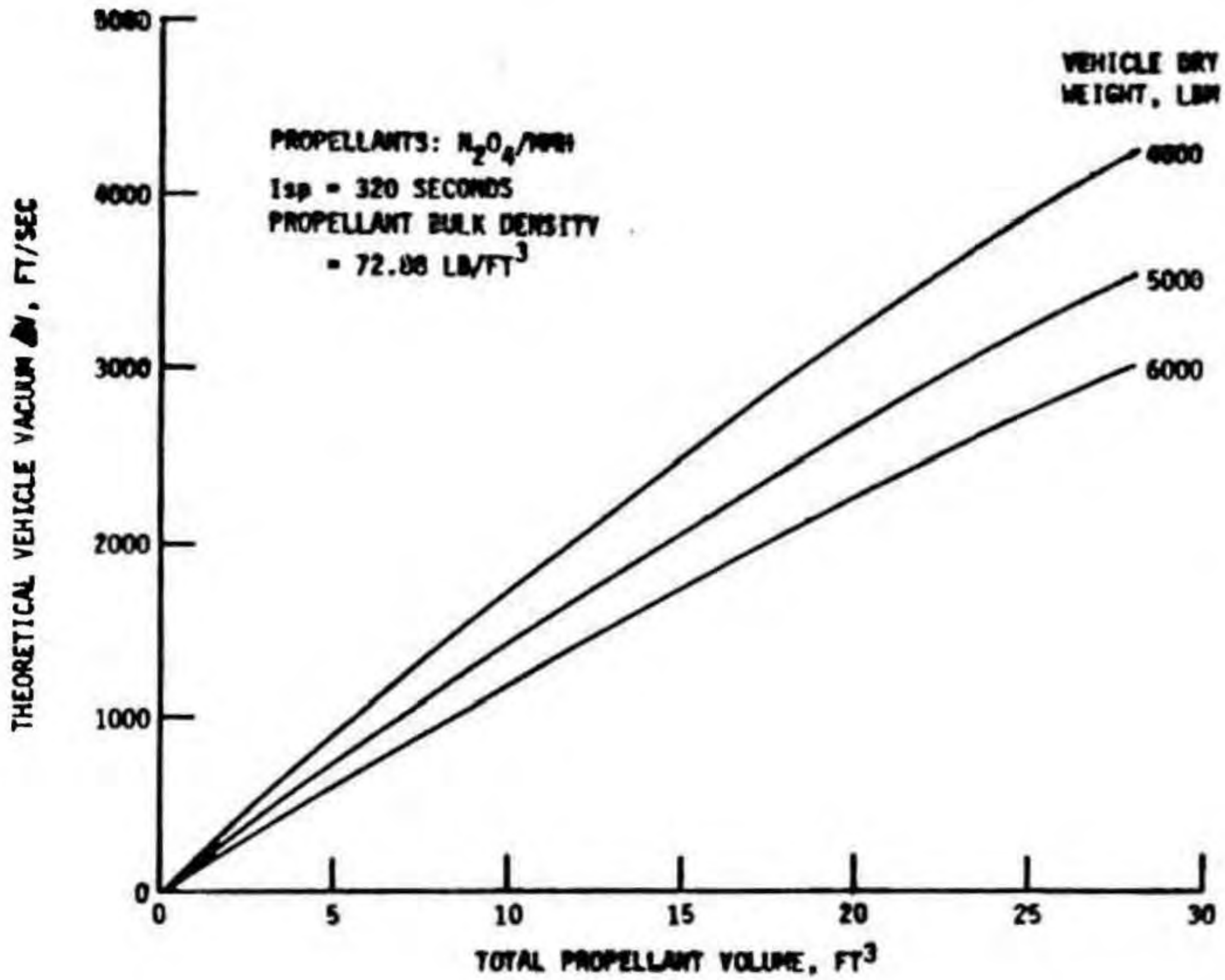


Figure 5-27 Spaceplane ΔV versus Propellant Volume (N_2O_4/MMH)

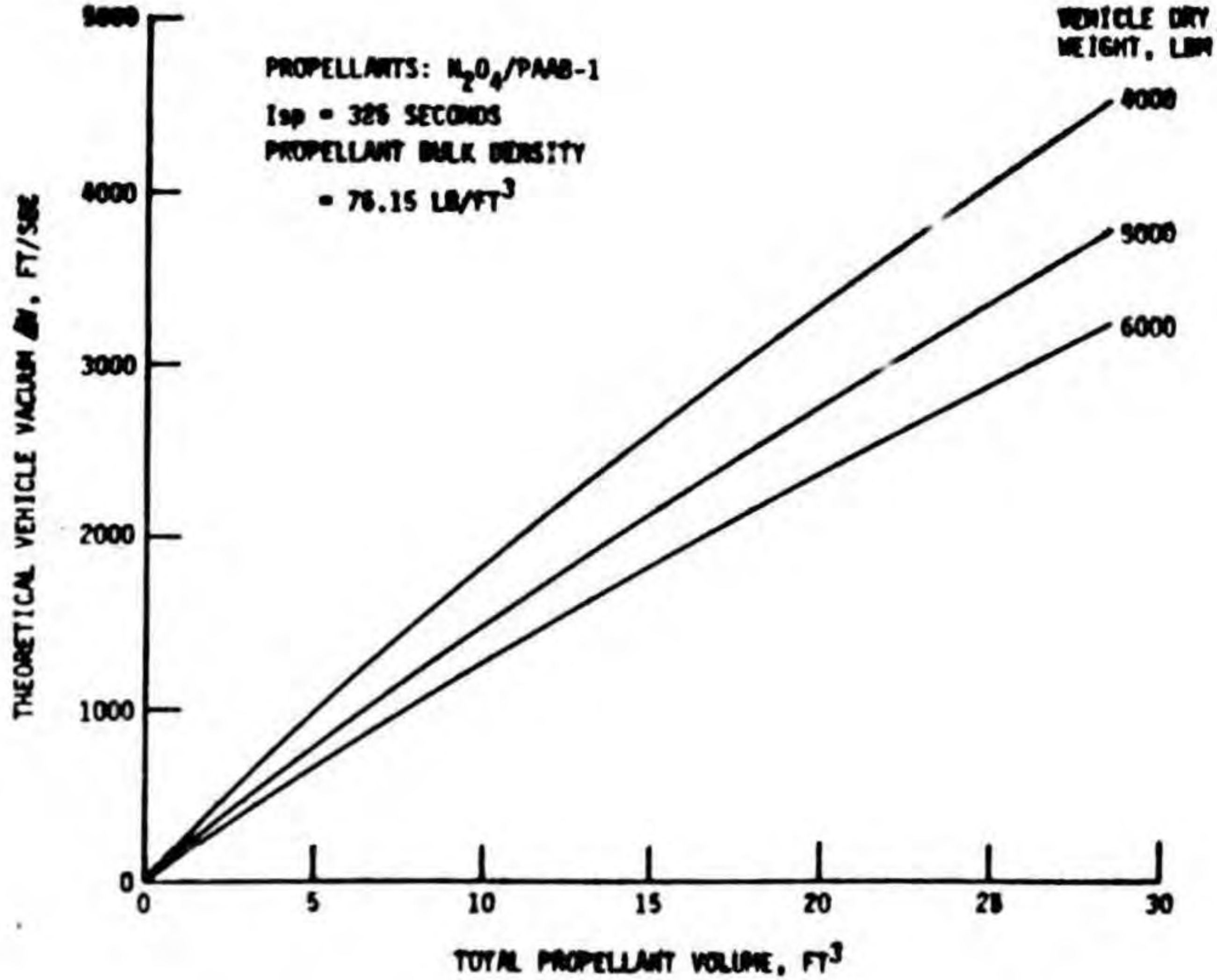


Figure 5-28 Spaceplane ΔV versus Propellant Volume (N_2O_4/PAAB-1)

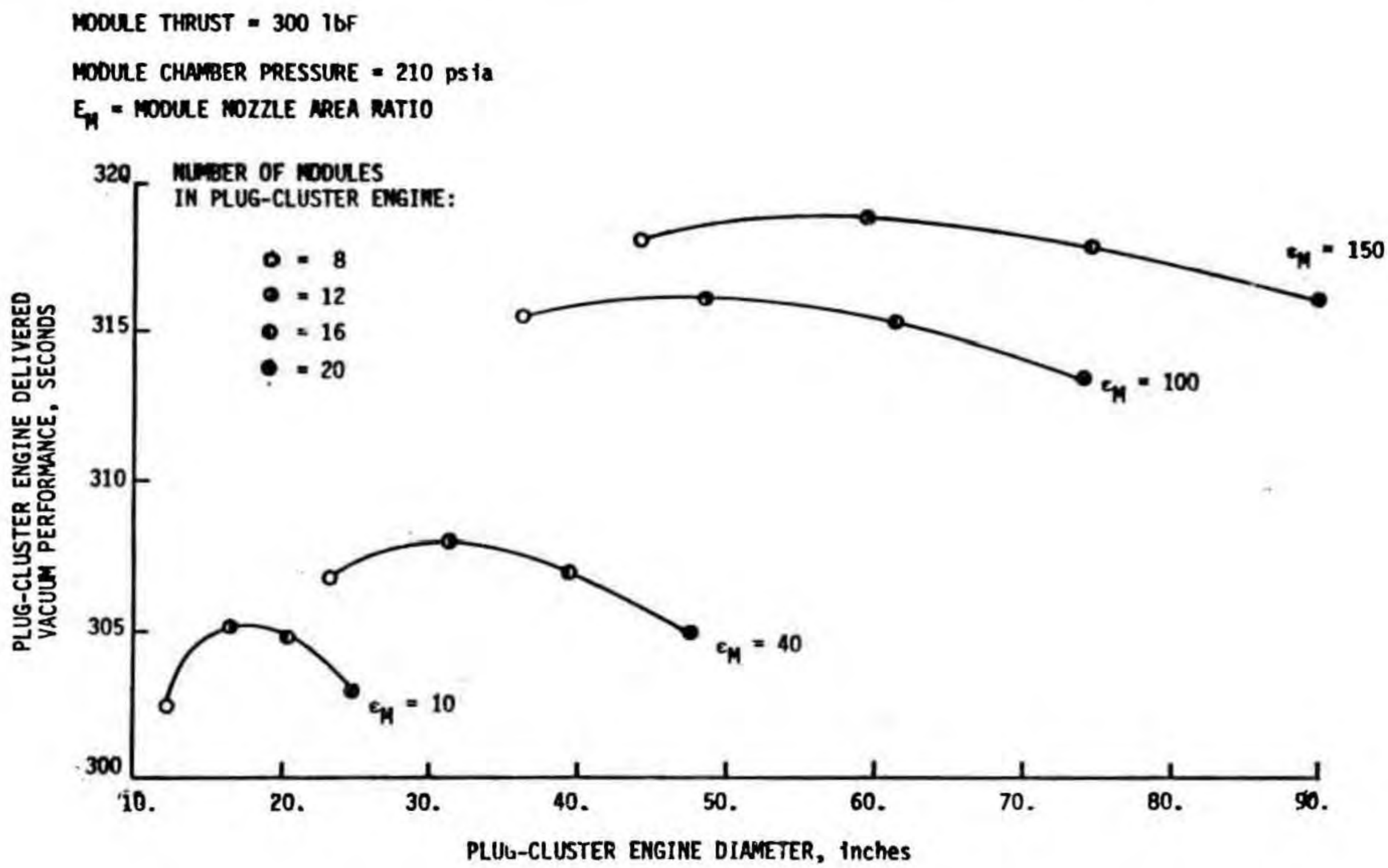


Figure 5-29 Plug Cluster Engine (PCE) Delivered Isp vs. PCE Diameter

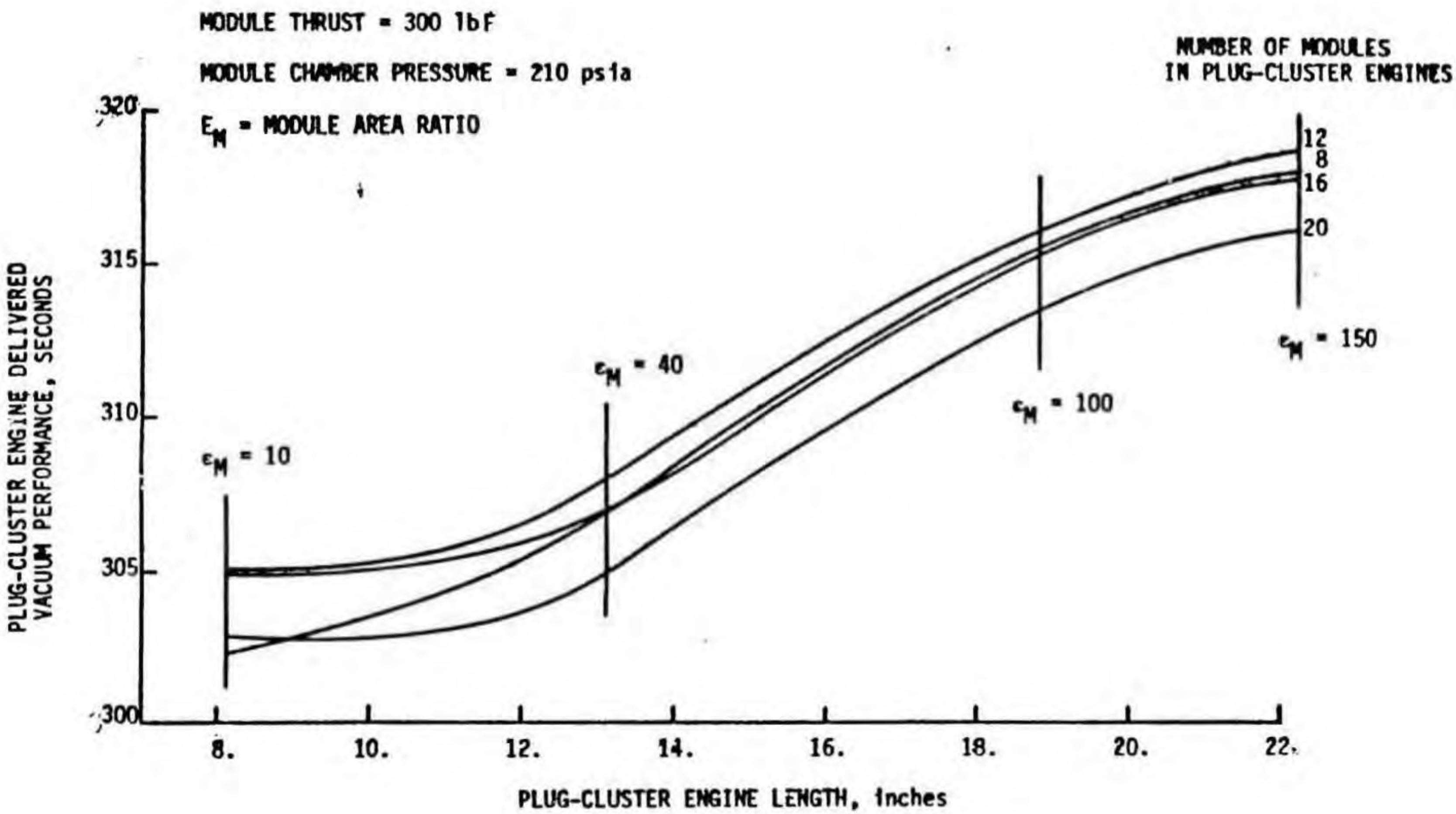


Figure 5-30 Plus Cluster Engine (PCE) Delivered Isp vs. PCE Length

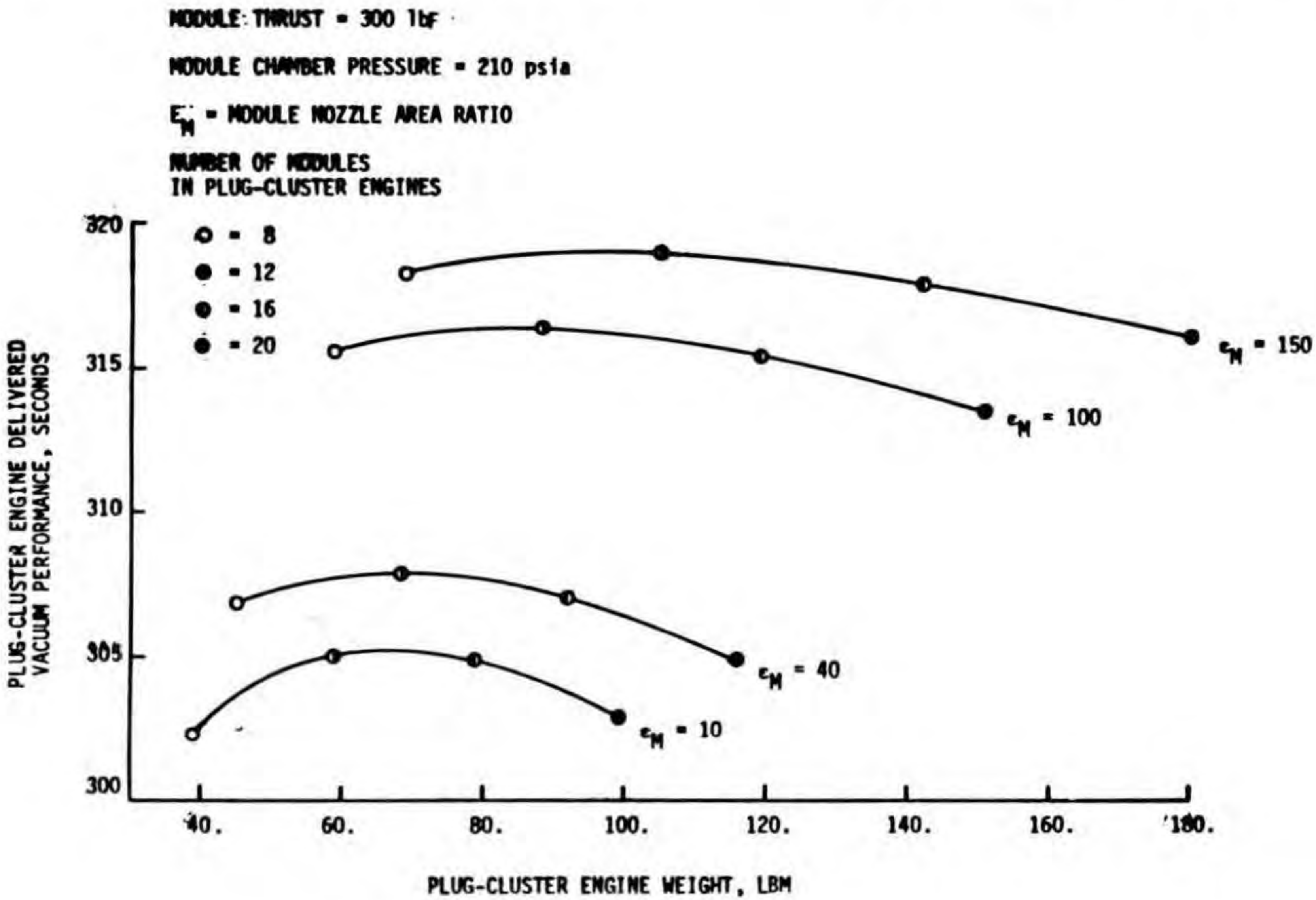


Figure 5-31 Plug Cluster Engine (PCE) Delivered Isp vs. PCE Weight

TABLE IV

SPACEPLANE ΔV VERSUS PROPELLANT MASS

Fuel/N_2O_4	Propellant Mass, lbM	Velocity Increment, ft/sec
MMH	1200.	2185.
MMH	1300.	2390.
MMH	1353.	2500.
PAAB-1	1200.	2211.
PAAB-1	1300.	2419.
PAAB-1	1339.	2500.

The trends illustrated by the parametric plots (Figures 29, 30 and 31) do not depend upon the absolute value of specific impulse. Several important observations can be drawn from these plots by considering the maximum Spaceplane diameter constraint of approximately 56 inches and initial thrust requirement of 4500 to 6500 lbF.

First, from Figure 29, the highest performance PCE will have module nozzle area ratios between 40 and 100. The resulting delivered performance will be somewhat in the 310 sec. range, not counting the 7-to-8 sec. bias discussed previously. Higher performing PCE's (up to approximately 320 sec.) can be obtained, but only at lower thrust levels. This can be seen in Figure 29, by assuming that PCE thrust is approximately equal to the number of modules times the thrust per module (300 lbF in this parametric study).

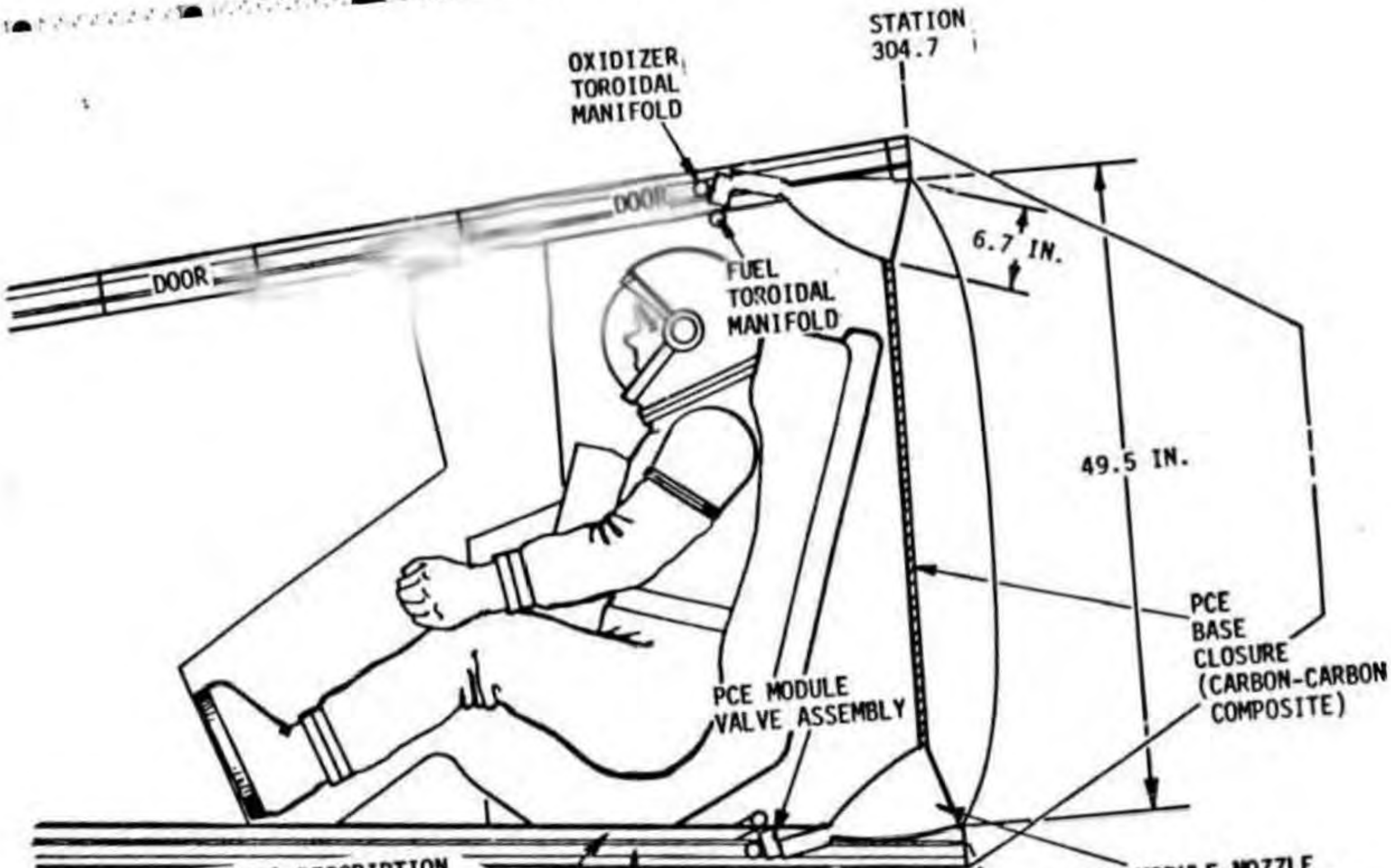
Figure 30 shows that the engine length, which consists essentially of the module chamber and nozzle, will be approximately 16 inches. Figure 31 shows that this PCE will weight approximately 130 to 140 lbM.

In addition to the performance prediction refinement already noted, several improvements to SPV1 were still required to complete Tasks 2.1 and 2.2. These modifications are noted here:

- Addition of He and He bottle weight (and size) calculation
- Improvement of Isp prediction for both N_2O_4/MMH and N_2O_4/PAAB-1
- Addition of PCE c.g. location calculation

SPV1 was then used to provide the important dimensions for several candidate PCE systems. These sketches were reduced, and reproduced in Figures 32 through 36 in order of decreasing PCE diameter. Note the relationship of delivered Isp to PCE diameter.

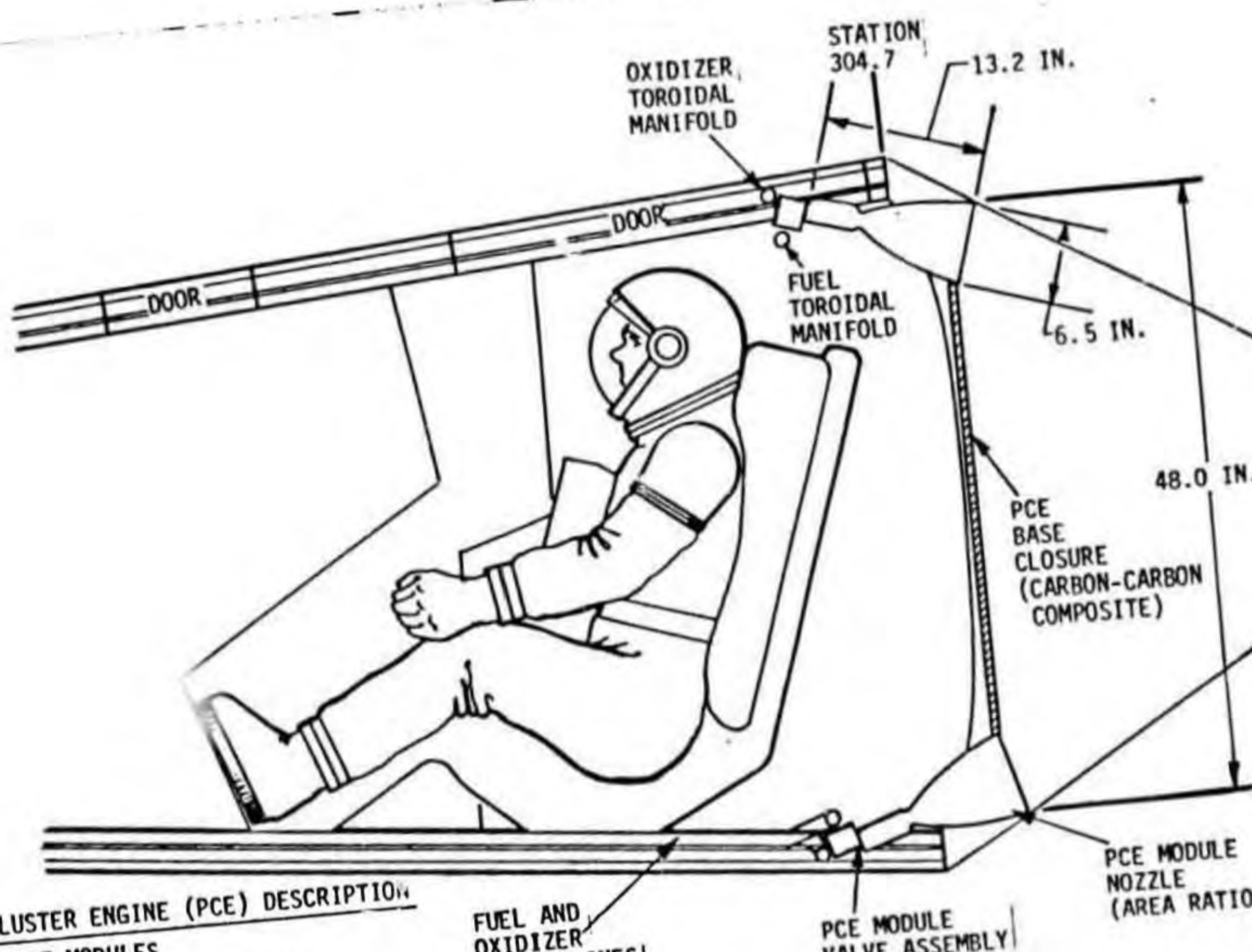
The PCE configuration shown in Figure 34 was selected by both H/S and SNLA for the initial Spaceplane vehicle integration effort. Although several propulsion system parameters changed (such as propellants used, tank pressure, PCE module chamber pressure, number of PCE modules, etc.) during the propulsion system optimization conducted in Task 3.8 (Selected Design), the PCE major dimensions (diameter of 43.5 in. and length of 12.2 in.) were assumed fixed for the balance of the study. These two values were considered design constraints in Task 3.8.



PLUG CLUSTER ENGINE (PCE) DESCRIPTION

1. 20 PCE MODULES
2. TOTAL PCE THRUST = 6050 LBF
3. PCE MODULE THRUST = 300 LBF
4. TILT ANGLE = 17°
5. FULL THRUST PCE DELIVERED SPECIFIC IMPULSE = 305.9 SECONDS
6. PROPELLANTS: N_2O_4/MMH (MIXTURE RATIO = 1.65)

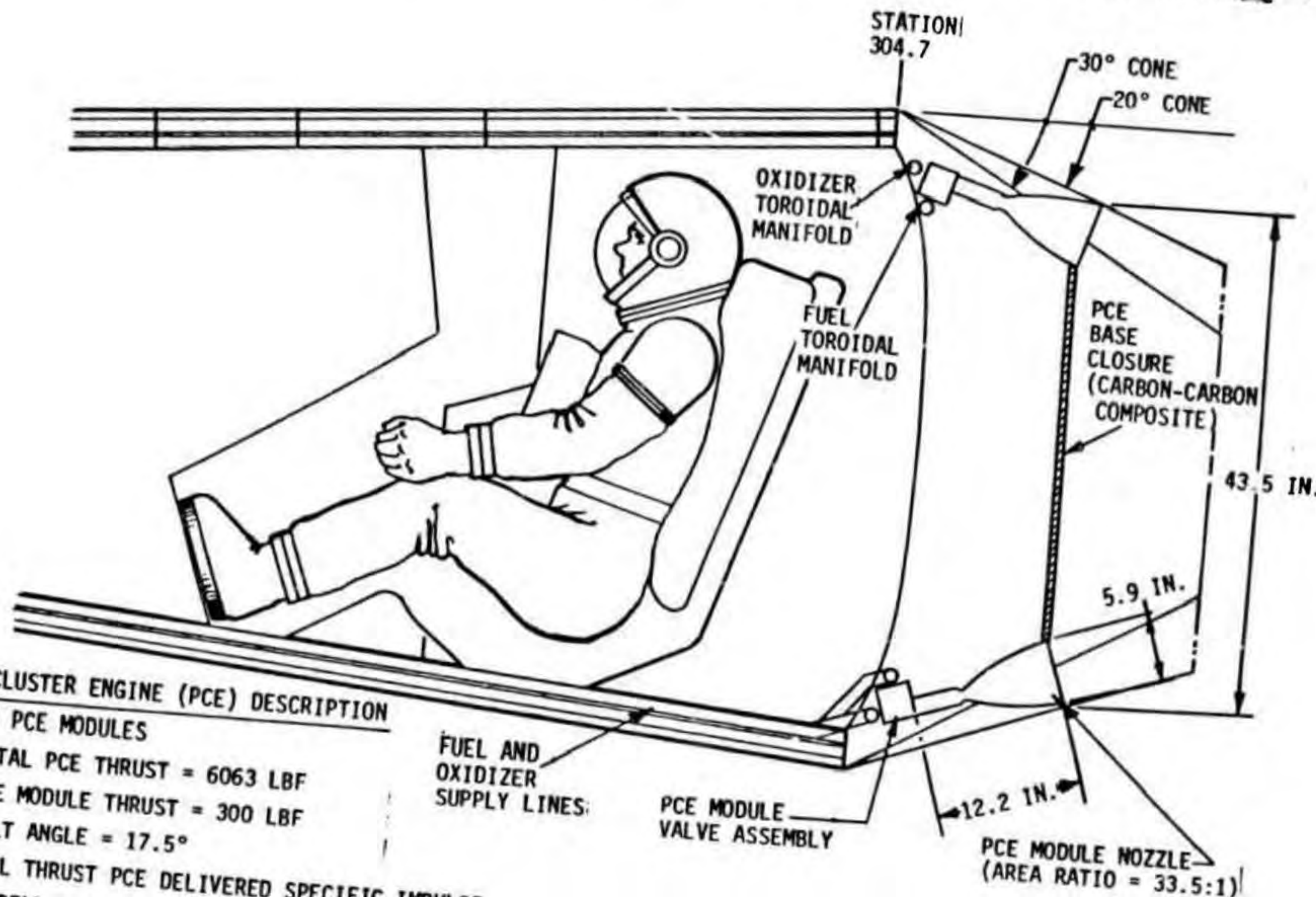
Figure 5-32 **Plug Cluster Engine Configuration No. 1**



PLUG CLUSTER ENGINE (PCE) DESCRIPTION

1. 20 PCE MODULES
2. TOTAL PCE THRUST = 6054 LBF
3. PCE MODULE THRUST = 300 LBF
4. TILT ANGLE = 17°
5. FULL THRUST PCE DELIVERED SPECIFIC IMPULSE = 305.2 SECONDS
6. PROPELLANTS: N_2O_4/MMH (MIXTURE RATIO = 1.65)

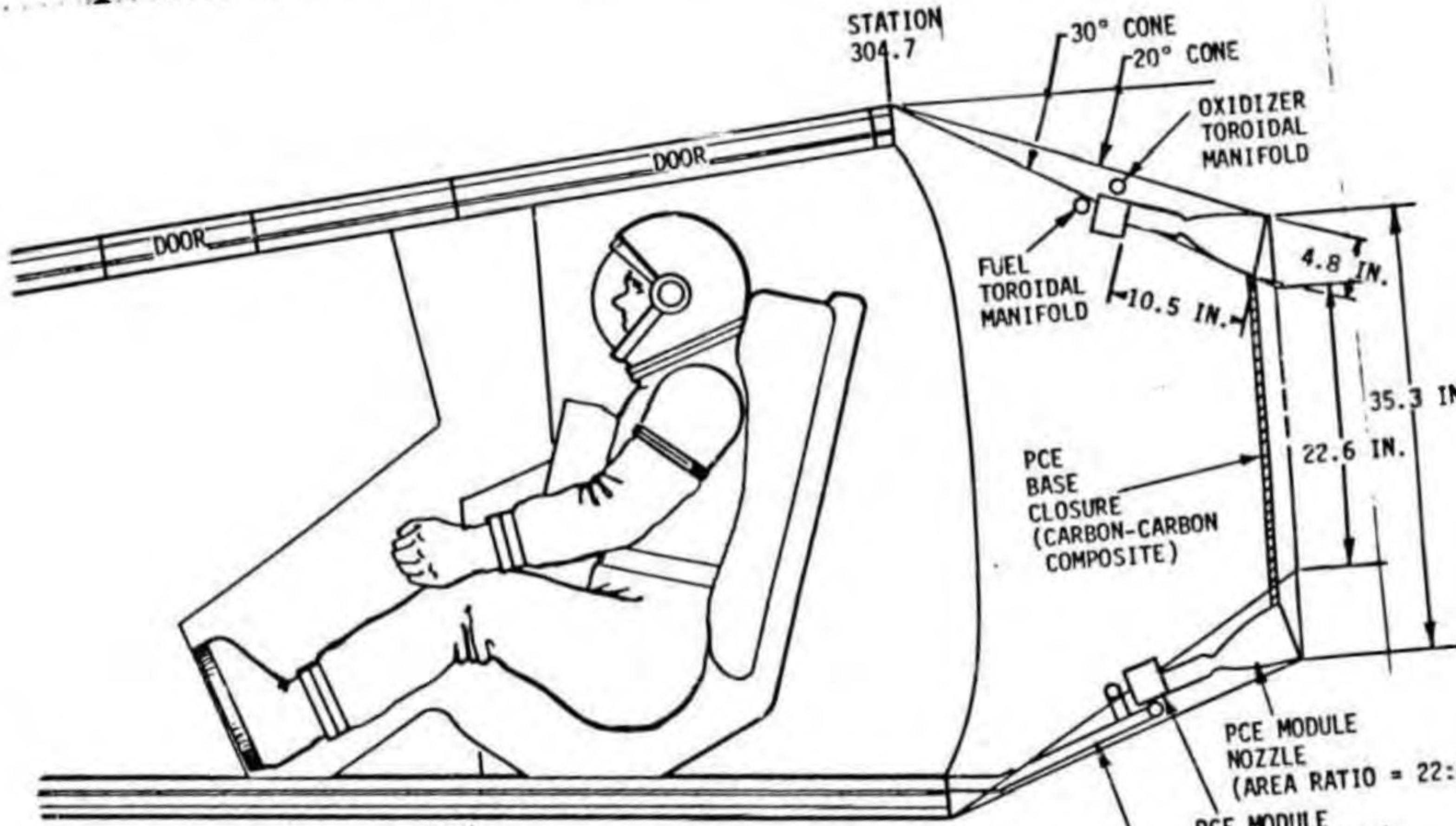
Figure 5-33 **Plug Cluster Engine Configuration No. 2**



PLUG CLUSTER ENGINE (PCE) DESCRIPTION

1. 20 PCE MODULES
2. TOTAL PCE THRUST = 6063 LBF
3. PCE MODULE THRUST = 300 LBF
4. TILT ANGLE = 17.5°
5. FULL THRUST PCE DELIVERED SPECIFIC IMPULSE = 303.2 SECONDS
6. PROPELLANTS: N_2O_4/MMH (MIXTURE RATIO = 1.65)

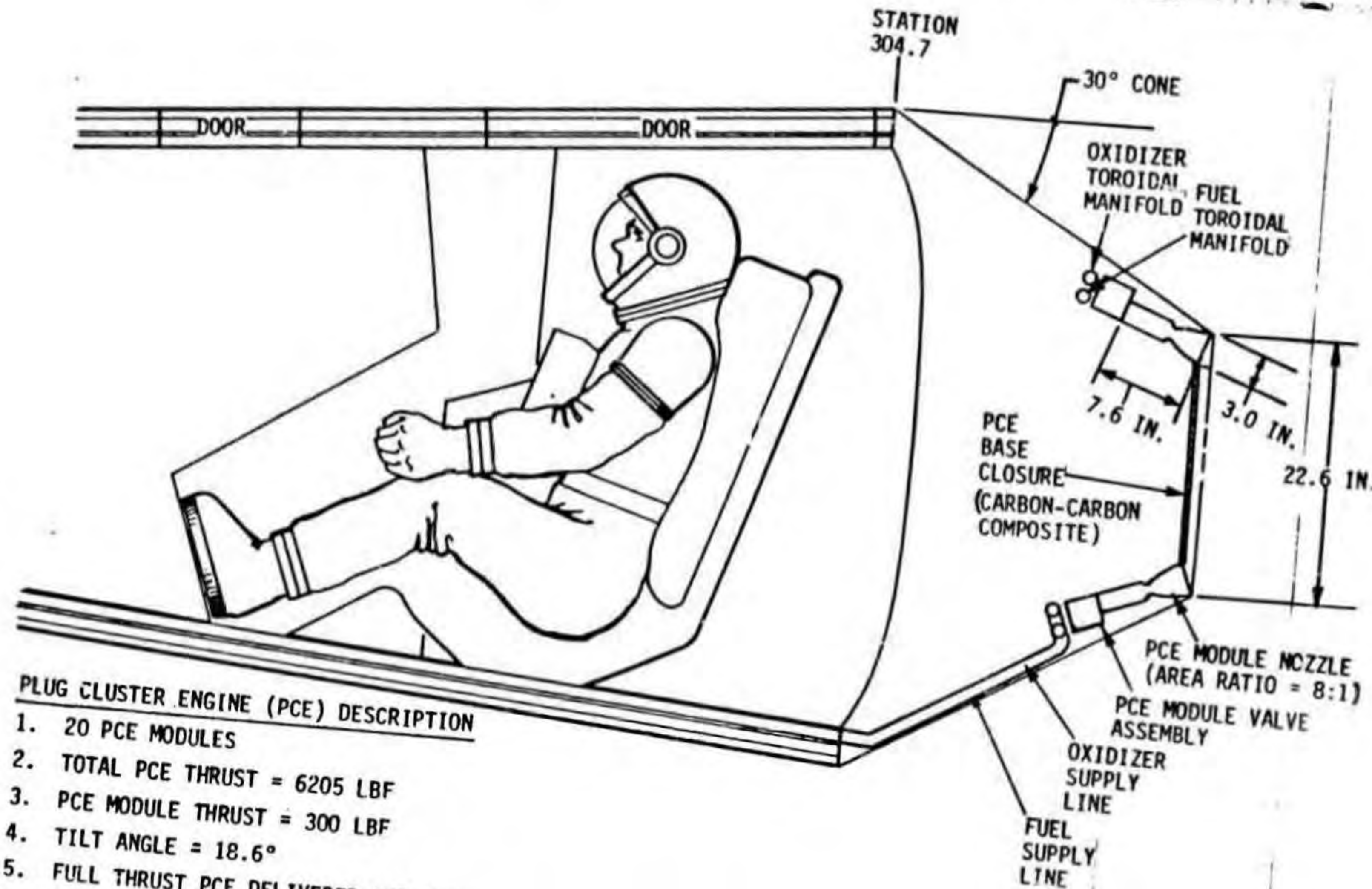
Figure 5-34 **Plug Cluster Engine Configuration No. 3 (Preliminary Baseline)**



PLUG CLUSTER ENGINE (PCE) DESCRIPTION

1. 20 PCE MODULES
2. TOTAL PCE THRUST = 6089 LBF
3. PCE MODULE THRUST = 300 LBF
4. TILT ANGLE = 18°
5. FULL THRUST PCE DELIVERED SPECIFIC IMPULSE = 299.7 SECONDS
6. PROPELLANTS: N_2O_4/MMH (MIXTURE RATIO = 1.65)

Figure 5-35 **Plug Cluster Engine Configuration No. 4**



PLUG CLUSTER ENGINE (PCE) DESCRIPTION

1. 20 PCE MODULES
2. TOTAL PCE THRUST = 6205 LBF
3. PCE MODULE THRUST = 300 LBF
4. TILT ANGLE = 18.6°
5. FULL THRUST PCE DELIVERED SPECIFIC IMPULSE = 287.4 SECONDS
6. PROPELLANTS: N_2O_4/MMH (MIXTURE RATIO = 1.65)

Figure 5-36 **Plug Cluster Engine Configuration No. 5**

5.6 TASK 2.2 WEIGHT AND C.G.

The main objective of this task was to provide: (1) propellant weights and propulsion system dry weights for all of the candidate propulsion subsystems considered in Task 1.2, and (2) the center of gravity for one Spaceplane vehicle. Much of the required weight and/or c.g. data was presented previously in the Task 1.2 (Candidate Subsystem Definition) discussion. This included the weight and c.g. of different propellant tank configurations as well as the weight of candidate propellant pressurization subsystems (PS's). The weight of the propellant control subsystem was assumed to be part of the propellant tank weight. Because of the relatively low weight of the selected propellant control subsystem (elastomeric diaphragms), this was a reasonable assumption.

All of the tank weights shown were based on a safety factor of 4. Although this results in a very conservative tank wall thickness, the resulting total tank weights were determined, on the basis of existing flight proven tanks, to be realistic. All of these tank-weight and envelope data were generated by the computer program SPV1.

A⁻ o of primary importance is the location of the c.g. attributable to the p ellant only. This position can be located by placing the propellant tanks in the appropriate positions. Figures 9 through 12 presented in the Task 1.2 description (Candidate Subsystem Definition), show four different tankage configurations. The c.g. locations noted in these figures are the c.g. locations of the full volume of propellant contained by the particular tanks. In most cases (except for the conformal tanks) this c.g. location can be considered coincident with the actual tank-only c.g. location.

Ultimately, based on discussions with SNLA, Station 221.6 corresponded most closely to the desired vehicle c.g. However, this does not influence the tank weight comparison as shown in Figures 9 through 12. This vehicle c.g. at Station 221.6 is represented by the H/S layout drawing shown in Figure 37.

A determination of the PS c.g. location was not practical at the time Task 2.2 was in progress since the number and location of He bottles was a vehicle integration issue that was not addressed until the end of the study. The bulk of the PS weight was ultimately between the rear payload bay and PLSS (Portable Life Support System) which was attached to the rear of the pilot's couch. This arrangement is also illustrated in Figure 37.

Engine system (plug cluster engine) parametric weight data was presented in the previous report section (Task 2.1, Performance). Engine system c.g. data is documented in the Task 3.8 (Selected Design) description. The computer program SPV1, described previously, calculates the c.g. of the PCE, propellants, and propellant tanks (full and empty).

The weight influencing parameters are generally thrust, chamber pressure, number of modules and module nozzle area ratio. The components and subsystems modeled by SPV1 in this way include:

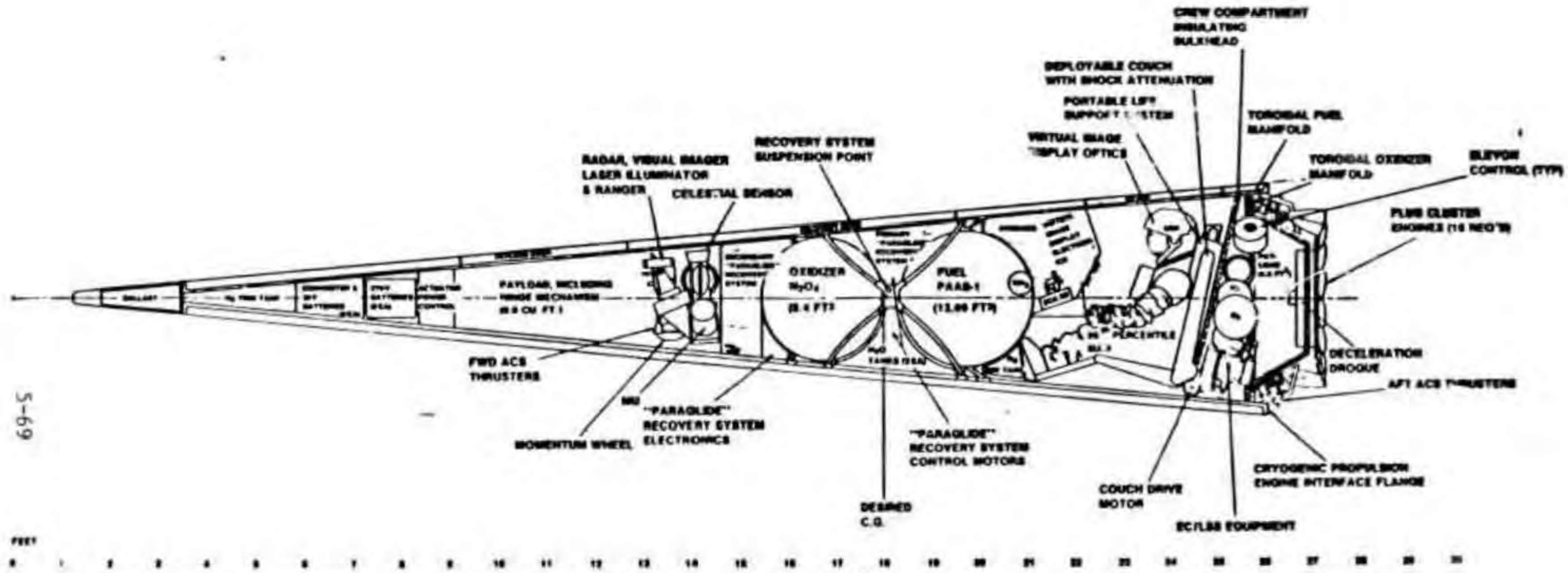


Figure 5-37 **Hamilton/Standard Integrated Spaceplane Vehicle**

- plug base
- propellant lines
- valves and actuators
- injectors
- combustion chambers
- nozzles
- conformal propellant tank(s) with flat or elliptical (2:1) bulkheads
- spherical propellant tanks
- toroidal propellant tanks
- He pressurization subsystem
- RCS

Besides generating parametric weight and c.g. data, SPV1 was later used to optimize the selected baseline PCE (Task 3.8, Selected Design). The RCS weight was assumed to be a constant 45 lbM. This preliminary weight estimate proved to be fairly accurate, as discussed in detail in the Task 2.4 (RCS Definition) description. RCS c.g. data was not practical, or particularly useful, since the RCS thruster weight was so low and since the location, and hence c.g., of the RCS thrusters was not defined until Task 2.4 (RCS Definition).

5.7 TASK 2.3 OPERATION AND CONTROL

The objective of this task was to describe the operation and control of a conceptual Spaceplane onboard propulsion system.

A preliminary assessment of what valves were required in the Spaceplane PCE was based on the assumption that any propellant carried in the Shuttle payload bay must be separated from its associated combustion device by at least 3 valves in series. For the Spaceplane, this will probably mean one tank cutoff valve and series redundant, bipropellant valve for each PCE module. This arrangement is shown schematically in Figure 38.

The series redundant valve indicated in Figure 38 is currently in development at MOOG. It is very similar, except for its redundancy, to the MOOG valve in use on the current ALRC 100-lb N_2O_4/MMH thruster.

Although valves on every thrust chamber represent the maximum operational flexibility, they also represent the maximum valve weight. This penalty will not be severe however, since each valve weighs approximately 2.5 lbs.

An updated operation and control schematic for the Spaceplane is shown in Figure 39. The schematic illustrates the interaction between the pressurization system, the propellant feed system, and the combustion device. Also included are proposed locations for the various purges, vents, bleeds, and drains necessary for servicing the vehicle.

5.7.1 Component Description and Function

The system pressure source, shown in Figure 39, consists of two (or more) gaseous helium (GHe) bottles connected in parallel, which tie in with a common pressure regulator. Both tanks have individual, direct acting, solenoid valves for redundancy in fill and vent capability. Check valves are included between each GHe bottle and the regulator to insure continued operation in the event of a single bottle failure or leak. Both GHe bottles are instrumented with pressure transducers and thermocouples to monitor temperature and pressure during the bottle pressurization phase and during the bottle blowdown when the engines are firing.

The pressurization system is activated by an electrical signal which opens the oxidizer tank shutoff valve (OTSV) and the fuel tank shutoff valve (FTSV), to supply GHe at regulated pressure to the propellant tanks. Both the OTSV and FTSV are direct-acting solenoid valves. The regulator is an ambient sensing device that is capable of reducing the stored 4000 psig bottle pressure to approximately 500 psig for pressurization of the propellant tanks. The regulator selected for this application should have a wide "flowrate-versus-regulated-presure" bandwidth because of plans to throttle by selective operation of individual PCE thruster pairs.

In the interest of conserving helium mass, thus reducing vehicle weight, the helium will be warmed by a heat exchanger installed between the GHe regulator and the propellant tanks. Since the regulator seeks only to supply a specified pressure to the propellant tanks, less GHe mass will be consumed if the GHe is heated (e.g., Perfect Gas Law...PV = nRT). Experience and analysis show that without the aid of this heat exchanger,

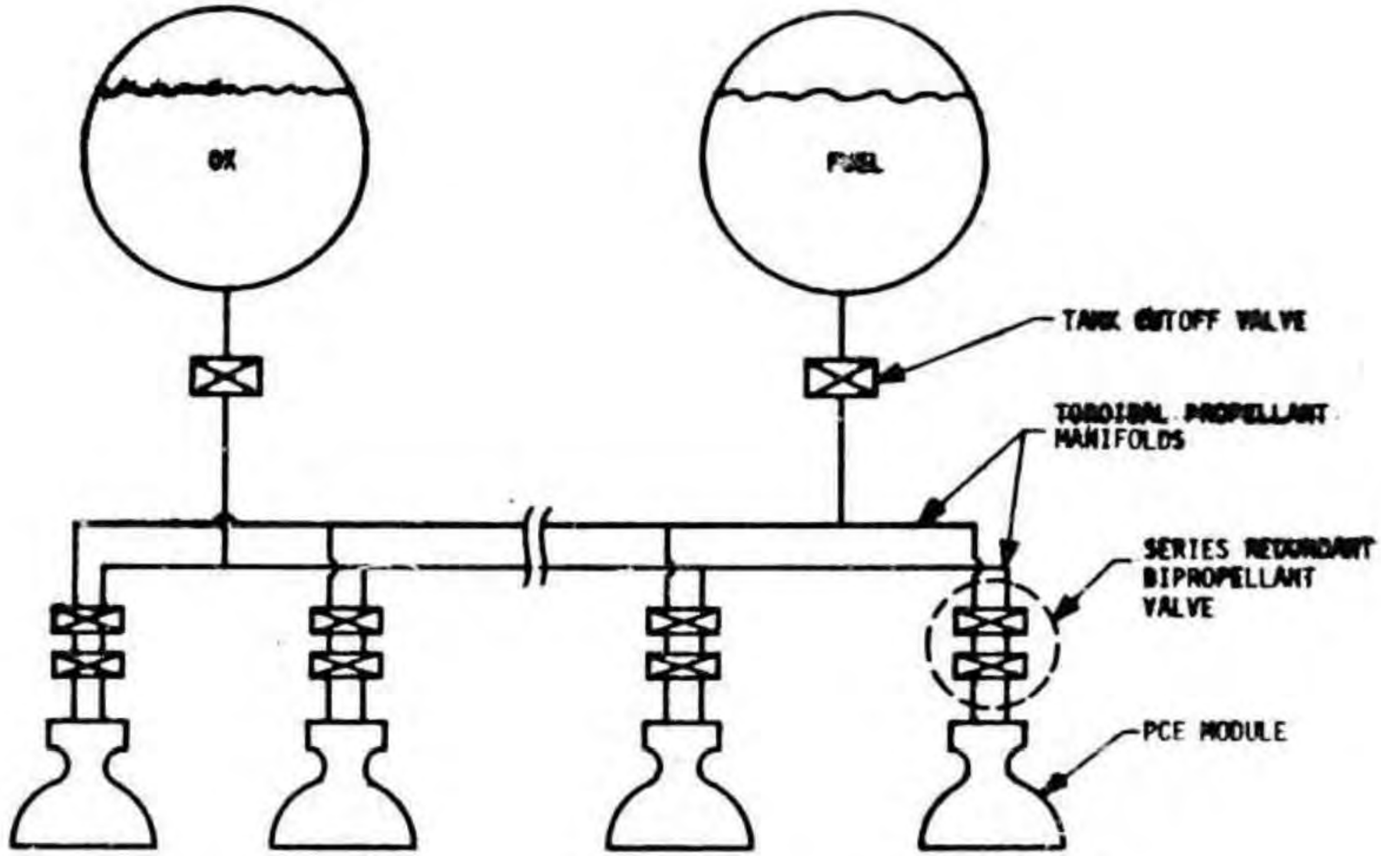


Figure 5-38 Spaceplane Preliminary Plug Cluster Engine Valve Schematic

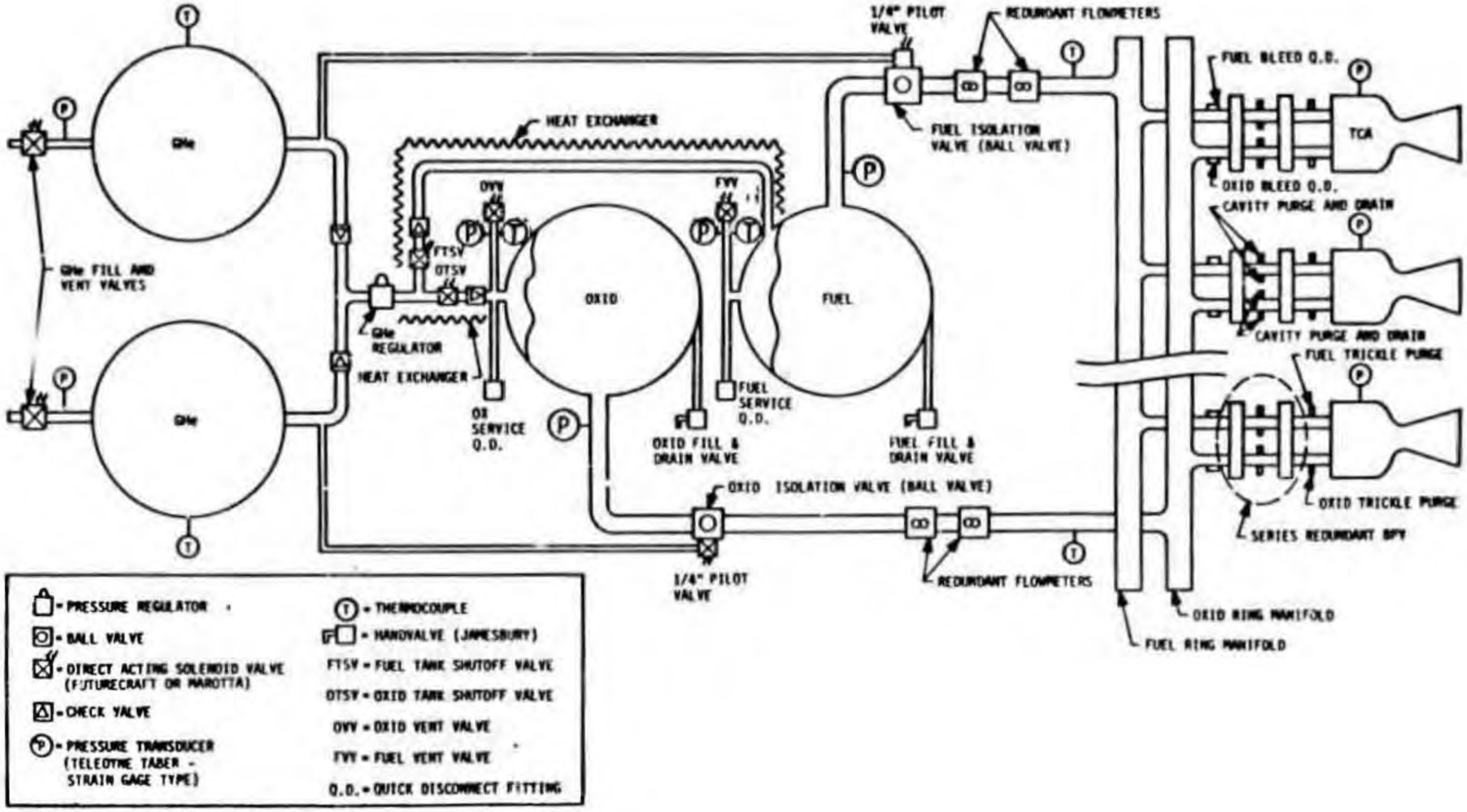


Figure 5-39 Preliminary Operation and Control Schematic

the rapid expansion of the GHe during tank blowdown would result in temperatures as low as -200°F or less. GHe flowrate demand would be excessive at -200°F to maintain the desired regulated pressure. It is anticipated that whether the heat exchanger is driven by the excess APU heat or by heat from the PCE firing, the regulated GHe will not experience temperatures greater than approximately 70°F.

Check valves are located directly downstream of both the OTSV and the FTSV to prevent the inadvertant mixing of residual propellant vapors in the event of leakage through either shutoff valve.

After passing through the check valves, the GHe enters the propellant tanks and maintains a constant regulated pressure during propellant depletion. If the tank design includes a propellant/gas interface, both propellant tanks will be supplied with vent valves (direct acting solenoid), tank overflow quick disconnect fittings (Q.D.'s) and propellant fill and drain valves (handvalve type). The function of these three items will be discussed at length later in this report.

If the tanks contain elastomeric bladders to prevent propellant slosh, no tank overflow Q.D. will be required. Both tanks are also provided with pressure transducers and thermocouples to monitor temperature and pressure during engine operation.

The fuel and oxidizer in the propellant tanks is separated from the engine system by isolation valves. These isolation valves are activated by GHe pressure which is admitted through a pilot valve connected to the helium line upstream of the pressure regulator. The pneumatic boost from the GHe insures that the valve will operate successfully over the wide range of pressures and propellant flowrates encountered in a throttled engine system.

Downstream of the isolation valves, the propellants pass through redundant turbine type flowmeters. Data from these flowmeters is used to determine the engine system mixture ratio (MR) and to confirm that all system hydraulic resistances are correct. In addition, software could be used to compute propellant residuals. An alternative to the flowmeters could be dhe use of liquid level sensors in the tanks. Liquid level versus time correlations have been used successfully to give approximate MR information. After a certain degree of confidence has been obtained in the system, the flowmeters could be deleted, while using only system pressure drops and known hydraulic resistances to calculate MR and flowrate.

The propellant then passes into a toroidal distribution manifold where it is directed toward the several separate PCE inlet lines. Each inlet line is separated from its PCE module by a series-redundant bipropellant valve (BPV). All BPV's will be equipped with the following attach and service points:

1. Propellant Bleed Q.D. - Used to bleed propellant up to the BPV before firing. Eliminates air bubbles and provides firm engine starts.

2. Purge and Drain Between the 2 Valves - Used to drain this cavity during post mission processing.

3. Trickle Purge Port - Used to provide a low flowrate purge downstream of the BPV to preclude injector contamination.

The possibility exists that a single series redundant BPV could be used to provide propellants to a pair of PCE modules. The cost and weight savings to be gained by doubling up like this would have to be balanced against the loss in thrust magnitude flexibility.

The BPV will most likely consist of two torque motor valves or two direct acting solenoid valves connected in series. Flight proven valves of this type are manufactured by companies such as MOOG, Hydraulic Research, and Marquardt.

After passing through the BPV, the propellants are evenly distributed and mixed in the combustion chamber by the injector. Chamber pressure for each thrust chamber is monitored by a pressure transducer.

5.7.2 System Operation

Methods of delivering the Spaceplane to orbit include several potential launch modes. It can be assumed that if the Spaceplane is transported in the Shuttle bay, at least two days will be available for launch preparation while awaiting the lengthy Shuttle countdown. If, on the other hand, a rapid launch is required, the Spaceplane will need to be poised in a state of constant readiness. The following paragraphs will address the operational procedures to be used for the rapid launch case.

Months Prior to Launch:

1. Open propellant tank fill and drain valves and connect the service Q.D.'s to a catc tank reservoir.

2. Fill the tanks until the service line overflows with no bubbles. Close the fill and drain valves.

3. Open the OVV and FVV and blow the service line clean with GN2. Cap off service Q.D.'s.

4. After checking propellant temperature, open the fill and drain valve and draw the proper tank ullage (if desired). Close fill and drain valve.

5. Pressurize the GHe spheres to approx. 30 psig.

6. Open the OTSV, FTSV, fuel isolation valve, and the oxidizer isolation valves simultaneously. Bleed propellants through BPV bleed Q.D.'s until air bubbles disappear.

7. Close the OTSV, FTSV, and the two isolation valves. Cap off BPV bleed Q.D.'s.

Ten Minutes Prior to Launch:

1. Pressurize GHe spheres to 4000 psia. Heat exchangers will be necessary to cool the GHe during a rapid fill such as this.

Ignition:

1. Energize the OTSV, FTSV, both isolation valves, and the BPV's simultaneously. The PCE inlet lines and injector manifolds will be sized such that the oxidizer will enjoy a 50 millisecond lead on the fuel, thus precluding chamber explosions or rough starts. Selection of engine pairs to be fired will be controlled by keyboard input and software for large ΔV requirements and by pilot hand controls for fine tuning.

Shutdown:

1. De-energize the OTSV, FTSV, both isolation valves and [illegible] simultaneously.

Subsequent Firings:

1. Same valve sequence as above.
2. GHe tank and propellant tank pressure and temperature must be monitored between firings. OVV or FVV may be energized to vent any excess propellant tank pressure obtained through thermal soakback.

5.7.3 Posttest Processing

1. Vent the GHe spheres to ambient pressure through their respective fill and vent valves.
2. Open the OTSV, FTSV, OVV, FVV, and both isolation valves simultaneously and vent the propellant tanks and lines to ambient pressure. Close same six valves.
3. Purge and drain propellants in cavity between series redundant BPV. Propellants are left in the tanks and lines between the tank and the upstream BPV. This procedure has been followed safely for 6 months on the Space Shuttle OMS engine.
4. Connect GN2 purge to trickle purge ports downstream of the BPV.
5. Reload propellants per the procedure already described.

The Spaceplane safety evaluation (see Task 3.6, Safety Issues) revealed some alternate controls configurations that should be noted here.

One series bipropellant per opposing pair of PCE modules should be adequate from both a safety and controls standpoint. If a PCE module malfunctions, its opposing module will have to be shutdown anyway. For this reason, one bipropellant valve to an opposing pair would serve this function. The disadvantage is the loss in flexibility due to the resulting inability to operate either of the paired PCE modules separately.

5.8 TASK 2.4 REACTION CONTROL SYSTEM (RCS) DEFINITION

The primary objective of this task was to define the thrust level and location of RCS thrusters to provide vehicle roll, yaw and pitch on the basis of moment requirements to be supplied by SNLA. Ultimately, the actual roll, pitch and yaw rates were established on the basis of discussions with Lincom of Houston, Texas, regarding the RCS requirements of manned space vehicles in general. Lincom specializes in spacecraft rendezvous and docking procedures. Rendezvous and docking capability was considered a Spaceplane onboard propulsion system requirement. The actual moment data from SNLA was delayed primarily because of wind tunnel scheduling problems. This occurred because much of the wind tunnel testing necessary to define vehicle moment requirements was not performed in SNLA's own facilities.

Some of the general considerations involved in the RCS definition are noted first:

- Based on discussions with Honeywell and Lincom, it is assumed that any RCS will be based on the use of pulsing thrusters regardless of RCS module thrust level. Most importantly, this method accommodates relatively large vehicle c.g. location changes with little difficulty.

- A high-thrust (i.e., 300 lbF or higher) module located in the nose opening to provide retro thrust may be necessary.

- Placement of small pitch and yaw thrusters in the nose will probably be necessary.

- Placement of RCS thrusters in the nose opening would eliminate openings and flap doors in the Spaceplane vehicle skin.

- The preliminary design of the RCS or ACS was impacted considerably by the requirement for pure translation capability in addition to pitch, yaw and roll requirements.

- An RCS system using dedicated thrusters (i.e., thrusters provide only pitch, yaw or roll) in addition to a dedicated maneuvering system (i.e., thrusters provide only vehicle translation) could require more than 30 separate RCS type engines.

- The 5 lbF module is easily scalable, primarily through photo enlarging of injector platelets, to higher thrust levels. Additionally, the required film cooling is less, and hence performance is improved, at higher thrusts.

- An important influence on the design of the RCS is the potential requirement for redundancy. The Space Shuttle orbiter has a completely redundant RCS. Although this requirement probably is not realistic for the Spaceplane, a partially redundant RCS may be required. A partially redundant RCS could sustain the loss of several RCS thrusters and still perform the basic maneuvers required for a safe re-entry or return to the Shuttle orbiter.

- A potential problem is the effect of the RCS plume impingement on the Spaceplane pilot. This potential problem is based on the assumption that the pilot will be at least partially outside of the Spaceplane conical envelope while RCS thrusters are firing.

To obtain a rough idea of impulse required to maneuver the Spaceplane, a preliminary calculation of total impulse requirements for making a 180° turnaround maneuver was accomplished. These calculations were based on predicted Spaceplane vehicle rotational inertia values of 156,000 lb-ft^2 (pitch and yaw) and 10,500 lbft^2 (roll). These values were also corroborated with both SNLA and H/S. Table V shows estimates of Spaceplane total impulse vs. maneuver time. The basis for the estimates is that application of 1000 in.-lbF torque results in an angular acceleration of 1 degree/sec^2. The total burn time shown is the sum of initial acceleration and final deceleration burns. The total impulse is approximately 400,000 lbF-sec.

TABLE V
SPACEPLANE TOTAL IMPULSE VS. MANEUVER TIME

Burn Time (sec)	Maneuver Time (sec)	Total Impulse (lbF-sec)	w_{max} (degrees/sec)
26.8	26.8	268	13.4
20.0	28.0	200	10.0
10.0	41.0	100	5.0
5.0	74.5	50	2.5
2.0	181.0	20	1.0
1.0	360.5	10	0.5

At least three RCS configurations were considered before the final RCS configuration was defined. These three configurations, and their associated benefits and deficiencies, are briefly described below.

5.8.1 Integral PCE/RCS

In order to reduce the total number of individual thrusters aboard the Spaceplane, an integral main propulsion/RCS/vehicle translation system was configured. A preliminary design is shown in Figure 40. The important points to be made regarding this system are noted here (refer to the figure for identification of the thrusters discussed below).

- The PCE provides +x (axial) ΔV (only two thrusters, No. 3 and No. 4, of the cluster are shown).

- Thrusters No. 2 and No. 4 together provide positive vehicle yaw (two corresponding thrusters in the perpendicular plane provide positive vehicle pitch).

- Thrusters No. 2 and No. 3 together provide +y vehicle translation (two corresponding thrusters in the perpendicular plane provide +z translation).

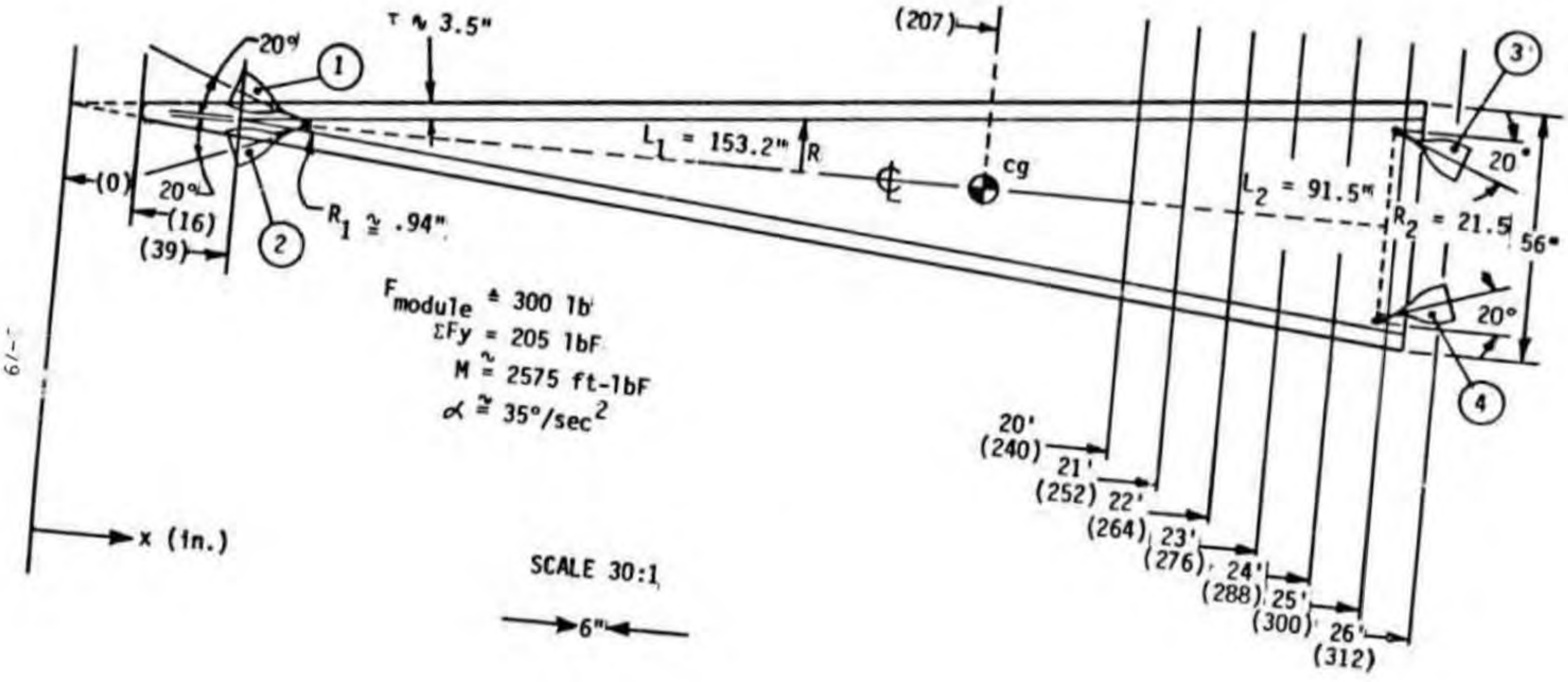


Figure 5-40 Preliminary Main Engine/ACS/Translation System Design Integral Plug Cluster Engine/RCS

- o Thrusters No. 1 and No. 4 together provide -y vehicle translation (two corresponding thrusters in the perpendicular plane provide -z translation).
- o Thrusters No. 1 and No. 2 together and/or two thrusters in the perpendicular plane provide -x (retro) ΔV.

The resulting torque for each of the thruster pairs for the pitch and yaw maneuvers is approximately 2575 ft-lb. The net $\pm$ y (or $\pm$ z) translation thrust is approximately 205 lb. The net retro (-x) thrust is 564 lb (2 modules) or 1128 lb (4 modules). The translating thrust levels result in vehicle g levels of approximately .033 to .043. The pitch and yaw engines would provide a vehicle angular acceleration of approximately $35^{o}/sec^{2}$. The forward thrusters should be, and can be, moved aft, hopefully to some point aft of the contemplated nose hinge. However, to provide side-to-side translation, the thrust level of these modules must be equal to or greater than that of the PCE modules. This preliminary design was based on 300 lbF/module but is adaptable to PCE systems using 4, 8, 12, 16, 20, etc. modules.

Roll engines were not positioned. Unlike the other modules in the integral main propulsion/RCS/translation design shown, these thrusters will provide only the roll maneuver. The reduction of this number to only 4 (plus whatever roll engines are required) was a considerable reduction in vehicle complexity and hence improvement in reliability.

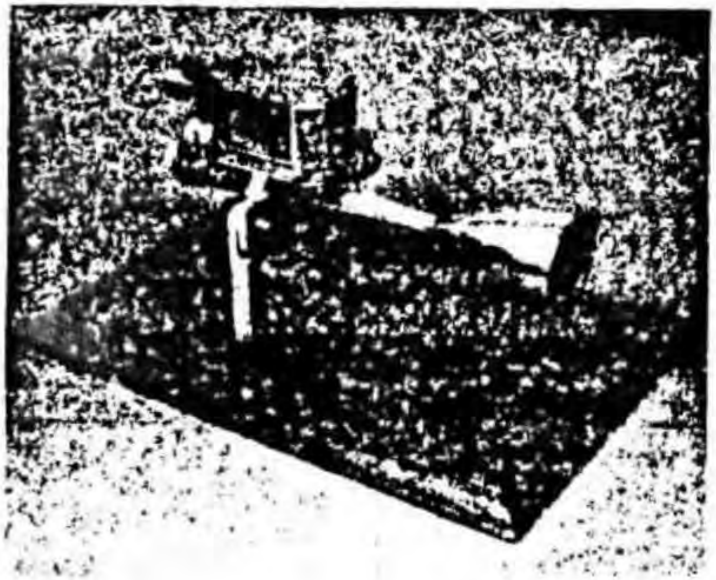
Subsequent discussions with both Honeywell and Lancom indicated that the vehicle rotational and translational acceleration rates inherent in this system were entirely excessive. On the basis of these discussions, it was determined that thrusters with thrust levels different from that of the PCE module will be required. ALRC has already developed RCS thrusters that deliver from 1 to 5 lb. of thrust. These, and possibly the ALRC 0.5 lbF thruster, will be required to perform the RCS vernier requirements. This type of application normally requires a fast engine response and the ability to perform rapid pulsing. Both of these ALRC engines are high response pulsing engines and are therefore ideal for use in the Spaceplane RCS.

5.8.2 5 LBF Thruster RCS

This Reaction Control Subsystem (RCS) definition was peformed on the basis of (1) the updated RCS requirements tabulated in the Task 1.1 (Requirements Definition) description and (2) the sole use of ALRC's existing 5 lbF N_2O_4/MMH thruster. These revised requirements were made on the basis of results of the January TI/TD meeting and discussions with Lincom. These RCS requirements are comparable to RCS performance of both Apollo and the Space Shuttle orbiter. A photograph and description of this thruster is shown in Figure 41.

The RCS based on the use of 5 lbF modules is shown schematically in Figure 42. The ideal performance of this system is outlined in Table VI below.

A comparison of this 5 lbF based RCS to the RCS requirements, listed in the Task 1.1 results description, shows that it is somewhat slower than



PERFORMANCE CAPABILITY

Parameter	Nominal	Demonstrated Operational Range
Propellants	N_2O_4/MMH	N_2O_4 and MON_3
Thrust (lbf)	5.0	2.1 - 5.7
Isp Steady State (sec)	291	279 - 297
Single Burn Duration (sec)	Unlimited	20,000
Valve Inlet Pressure (psia)	300	400 to 100 (blowdown 4:1)
Propellant Inlet Temperatures (°F)	70	20 - 120
Mixture Ratio	1.65	1.2 to 1.8
Isp for 0.05 lb-sec MIB (sec)	229	
MIB Demonstrated (Feed 300 psia) (lbf-sec) (Feed 220 psia)	0.025 0.003	0.003 (Cold Wall 220 psia feed. EPW .003 sec. 10 pulse average)
Maximum Number of Starts	300,000	+500,000 (similar valve has 10^6 cycles wet and dry)
Expansion Ratio	150:1	50, 100 and 150
Cumulative Firing Life (hrs)	+100	0.33 Design estimate of +100 thermally. TBD cooling life
Heat Load to Satellite (watts)	<20 (<10)	(Duty cycle, insulation thickness and thermal standoff dependent)
Mission Life (yrs)	10	TBD
Valve Voltage (vdc)	28 +4 -0	6 to 36
Flightweight TCA Reliability	Better than 0.99	Achieved by Analysis
Envelope (in.)	7.48 x 2.83 dia	
Engine Weight (Including insulation, lbm)	1.5 (150:1)	
Time to 90% Pc, (sec)	0.006	
Valve Data (sec)		
Minimum EPW	0.003	0.003 sec to 8 hrs
Response Time to Full Open	0.003	
Response Time to Full Close	0.003	
Valve Stab (psia)	+1000	(28 vdc)

Figure 5-41, AJ10-181-1 Five-Pound Thrust Bipropellant RCS Engine

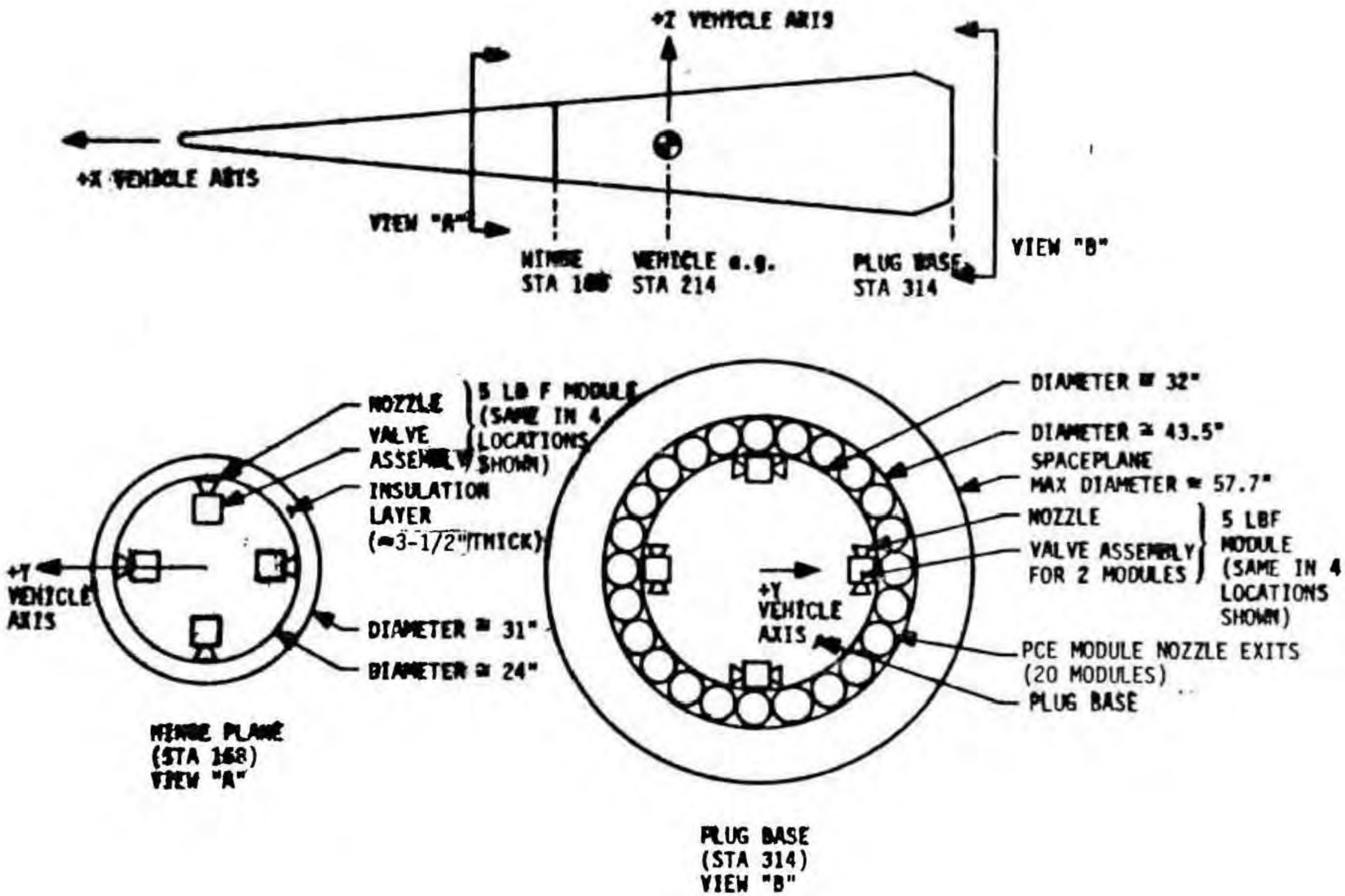


Figure 5-42 5 LBF Module RCS

TABLE VI

5 lbF MODULE RCS PERFORMANCE

Function	Angular Acceleration (°/sec^2)	Time to Reach Angular Velocity of 2.0°/sec (sec)	Linear Acceleration (ft/sec^2)	Time to Obtain Spaceplane ΔV = 0.02 ft/sec (sec)
Pitch and Yaw (Y and Z axes)	0.91	2.2	-	-
Roll (X Axis)	2.00	0.98	-	-
Retro Thrust (-X Axis)	-	-	0.30	0.065
2 Axes Translation (Y and X Axes)	-	-	0.075	0.270

desired. This is particularly true for the pitch and yaw maneuvers. The roll maneuver response is adequate as is the 2 axis translation.

An equivalent arrangement for the plug base arrangement requiring only two RCS clusters is shown in Figure 43. It should be noted that the retro thrust maneuver would be provided by two 100 lbF modules positioned between opposing pairs of PCE modules. This concept is illustrated in Figure 44.

The placement of these retro engines in the rear of the vehicle eliminates the need for placement of additional axial modules in the hinge plane (approximate station 168) of the vehicle.

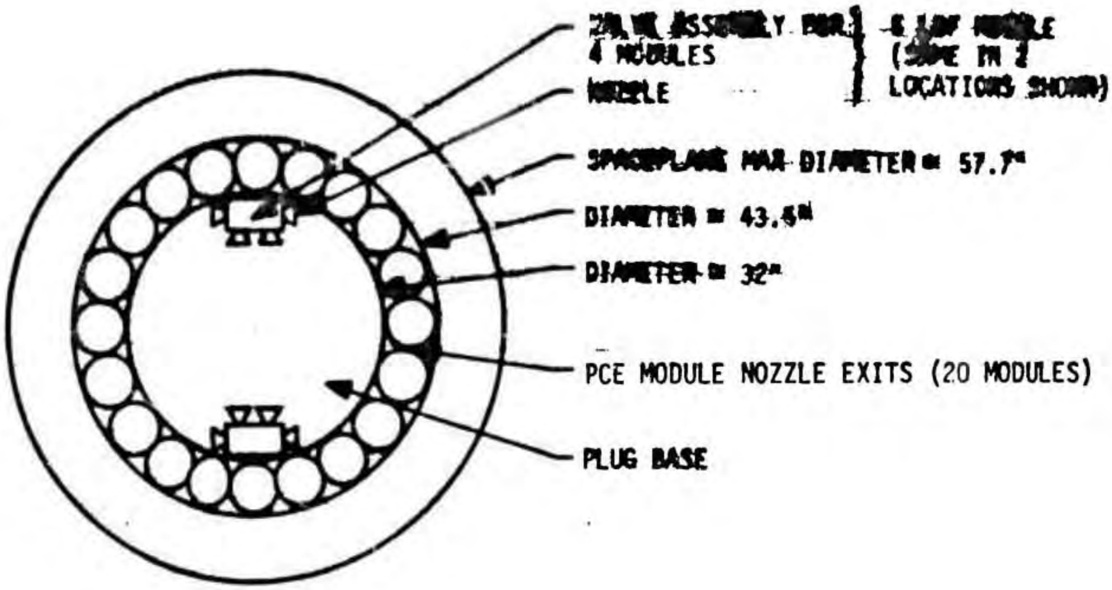
5.8.3 100 LBF Thruster RCS

This RCS was based on the sole use of ALRC's existing 100 lbF N_2O_4/MMH thruster. A photograph and description of this thruster is shown in Figure 45. The arrangement of these engines is identical to that shown for the 5 lbF module RCS (see Figure 41). The major difference between these two systems, besides the obvious module thrust, is the fact that the 100 lbF module nozzles are truncated, giving a reduction (10 to 20%) in performance and thrust level, to occupy an envelope approximately equal to that of the 5 lbF module. The resulting difference then between the two systems is primarily performance rather than weight or envelope. The performance of the 100 lbF module based RCS is listed in Table VII.

As Table VII shows, the potential difficulty with this system is excessive acceleration rates. However, the use of very short burn times should lessen the overall impact of these accelerations on both the pilot and vehicle.

5.8.4 15 LBF Thruster RCS

Finally, in an effort to incorporate both 5 lbF and 100 lbF thrusters, a 15 lbF thruster RCS was configured. This system is described in Table VIII below.



PLUG BASE
(STA 314)
VIEW "B" (SEE FIGURE 13)

Figure 5-43 5 LBF Module RCS Alternative (Aft View)

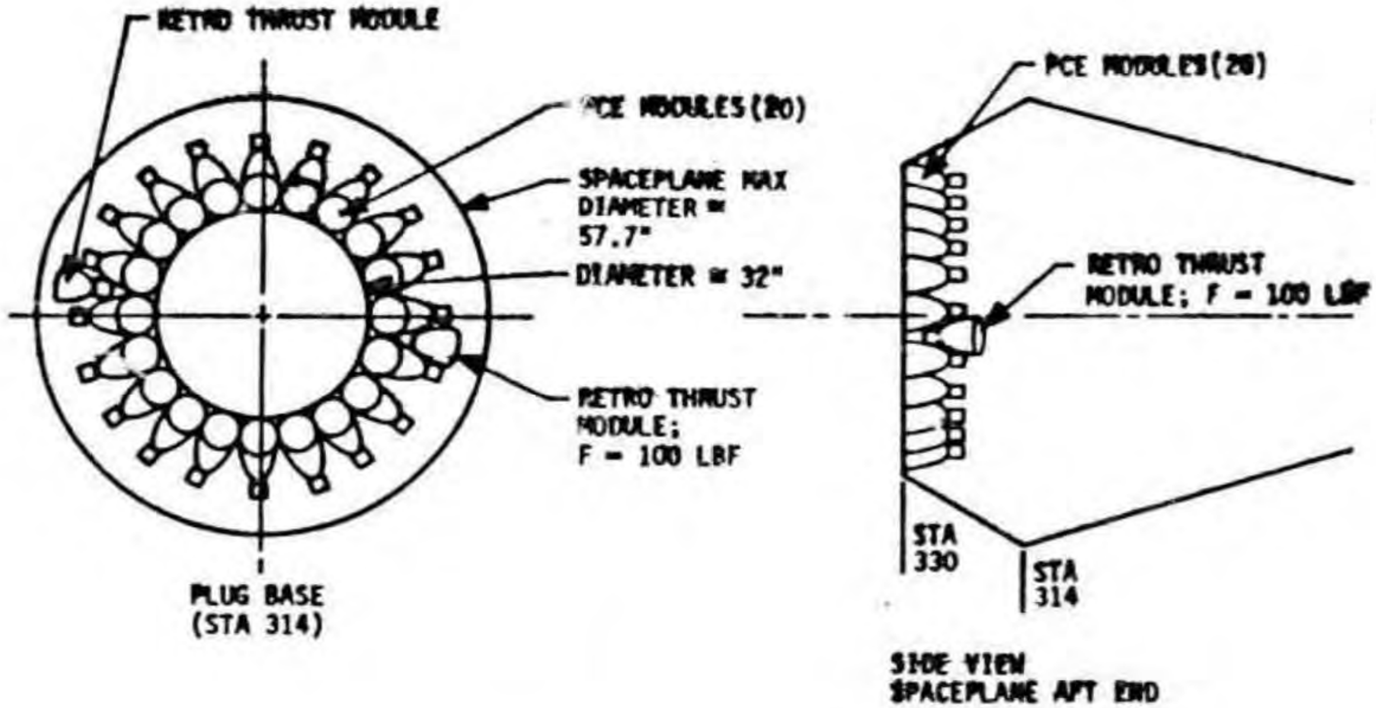
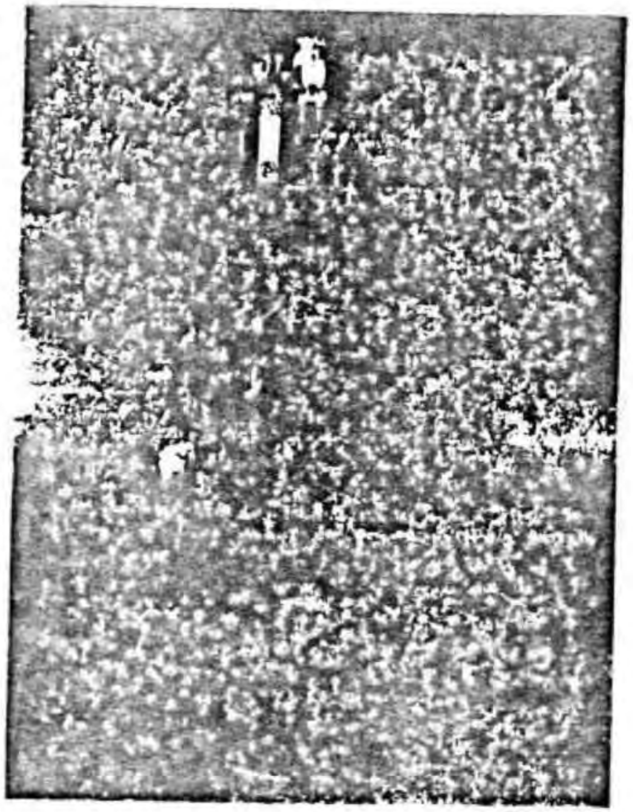


Figure 5-44 Retro Thrust Module Location



PERFORMANCE CAPABILITY

Parameters	Nominal	Demonstrated Operational Range
Propellants	N_2O_4/MMH	N_2O_4/MON_3
Thrust, (lbf)	100	116 - 70
Isp Steady State (sec) Pressure Regulated	309*	308 - 311
Single Burn Duration (sec)	11,500	1591, facility limited**
Valve Inlet Pressure (psia)	340	430-185 2.3 to 1 blowdown range
Propellant Inlet Temperature (°F)	70	40 - 110
Mixture Ratio	1.65	1.40 to 1.83
Maximum Number of Starts	1500	+1700
Expansion Ratio	150:1	150:1
Minimum Impulse Bit (lbf-sec)	< 3.0	±5% 1 σ
Cumulative Firing Life (sec)	11,500	+11,500**
Mission Life (yrs)	10	(Same valve design, has demonstrated a 3 yr. wet life in space)
Valve Voltage (vdc)	28 ±4	
Flightweight Reliability	Better than 0.99	
Engine Weight (lbm)	4.1 (ϵ = 150:1)	

*Engine delivers 310 sec at 85 lbf thrust, MR 1.65, and valve inlet of 235 psia.

**Max single demonstrated burn duration at altitude of 26 min 31 sec, limited only by test facility capability.

Figure 5-45 AJ10-210 One-Hundred Pound Thrust Bipropellant RCS Engine

TABLE VII

100 lbF MODULE RCS PERFORMANCE

Function	Angular Acceleration (°/sec^2)	Time to Reach Angular Velocity of 2.0°/sec (sec)	Linear Acceleration (ft/sec^2)	Time to Obtain Spaceplane ΔV = 0.02 ft/sec (sec)
Pitch and Yaw (Y and Z axes)	14.7	0.136	-	-
Roll (X axes)	37.7	0.053	-	-
Retro Thrust (-X axes)	-	-	0.30	0.065
2 Axes Translation (Y and Z Axes)	-	-	0.99	0.029

TABLE VIII
N_2O_4/MMH RCS DESCRIPTION

Total number of RCS modules	16
RCS module thrust level, lbF	15
RCS module chamber pressure, psia	142
RCS propellants	N_2O_4/MMH
RCS module propellant mixture ratio	1.65
Total RCS weight, lbM	45.0

A nearly equivalent RCS based on the use of N2O4/PAAB-1 is outlined in Table IX below.

TABLE IX
N_2O_4/PAAB-1 RCS

Total number of RCS modules	16
RCS module thrust level, lbF	15
RCS module chamber pressure, psia	100
RCS propellants	N_2O_4/PAAB-1
RCS module propellant mixture ratio	1.20
Total RCS weight, lbM	45.0

Figures 46 through 49 show the location and thrust magnitude of all of the N_2O_4/PAAB-1 RCS thrusters, including the two aft mounted retro thrusters (188 lbF each). Because the N_2O_4/PAAB-1 RCS operates at a slightly lower chamber pressure (100 psia) compared to the N_2O_4/MMH RCS (142 psia), its performance will be somewhat lower, assuming a fixed envelope for the RCS thruster nozzles. Because the total RCS propellant mass requirement is relatively l this difference in RCS Isp's will have a negligible effect on the SP du to either RCS.

The N_2O_4/PAAB-1 RCS capability is outlined in Table X. This table has been formatted to allow a direct comparison to the RCS requirement described previously in the Task 1.1 Propulsion Requirements Definition.

These capabilities are comparable to those of the RCS systems used on other manned spacecraft. This comparison is illustrated in Figure 50, a plot of angular (rotational) vs. linear (translational) acceleration levels for existing (Shuttle) and past manned spacecraft. The Spaceplane RCS envelope (except for the +X axis translational acceleration which is not really part of the RCS requirement) has been superimposed on this plot. This figure was supplied to ALRC courtesy of Lincom.

An RCS module thrust level of 15 lbF represents a very modest modification of either ALRC's 5 lbF or 100 lbF bipropellant RCS thrusters.

A preliminary study of the potential RCS plume/pilot impingement problem indicates that the 15 lbF RCS thruster plume will not be hazardous to the pilot (this despite the fact that the pilot may occasionally extend somewhat beyond the Spaceplane conical envelope during a mission).

Finally, the question of RCS redundancy should be addressed. A completely redundant RCS will be heavy and require numberous propellant lines. On the other hand, at least some redundancy is recommended for a man-rated system.

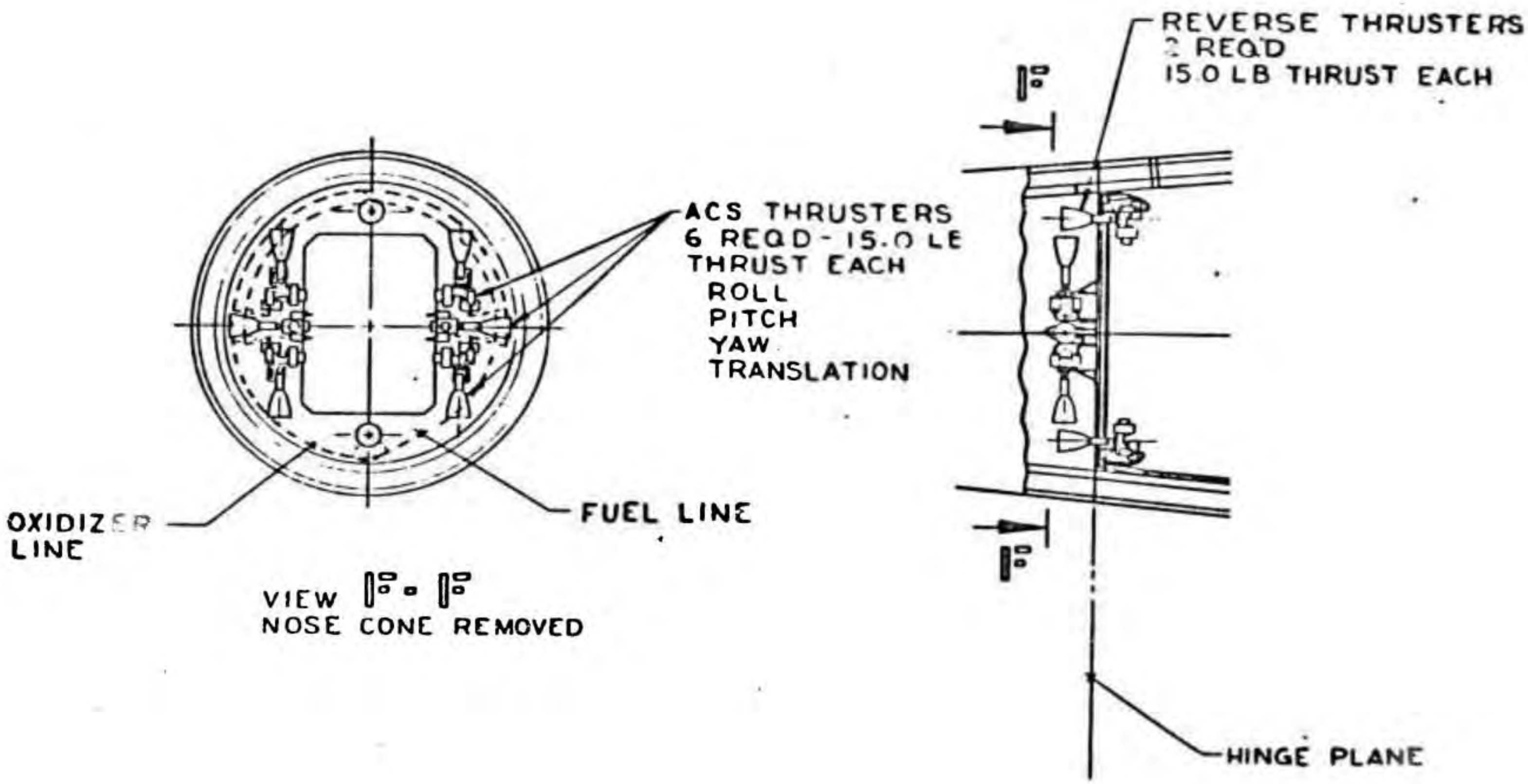


Figure 5-46 **N_2O_4/PAAB-1 Foreward RCS Thrusters**

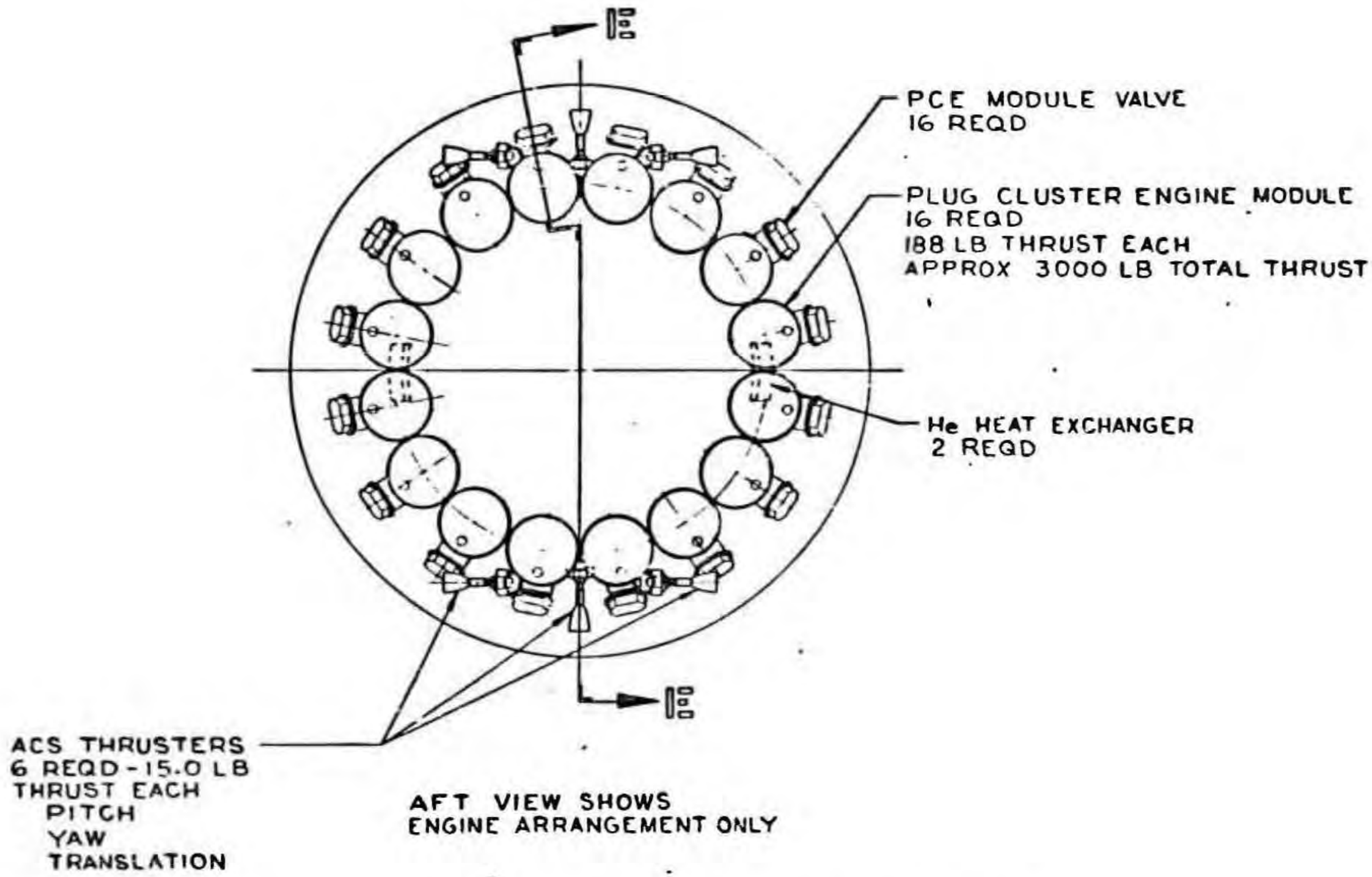


Figure 5-47 N_2O_4/PAAB-1 Plug Cluster Engine (Rear View)

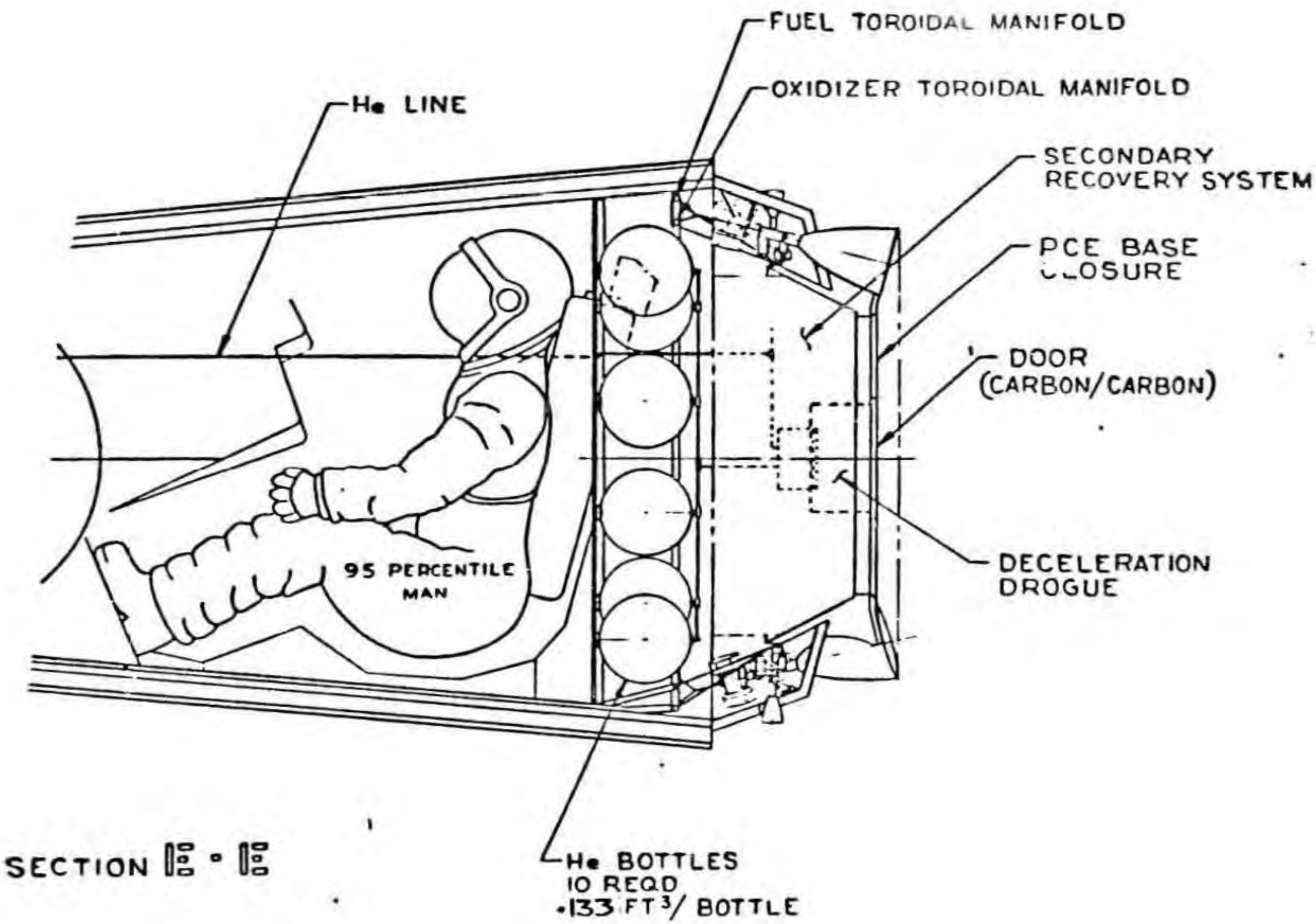


Figure 5-48 N_2O_4/PAAB-1 Plug Cluster Engine (Side View)

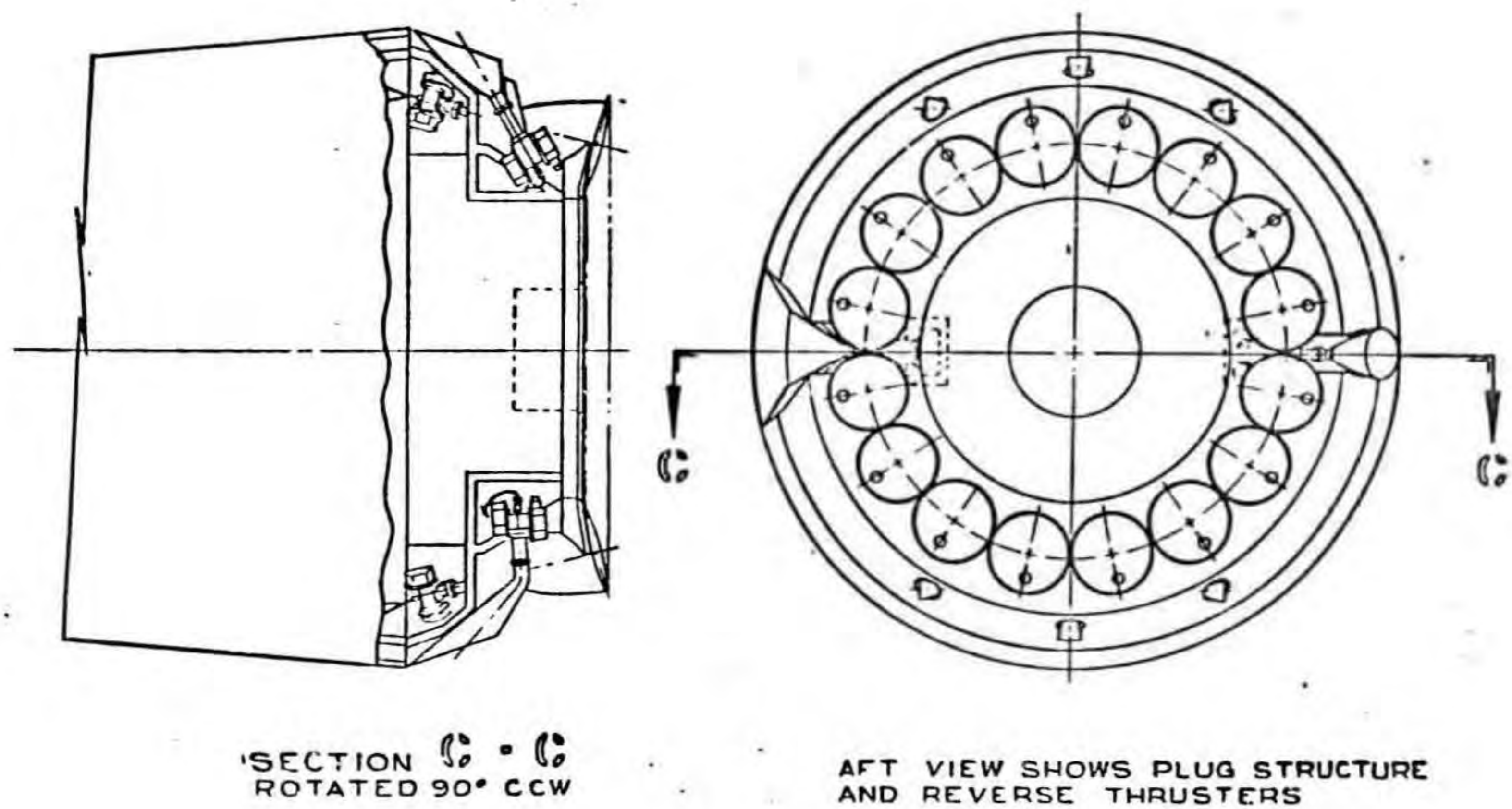


Figure 5-49 **N_2O_4/PAAB-1 Retro Thrusters**

TABLE X

SPACEPLANE RCS CAPABILITY

Function	Maximum Acceleration Angular (°/sec²)	Maximum Acceleration Linear (ft/sec²)	Max time Required to give Vehicle $\Delta\nu$ = 0.02 ft/sec (sec)
Pitch	2.72	-	-
Yaw	1.36	-	-
Roll	3.98	-	-
Retro Thrust (-X Axis)	-	0.154	0.13
Forward Thrust (+X Axis)	-	2.014	0.01
Translation (±Y Axes)	-	0.112	0.178
Translation (±Z Axes)	-	0.225	0.089

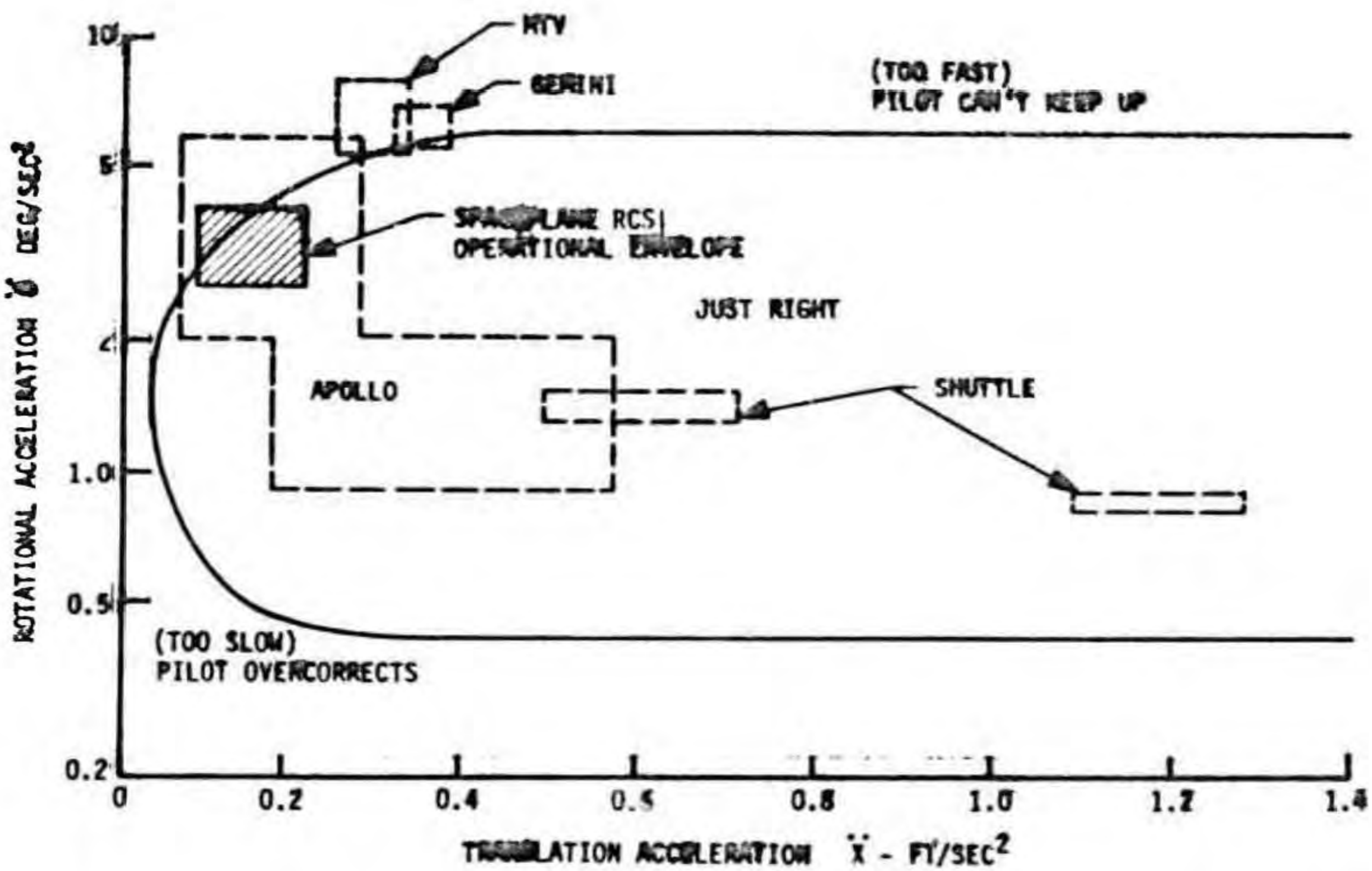


Figure 5-50 Stability and Control RCS Thruster Sizing (Man-In-Loop Docking/Handling Qualities)

5.9 TASK 3.1 PROPULSION SYSTEM INSTALLATION

This task had three specific subtasks, noted here: (1) update the selected propulsion system design per vehicle requirements supplied by SNLA, (2) specify RCS nozzle configurations, and (3) define all propulsion system thrust load paths.

5.9.1 Propulsion System Design

Drawings incorporating the results of the PCE optimization performed in Task 3.8 (Selected Design) were generated for both the N_2O_4/PAAB-1 and N_2O_4/MMH onboard propulsion systems. The engine operating specs, and the justification for selection of these configurations, are found in the Task 3.8 discussion.

The N_2O_4/PAAB-1 onboard propulsion system is shown in Figures 51 through 55, which were taken directly from the latest version of ALRC Drawing No. 1195445. Figure 51 shows the Spaceplane aft end. Both the PCE module nozzles and six aft mounted RCS thrusters are visible. Figure 52 shows the Spaceplane aft end from the side. Although the He bottle arrangement is arbitrary, the total He bottle volume is not. The final arrangement of the He bottles, as determined by H/S, was shown previously in the H/S Spaceplane vehicle layout drawing, Figure 37. Figure 53 shows the two aft mounted retro thrusters, each of 188 lbF. Two different nozzle configurations are shown, although in practice only one would be used. This figure serves to show that the room available for the retro thrust nozzles and valve assemblies is very limited. Figure 54 shows ALRC's spherical propellant tank design and installation. The volume of the PAAB-1 tank does not reflect the requirement for additional PAAB-1 to operate the Spaceplane APU's. An enlarged PAAB-1 tank having this extra capacity was also shown previously in Figure 37. Figure 55 shows both the front and side view of the eight foreward mounted RCS thrusters.

Figures 56 through 59 show a corresponding series of drawings which document the N_2O_4/MMH onboard propulsion system. The two retro thrusters, which would be identical to those in the N_2O_4/PAAB-1 system, are not shown.

Some of the revisions to the preliminary design selected in Task 2.1, as well as some of the subtle differences between the N_2O_4/PAAB-1 and N_2O_4/MMH systems, are described here.

- Two retro-thrust engines were added to the Spaceplane aft end.

- The N_2O_4/PAAB-1 PCE will have approximately the same envelope (i.e., PCE module length and PCE diameter) as the current N_2O_4/MMH PCE.

- The N_2O_4/PAAB-1 PCE module throat radius will be somewhat larger, since the chamber pressure is slightly lower (100 psia vs. 142 psia) than the N_2O_4/MMH PCE.

- The N_2O_4/PAAB-1 propellant tanks, although containing equal total propellant weight, are slightly different in size because of the density difference of PAAB-1 (61.39 lbM/ft^3) compared to MMH (54.85 lbM/ft^3) and because of a lower mixture ratio (1.2 for N_2O_4/PAAB-1 vs. 1.65 for N_2O_4/MMH).

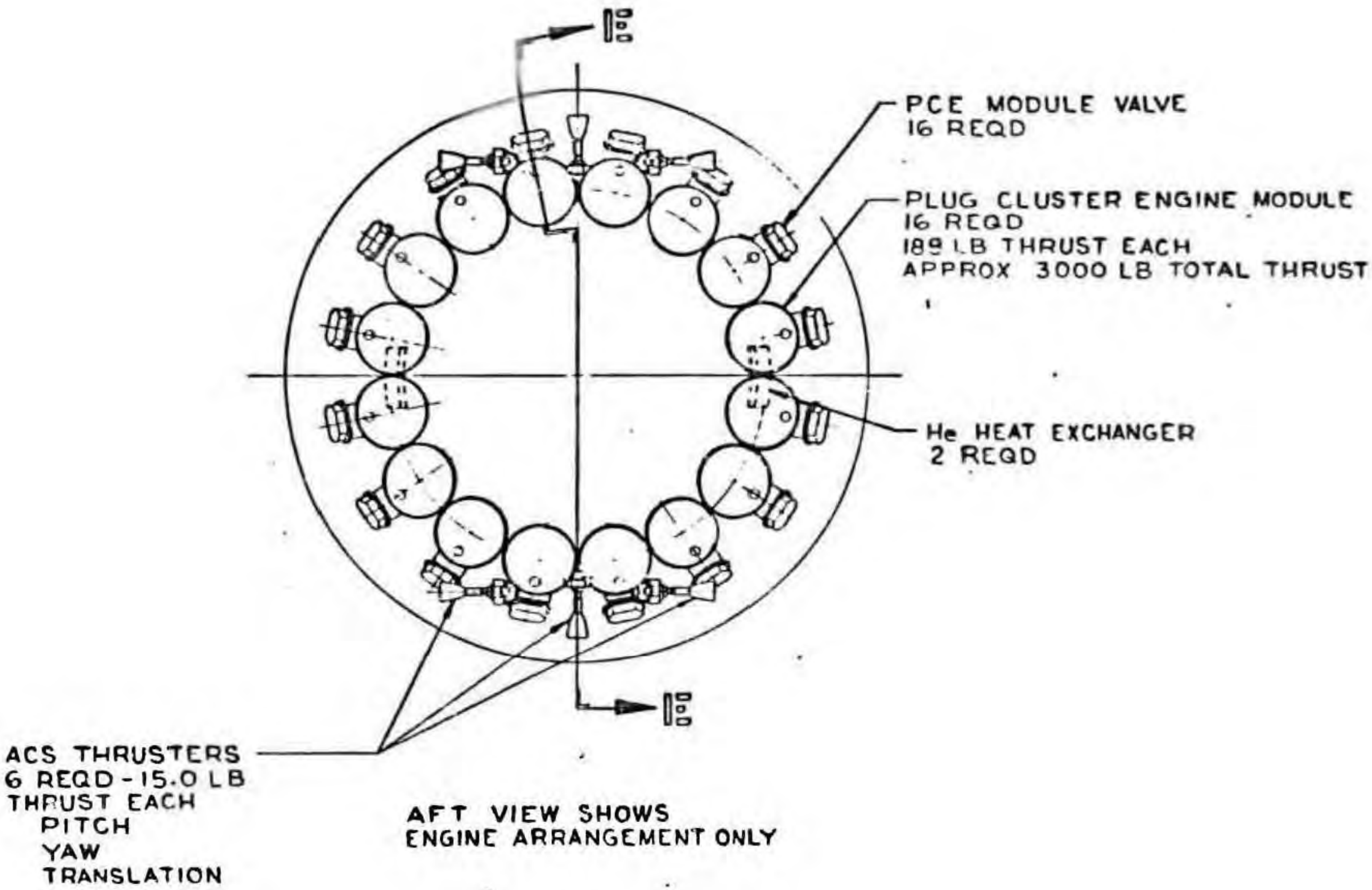


Figure 5-51 N_2O_4/PAAB-1 Plug Cluster Engine (Rear View)

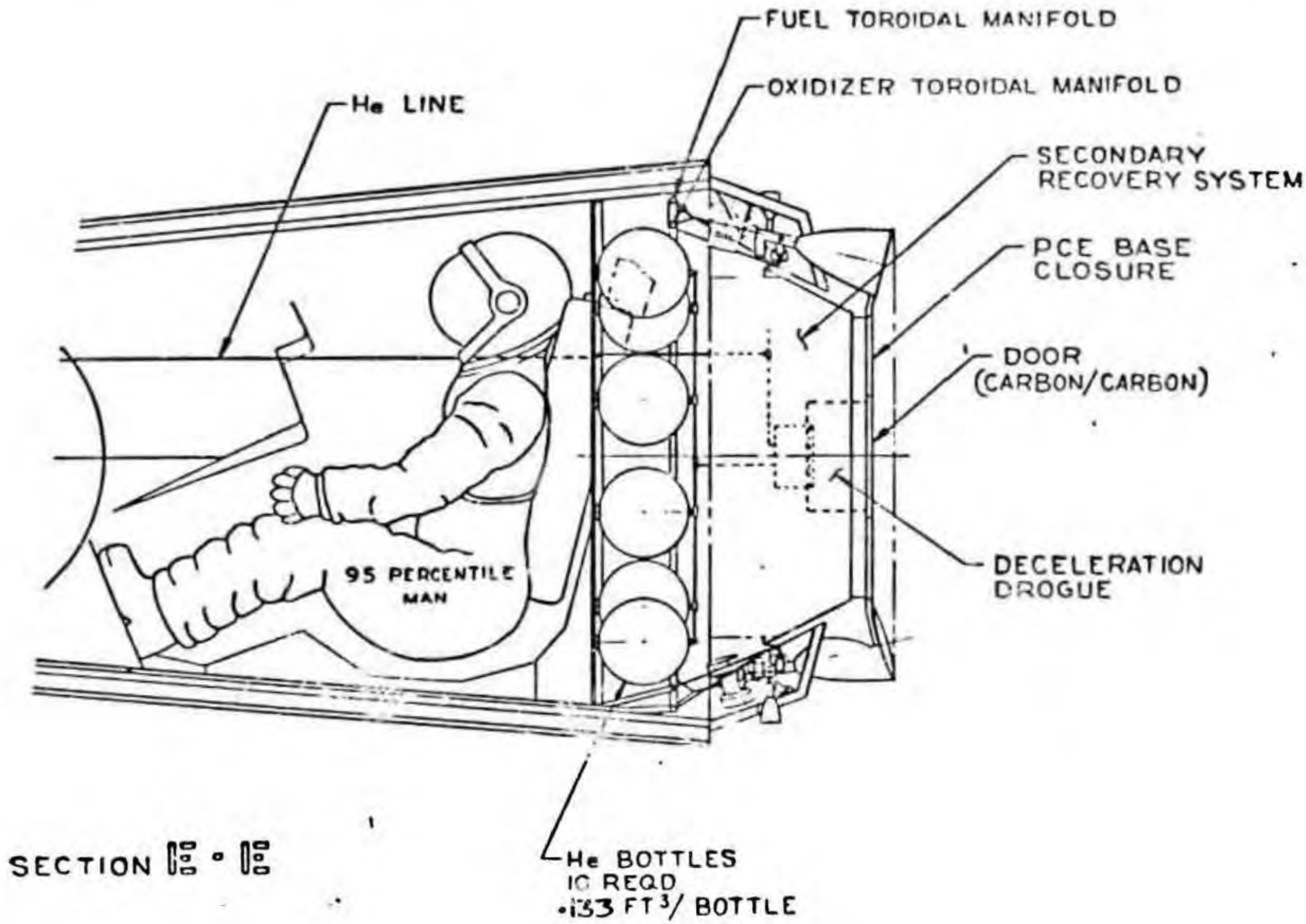
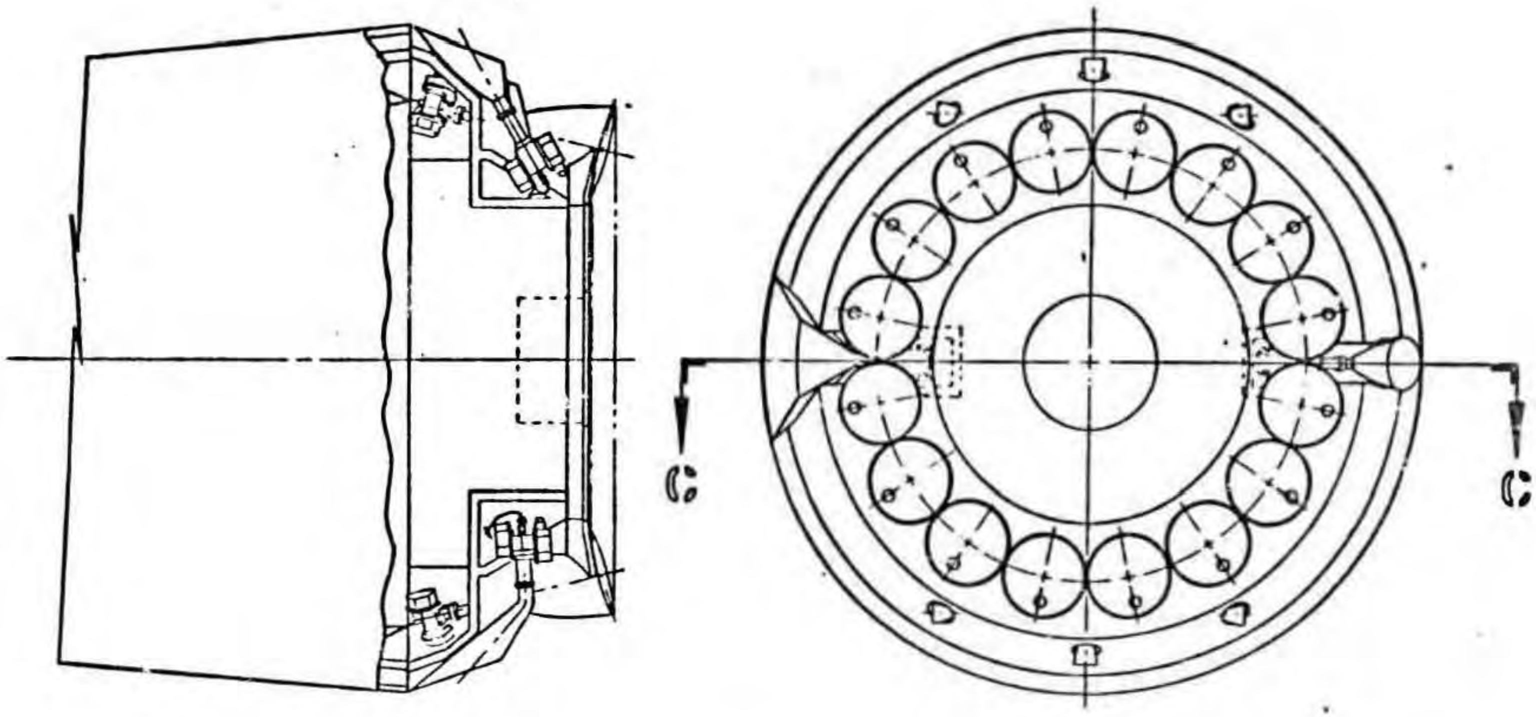


Figure 5-52 N_2O_4/PAAB-1 Plug Cluster Engine (Side View)



SECTION C - C
ROTATED 90° CCW

AFT VIEW SHOWS PLUG STRUCTURE
AND REVERSE THRUSTERS

Figure 5-53 N_2O_4/PAAB-1 Retro Thrusters

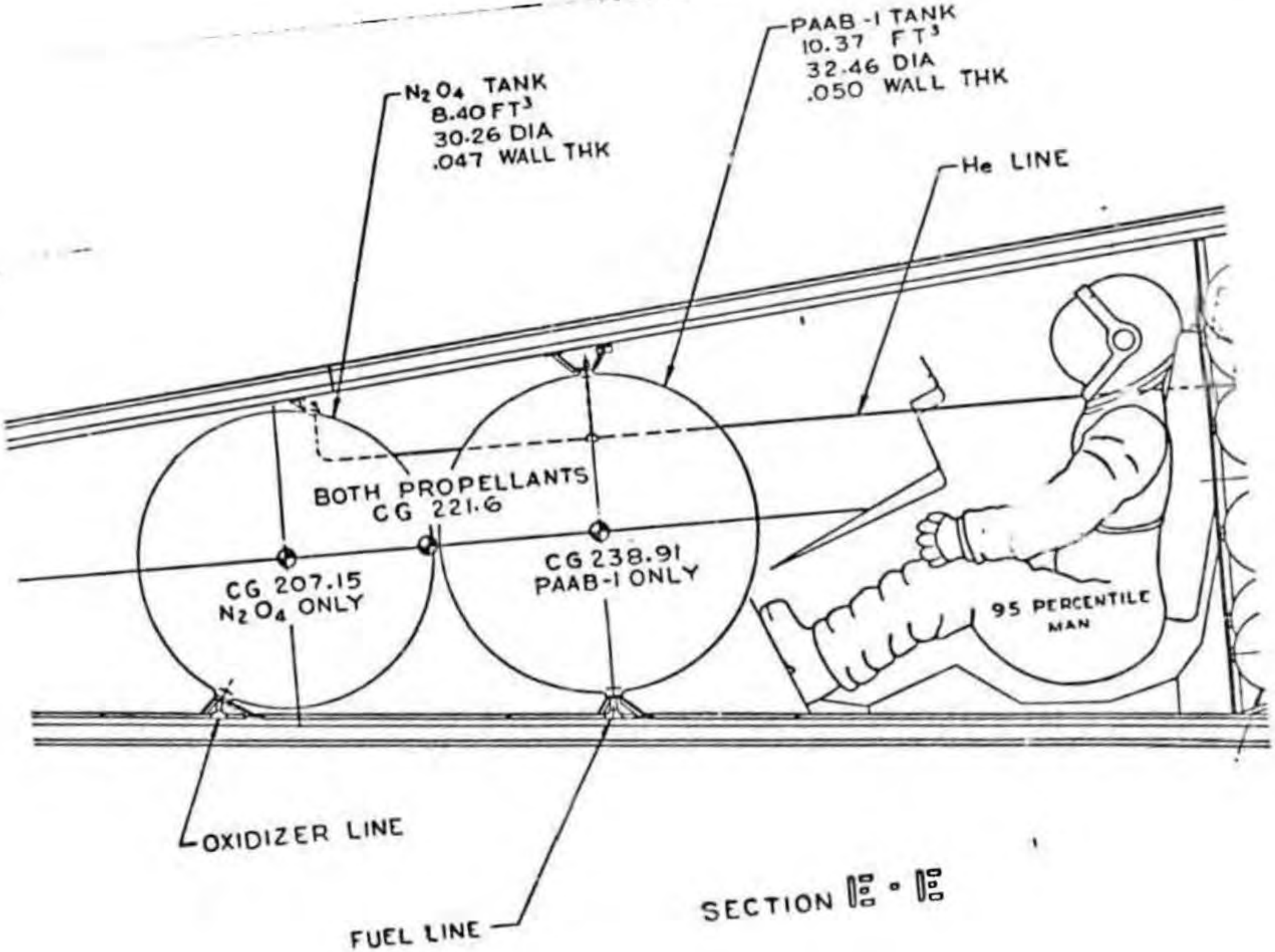


Figure 5-54 N_2O_4/PAAB-1 Propellant Tanks

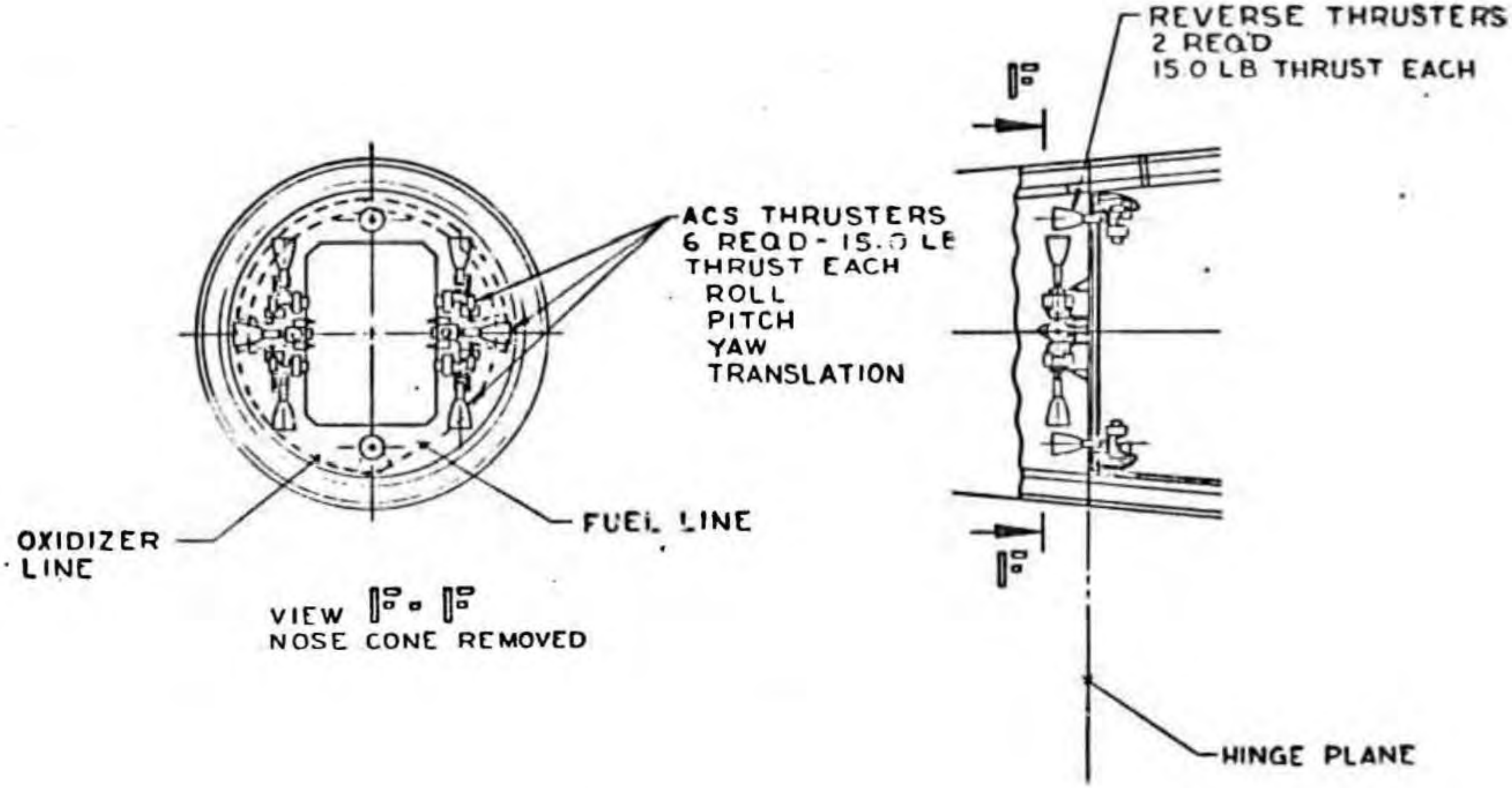


Figure 5-55 **N_2O_4/PAAB-1** Forward **RCS Thrusters**

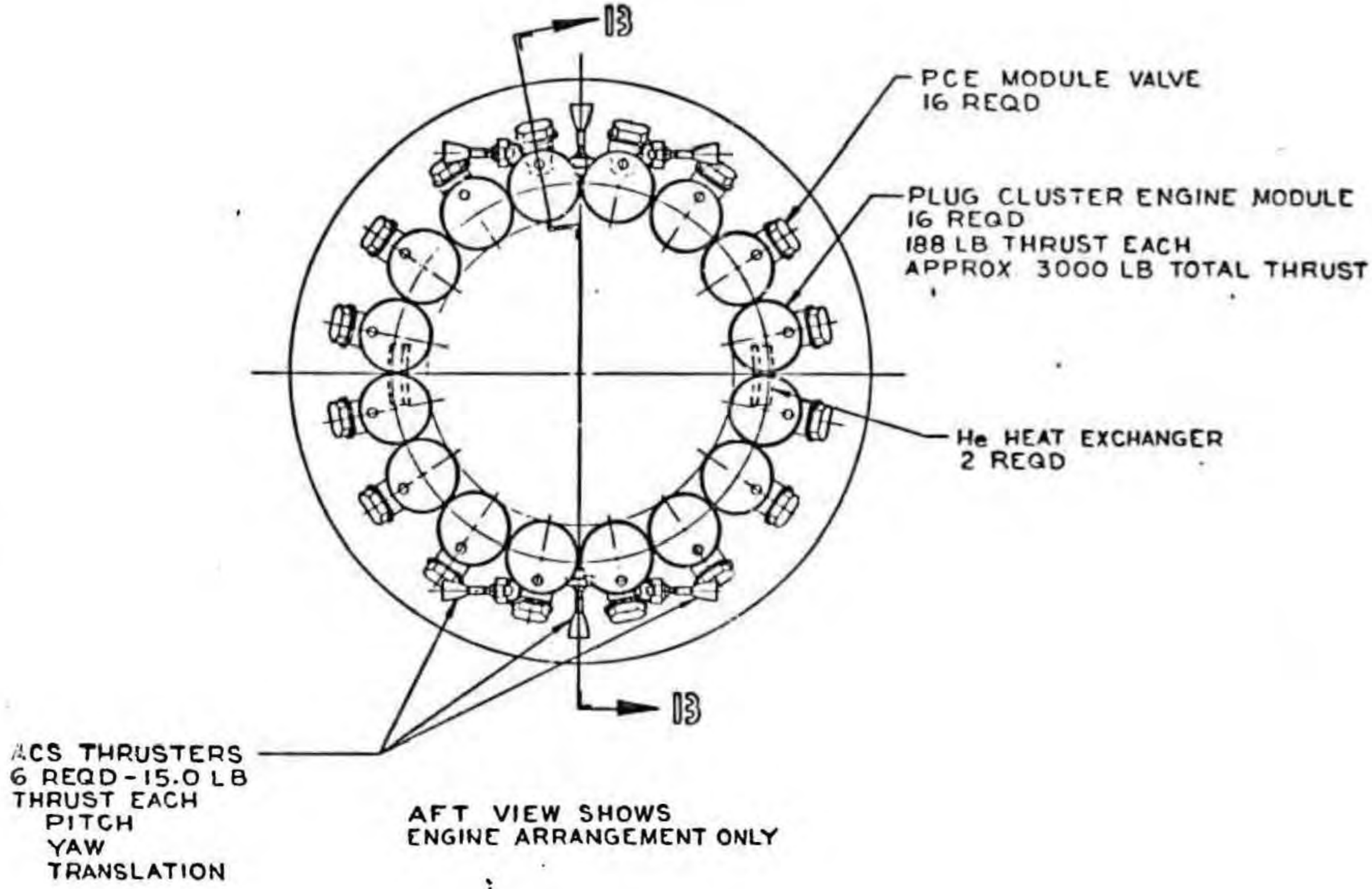


Figure 5-56 N_2O_4/MMH Plug Cluster Engine (Rear View)

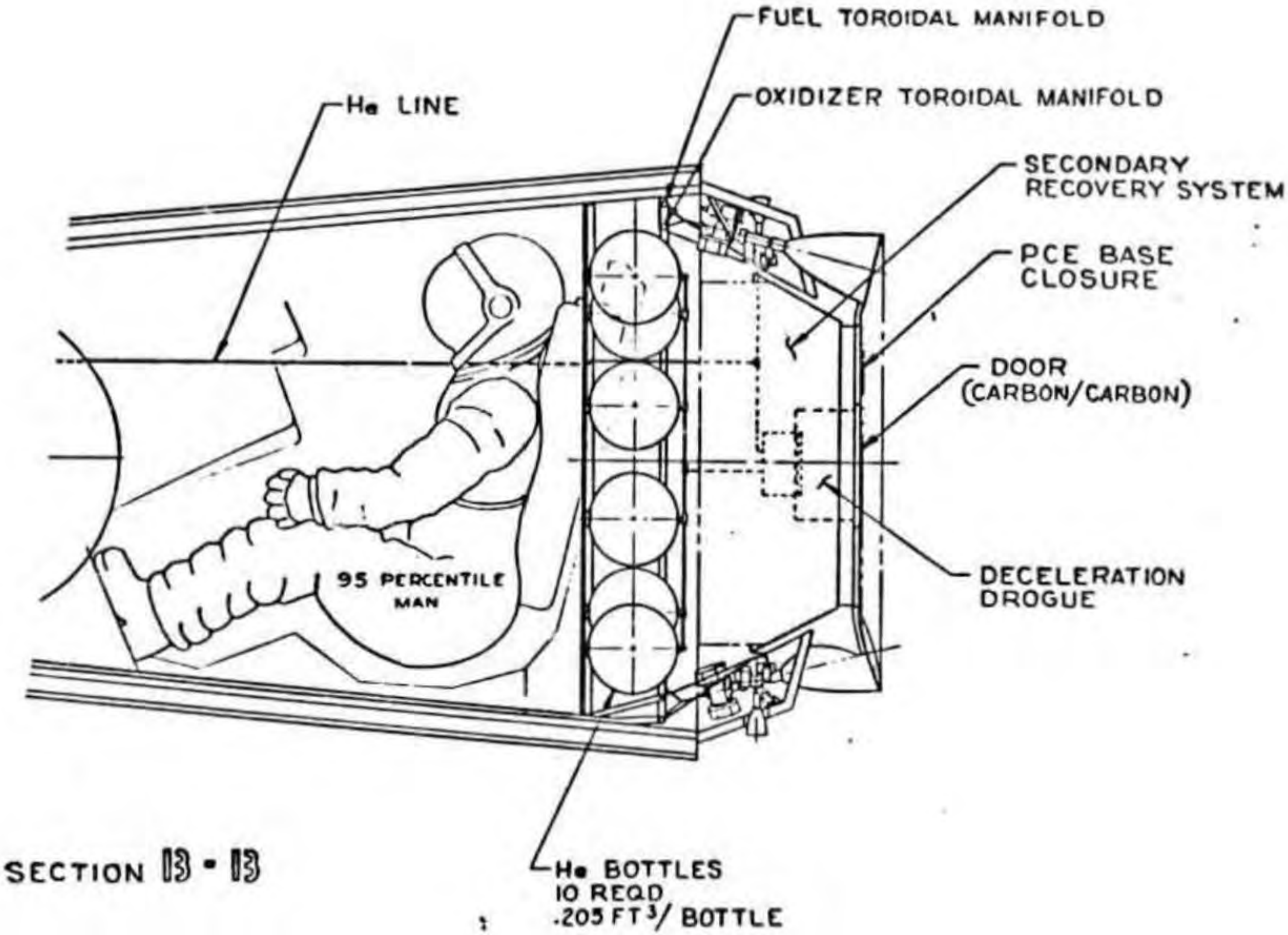


Figure 5-57 N_2O_4/MMH Plug Cluster Engine (Side View)

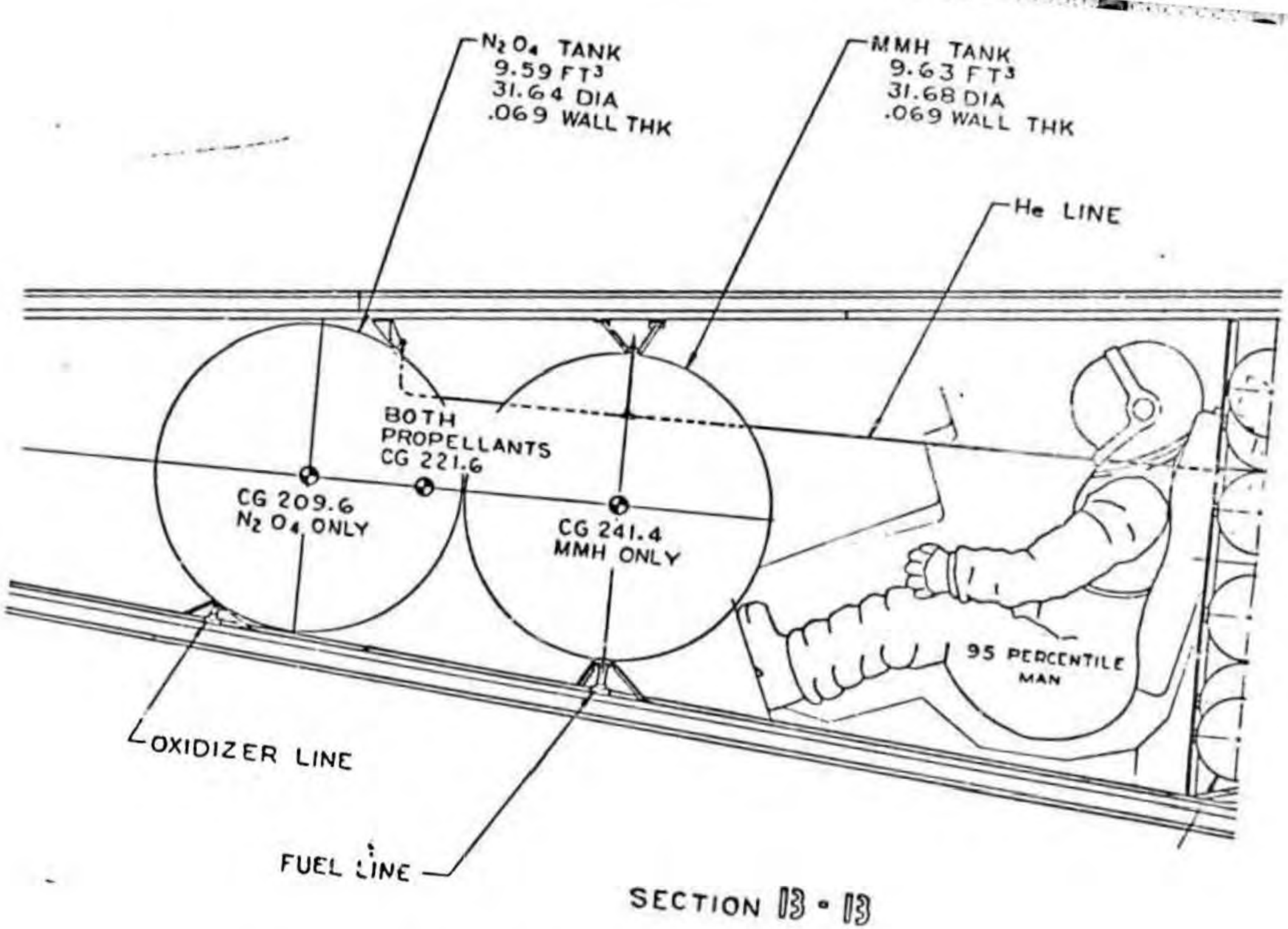


Figure 5-58 N_2O_4/MMH Propellant Tanks

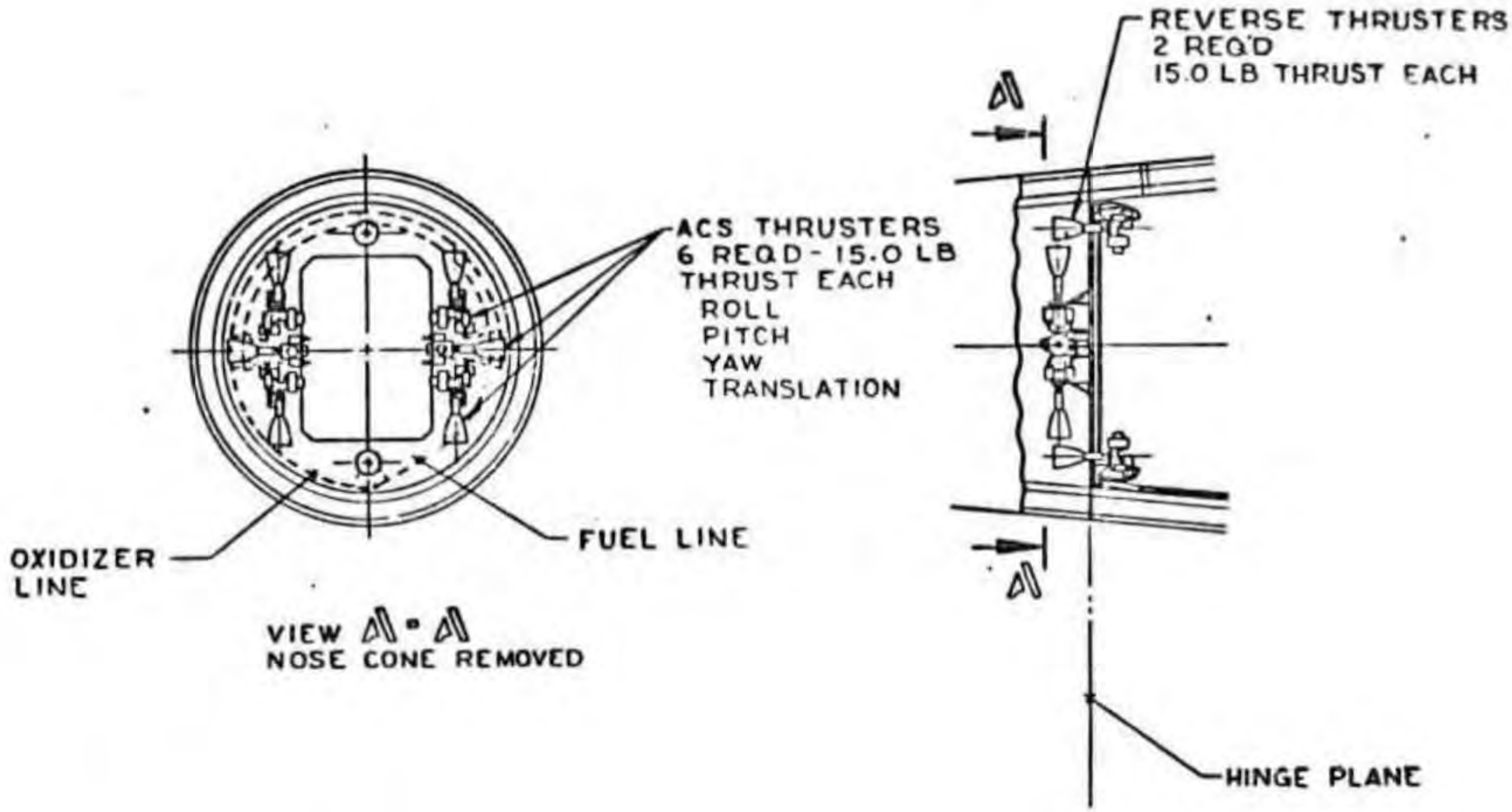


Figure 5-59 N_2O_4/MMH Foreward RCS Thrusters

- The N_2O_4/PAAB-1 RCS thrusters, which will also operate at a chamber pressure of 100 psia, will have approximately the same nozzle envelope as the N_2O_4/MMH RCS thrusters but with a larger throat radius. The smaller throat size in turn results in a smaller nozzle area ratio with resulting lower performance.

5.9.2 RCS Nozzle Definition

All N_2O_4/MMH RCS nozzles, except for the two aft-mounted retro thrusters, have an area ratio of 40:1. The size and location of these 14 RCS thrusters are shown in Figures 59 and 56, and include 8 forward mounted and 6 aft-mounted, respectively.

A N_2O_4/PAAB-1 RCS will utilize nozzles of a somewhat smaller area ratio (10 to 30) because of the lower RCS thruster chamber pressure (100 psia for N_2O_4/PAAB-1 vs. 142 psia for N_2O_4/MMH). The lower area ratio will result in a N_2O_4/PAAB-1 RCS thruster nozzle approximately the same size as the N_2O_4/MMH RCS thruster nozzle.

The requirement for additional retro thrust capability was met by placing two 188 lbF thrusters between two pairs of PCE modules as shown in Figure 53. This configuration operated alone, that is without the two retro thrusters mounted in the nose cone hinge plane, will provide a total axial reverse thrust of approximately 185 lbF, which generates a vehicle deceleration of 1.2 ft/sec^2 or 0.037 g's. Packaging of other Spaceplane subsystems did not require any modification to any of these existing nozzles.

5.9.3 Thrust Load Path Definition

A preliminary thrust load path definition, based on the current PCE/RCS configuration is illustrated in Figures 60, 61 and 62.

Figure 60 (taken from ALRC Drawing 1195445) shows the RCS thruster assemblies located on the Spaceplane nose cone hinge plane. All eight RCS thrustters are attached directly to the mounting bulkhead. This bulkhead is attached at its periphery directly to the Spaceplane vehicle structure. During forward RCS firing, this bulkhead will experience net forces in the -X, ±Z and ±Y vehicle axes. The roll maneuver will result in a shearing stress between the bulkhead and vehcile structure at the bulkhead periphery. These forces, as they act on the vehicle structure, are indicated in Figure 60.

Figures 61 and 62 (also taken from ALRC Drawing 1195445) show the RCS thruster assemblies located on the aft end of Spaceplane vehicle. In this case, all 6 RCS thrusters are mounted, with brackets, on the Spaceplane tail cone structure. This tail cone structure surface should require less TPS than the forward conical surface which experiences the full re-entry thermal environment. This tail cone structure, restricted to an imaginery 20° half angle cone extending aftwards from the Spaceplane widest diameter, should be exposed only to recirculating exhaust gases from the PCE modules and aft RCS thrusters and recirculating hot air during atmospheric re-entry. During PCE firing the tail cone structure will experience a net force in the positive Y-axis direction due entirely to the PCE thrust.

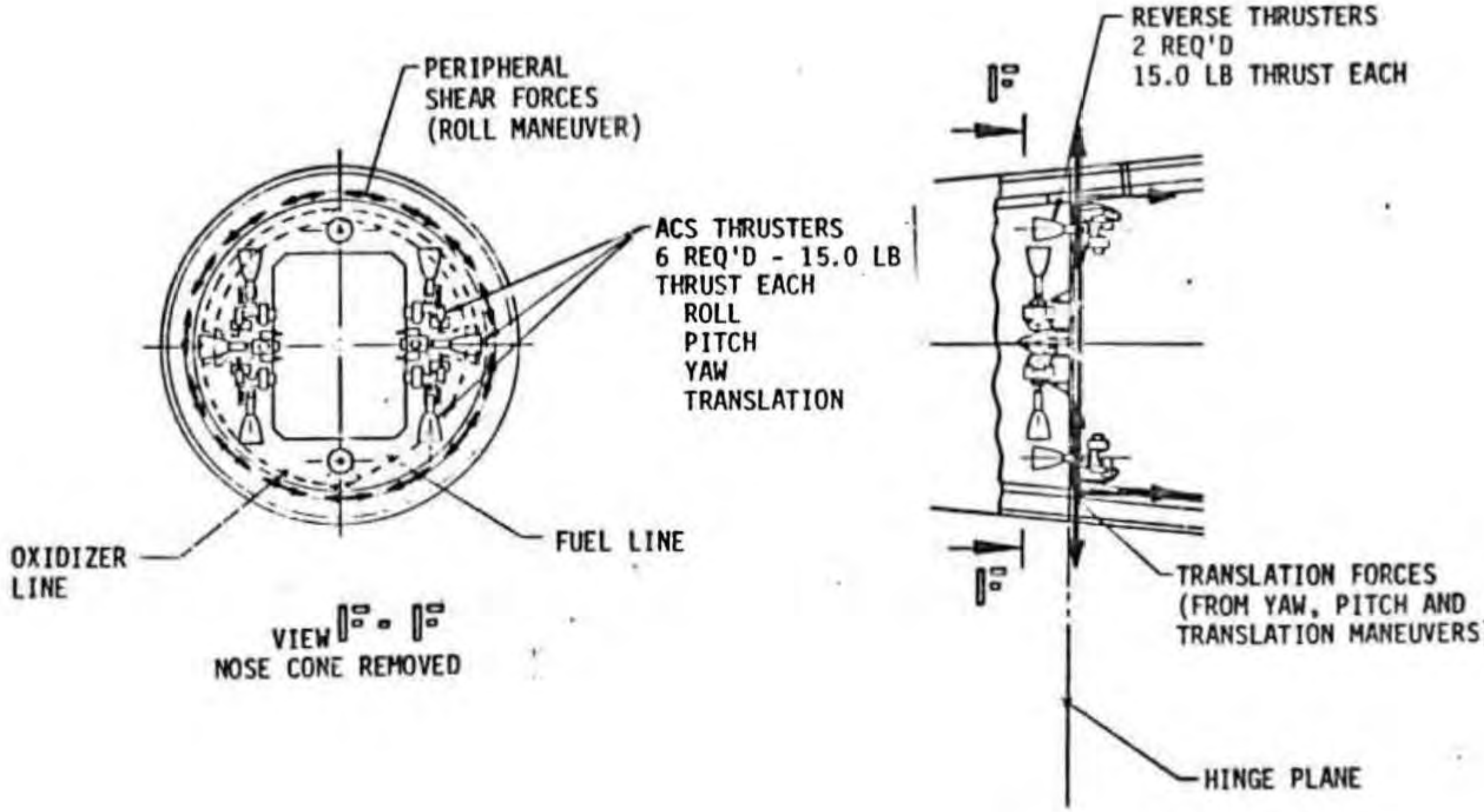


Figure 5-60 Forward RCS Thrust Load Path Definition

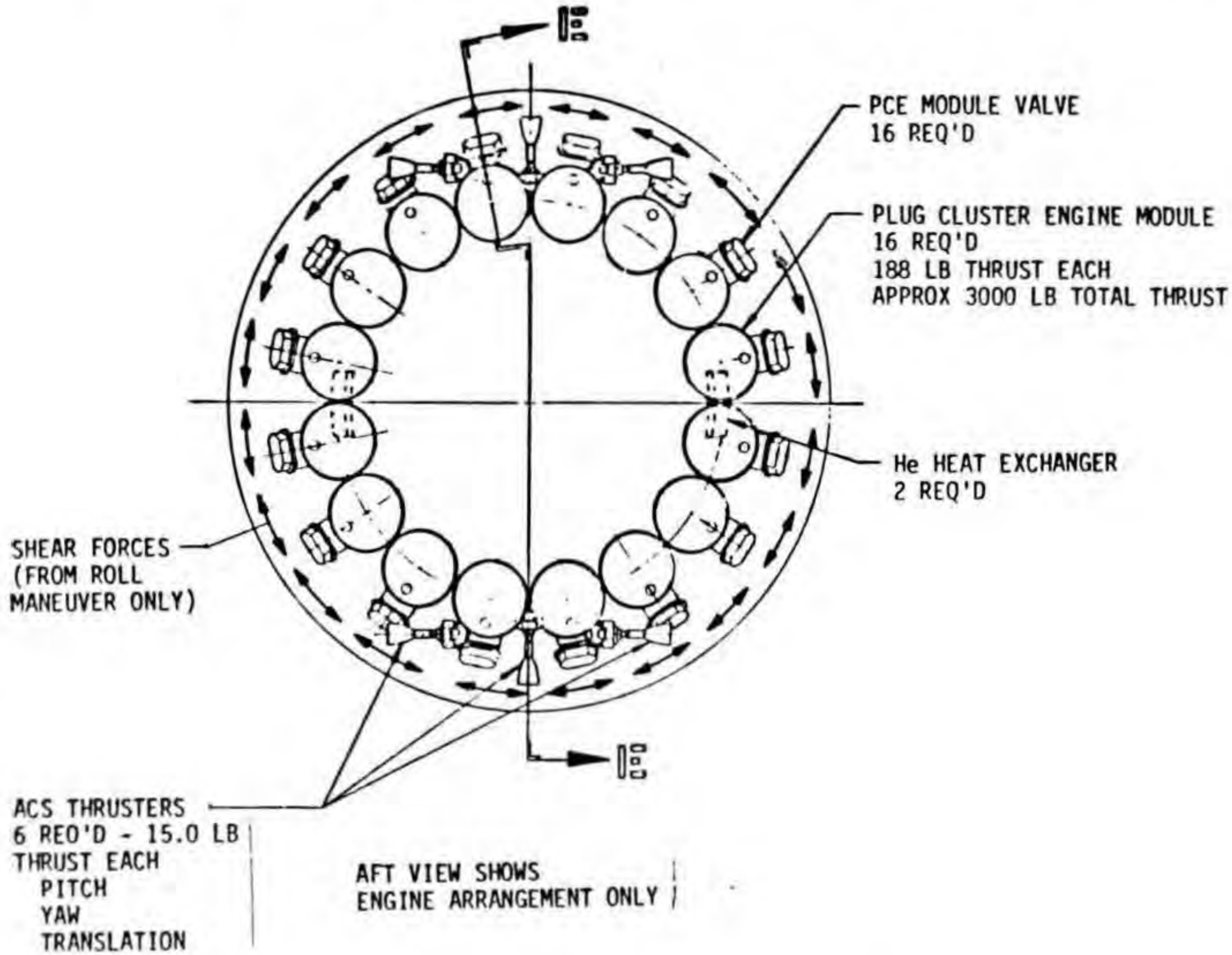


Figure 5-61 **Aft Plug Cluster Engine/RCS Thrust Load Path Definition (Rear View)**

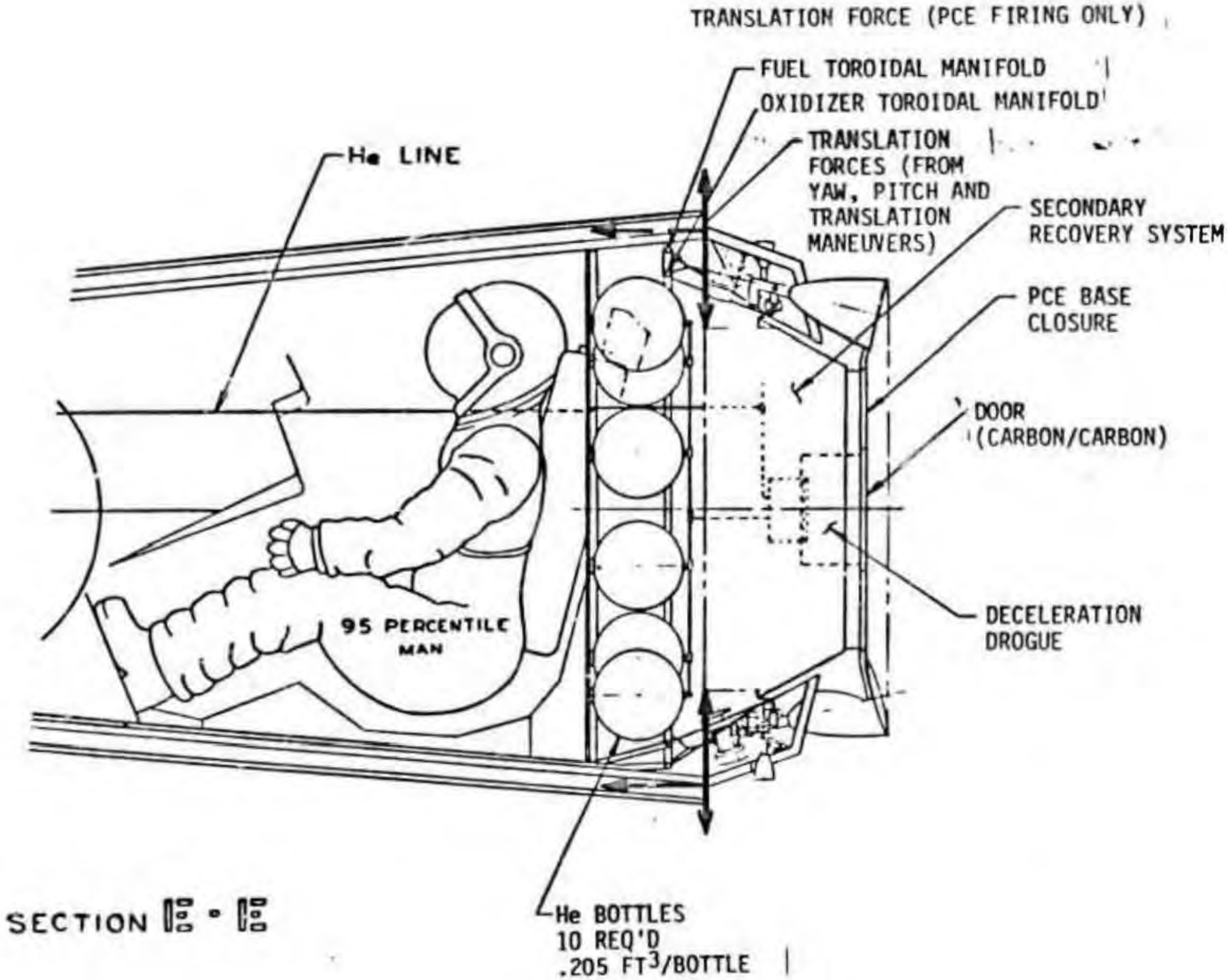


Figure 5-62 **Aft Plug Cluster Engine/RCS Thrust Load Path Definition (Side View)**

Fortunately, this thrust is distributed evenly around the periphery of the tail cone structure. This thrust loading arrangement eliminates the need for, and problems associated with, a structural assembly to distribute the force of a single large thruster to the vehicle structure. There is also a pressure force of approximately 148 lbF (average pressure of 0.20 psia) acting on the surface of the PCE base.

After RCS thruster operation will result in net forces acting on the tail cone in the $\pm$Y axes, and $\pm$Z axes. Although not intended for performing the roll manuever, the aft RCS thrusters can perform an emergency, or backup, roll maneuver. This would result in a net shearing force between the tail cone structure and main Spaceplane structure. These forces, as they act on the Spaceplane forward conical structure, are indicated in Figures 61 and 62.

5.10 TASK 3.2 PROPULSION SYSTEM INTERFACE DEFINITION

The objective of Task 3.2 was to establish Spaceplane onboard propulsion system fluid, power, command and control interfaces. First, the four general interface types were defined:

- o Fluid (i.e., He and propellants)
- o Mechanical
- o Command (interface activation determines status of propulsion system operation)
- o Control (interface activation monitors status of propulsion system operation)

The fluid interfaces are listed and briefly described in Table XI. All of these fluid interfaces are strictly propulsion system interfaces (i.e., propulsion subsystems interfacing with other propulsion subsystems). The definition of these interfaces is an important part of the vehicle integration work because the propulsion subsystems (PCE, RCS, propellant pressurization and propellant tanks) must be located in several different areas of the Spaceplane. The vehicle contractor (H/S) also needs to know if and how these subsystems interface since this can impact the placement of other, non-propulsion Spaceplane subsystems.

TABLE XI

SPACEPLANE PROPULSION SYSTEM FLUID INTERFACES

Fluid	Interfaced Subsystems/Components From	To	Approx Line ID (in.)
N_2O_4	N_2O_4 Tank	PCE	0.50
MMH	MMH Tank	PCE	0.50
He	He Bottle(s)	He Heat Exchanger(s)	0.16
He	He Heat Exchanger(s)	Propellant Tanks	0.16
N_2O_4	MMH Tank	ACS Thrusters	0.10
MMH	MMH Tank	ACS Thrusters	0.10

All identified mechanical interfaces are propulsion system/vehicle interfaces. These mechanical interfaces are identified in the Task 3.4 discussion, which deals exclusively with propulsion system/vehicle mechanical interfaces. The command interfaces are identified in Table XII below.

TABLE XII

SPACEPLANE PROPULSION SYSTEM COMMAND INTERFACES

Valve	Valve Requirement (watts) at 28 volts dc
PCE Module Bipropellant Valve(s) (16)	30 watts/valve
ACS Thruster Bipropellant Valves	15 watts/valve
N_2O_4 Tank Shutoff Valve (OTSV)	45 watts
MMH Tank Shutoff Valve (FTSV)	45 watts
N_2O_4 Tank Isolation Pilot Valve	30 watts
MMH Tank Isolation Pilot Valve	30 watts
He Bottle(s) Vent Valve(s)	30 watts
N_2O_4 Tank Vent Valve	30 watts
MMH Tank Vent Valve	30 watts

A command interface has been defined as an electrical power interface in which electrical power is utilized by one or more components of the Spaceplane internal propulsion subsystem (i.e., valves). In some cases the activation of this interface will be implemented by the pilot (manual operation). In other cases, a command interface will be activated automatically by the Spaceplane avionics/control system in an automatic feedback control mode of operation. For example, a desired on orbit Spaceplane vehicle attitude may be required. The Spaceplane avionics/control system may operate selective RCS thrusters, via the electrically powered RCS thruster solenoid propellant valves, to maintain the specified attitude without pilot intervention. All of the command interfaces, therefore, originate at the Spaceplane electrical power source (batteries and/or APU powered generators). The control interfaces are identified in Table XIII.

A control interface has been defined as one in which low level electrical power (FF1 watt) is required to monitor a particular propulsion system parameter, such as PCE module chamber pressure, propellant tank pressure, etc. This interface type would be represented by all pressure transducers, thermocouples (or thermistors) and any flowmeters. In other words, control interfaces monitor the propulsion subsystem rather than direct it. The state of the various propulsion subsystem parameters (e.g., flowrates, PCE module chamber pressure, He bottle pressure, etc.) will determine what command interfaces are activated to operate the Spaceplane propulsion system.

All of the control interfaces are measured and/or monitored in the cockpit. For this reason, all electrical leads originating at pressure transducers, thermocouples or thermistors, and flowmeters will terminate in the cockpit avionics/control system. The power requirement for all of the control interfaces would be much less than 1.0 watt (i.e., 0.1 to 0.5 watts).

TABLE XIII

SPACEPLANE PROPULSION SYSTEM CONTROL INTERFACES

Monitoring Device Location	Parameter Being Measured	Monitoring Device
PCE Modules	PCE Module Chamber Pressure	2 or 4 lead strain gage pressure transducer
PCE Modules	PCE Module Nozzle Temp	2 lead thermocouple
N_2O_4 Line to PCE	N_2O_4 Temperature (to PCE)	2 lead thermocouple
MMH Line to PCE	MMH Temperature (to PCE)	2 lead thermocouple
N_2O_4 Line to PCE	N_2O_4 Flowrate to PCE	Turbine type flowmeter
MMH Line to PCE	MMH Flowrate to PCE	Turbine type flowmeter
N_2O_4 Tank	N_2O_4 Tank Pressure	2 or 4 lead strain gage pressure transducer
MMH Tank	MMH Tank Pressure	2 or 4 lead strain gage pressure transducer
N_2O_4 Tank Vent Valve	He Temperature in N_2O_4 Tank	2 lead thermocouple
N_2O_4 Tank Vent Valve	He Pressure in N_2O_4 Tank	2 or 4 lead strain gage pressure transducer
MMH Tank Vent Valve	He Temperature in MMH Tank	2 lead thermocouple
MMH Tank Vent Valve	He Pressure in MMH Tank	2 or 4 lead strain gage pressure transducer
He Bottle Fill and Vent Valve(s)	He Temperature in He Bottle	2 lead thermocouple
He Bottle Fill and Vent Valve(s)	He Pressure in He Bottle	2 or 4 lead strain gage pressure transducer
N_2O_4 Tank	Propellant Qty (% or abs)	TBD
Fuel Tank	Propellant Qty (% or abs)	TBD

5.11 TASK 3.3 INSTALLED PROPULSION SYSTEM PERFORMANCE

The objective of this task was to define: (1) the total impulse (specific impulse multiplied by total propellant load or thrust multiplied by total burn time) of the selected baseline Spaceplane onboard propulsion system, and (2) the resulting Spaceplane vehicle (SP). Potential improvements to either parameter were also to be defined.

Three methods of total Spaceplane vehicle onboard propulsion system improvement evaluated were:

- o Use of Orbit Transfer Vehicles (OTV's) to increase total Spaceplane
- o Use of alternate propellant combinations
- o Use of auxiliary propellant tanks located in forward and aft Spaceplane payload bays

5.11.1 Total Impulse and Spaceplane

The current value of these parameters are listed in Table XIV below.

TABLE XIV

SPACEPLANE INSTALLED PROPULSION SYSTEM PERFORMANCE

Spaceplane Propulsion System Performance Parameter	Propellant Combination N_2O_4/MMH	N_2O_4/PAAB-1
PCE Thrust, lbF	3,052.7	3,058.1
Isp VAC of PCE, seconds	312.9	316.9
Total Usable Propellant Load, lbM	1281.	1281.
PCE Thrust, lbF	3052.7	3058.1
PCE Burn Time, seconds	131.3	132.7
Isp Total, lbF-seconds	400,812.1	405,884.9
SP , ft/second	2,497.2	2,559.7

Some of the important assumptions on which this table are based are discussed below.

- o The Isp values are based on full PCE thrust operation (i.e., all PCE modules are firing). One or more PCE modules not firing will result in a slight Isp decrement because of the partial or total loss of base thrust which contributes to the PCE total thrust and hence Isp. This Isp loss will increase with an increasing number of non-firing PCE modules. The actual Isp loss will range from 0 to a maximum of 14 seconds.

- o The Spaceplane dry vehicle weight, not including the propulsion system (i.e., PCE, RCS, propellant pressurization and propellant control systems) or propellant, is assumed to be 4155 lbM per directions from H/S.

- Approximately 30 lbM of the total possible propellant load (1400 lbM) has been reserved for use by the RCS. This is equivalent to 50 ft/sec of SPΔV.

- A 3.4% propellant tank ullage volume has been assumed. This is equivalent to approximately 48 lbM of propellant.

- A propellant residual of 1% of the total PCE propellant load has been assumed. This is equivalent to 13 lbM of propellant which is considered not usable for either the PCE or RCS.

- A propellant reserve adequate to provide an RCS based SP of approximately 50 ft/sec has been assumed. This, like the RCS allocated propellant weight is approximately 28 lbM.

- The effect of the four previous assumptions is to reduce the total usable axial thrust propellant load from 1400 lbM to 1281 lbM (a reduction of 119 lbM).

- In the N_2O_4/PAAB-1 case, no allowance has been made for the PAAB-1 (approximately 100 lbM) required to run the Spaceplane APU's. This will not degrade the Spaceplane ΔV capability, however, since the PAAB-1 tank would be enlarged to include the extra PAAB-1 needed.

- The SP values shown in the tables are slightly different from those generated by the computer program SPV1 primarily because SPV1 assumes the total propellant load (1400 lbM) is usable by the PCE.

As a final note, the values in Table XIV, especially SPΔV, will change as the Spaceplane vehicle dry and wet weights are better defined. The Isp values will not vary unless a PCE design parameter (such as P_c or available PCE diameter) is changed.

5.11.2 Orbit Transfer Vehicles

A comparison was made of storable and cryogenic Orbit Transfer Vehicles (OTV's) for transporting the Spaceplane from LEO (Low Earth Orbit) to Geosynchronous Earth Orbit (GEO). One of the major drivers of OTV performance (i.e., total ΔV capability) is stage propellant mass fraction (MF) which is defined as:

$$MF = M_{prop}/M_{bo}$$

where M_{prop} = Total usable main propulsion propellants, lbM
and M_{bo} = Stage burnout mass, lbM

For the purposes of this comparison, M_{prop} does not include RCS propellant weight. M_{bo} includes the vehicle structure, propulsion system, consumables, unusable propellants and material normally ejected non-propulsively (e.g., H_2O). The purpose of these somewhat strict definitions was to enable an "apples to apples" comparison of conceptual and existing OTV-like vehicles (such as Centaur).

MF data for over 60 conceptual cryogenic and storable OTV's was collected. Most of the data could be included in the following categories:

- Centaur
- Space Tug Studies (Storable OTV)
- Low Thrust OTV Studies (or COTV for Cargo OTV)
- Manned OTV Studies (MOTV for Manned OTV)
- DoD Advanced Spacecraft Deployment System Studies (Military MOTV)
- Space based OTV's

The MF data considered was generated between early 1974 and early 1982. An analysis of the compiled data showed two important trends:

- Study MF values tended to decrease with time. That is, the later the data, the more conservative it became and more consistent with the only existing cryogenic OTV-like vehicle, Centaur.
- With the above observation taken into account, MF values for cryogenic OTV's appear to be in the .83 to .89 range. Storable OTV's have MF values in the .86 to .95 range. The higher MF values for the storable OTV is consistent with existing aerospace vehicle data which shows that storable propellant stages or vehicles do in fact possess higher propellant mass fractions.

On the basis of these trends, an MF of 0.860 was selected for cryogenic OTV's and 0.905 was selected for storable OTV's. These values are somewhat higher than the MF values (.746 and .840) assumed for the drop tank assemblies evaluated previously in Task 3.5 (Maintenance and Refueling). The primary reason for this difference is that virtually all OTV concepts, including the Centaur, utilize pumpfed propulsion systems, whereas the drop tank assemblies considered were exclusively pressure fed. In general, except for very small ΔV vehicles, a pump-fed based vehicle will weigh less than a pressure-fed based vehicle of equal propellant mass, and hence, possess a higher MF.

The final results of the cryogenic vs. storable OTV (Orbit Transfer Vehicle) comparison are listed below.

- A typical cryogenic OTV with a Spaceplane as a payload will have a ΔV capability of about 22,620 ft/sec (this compares reasonably with the 25,088 ft/sec value for the Centaur which includes the ΔV due to use of the RCS)
- A typical storable (N_2O_4/PAAB-1 or N_2O_4/MMH) OTV with a spaceplane as a payload will have a ΔV capability of about 18,300 ft/sec.
- The cryogenic OTV possess a total ΔV advantage of approximately 4300 ft/sec over the storable OTV with equal propellant masses.
- A 14-foot diameter storable OTV will be about 9.7 feet shorter than the equal propellant mass, equal stage diameter cryogenic OTV.

5.11.3 Alternate Propellants

Another subtask performed in support of Task 3.3 was the identification and evaluation of alternate propellant combinations which could yield substantial Spaceplane ΔV (SP ΔV) improvements. The results of this investigation are discussed below.

In general, the selection of the optimum liquid rocket propellants for use in a space vehicle depends on many factors in addition to the delivered vacuum specific impulse (Isp_v). This concept can be illustrated by examining the ideal ΔV equation:

$$\Delta V = g_c \, Isp_v \ln (M_I / M_{bo}) \quad (1)$$

where

ΔV = ideal vehicle velocity increment neglecting gravity and atmospheric losses, ft/sec

g_c = acceleration due to gravity, 32.174 ft/sec^2

M_I = total vehicle mission start weight (i.e., loaded with propellants), lbM

M_{bo} = vehicle burnout weight (i.e., usable propellants burned and expelled), lbM

Equation (1) shows that, indeed, delta-V is directly proportional to Isp_v. Therefore, all other parameters being constant, delta-V will improve with increasing Isp_v.

The other effect of propellant selection on delta-V is not as easily identified. It may be illustrated by first defining:

$$M_I = M_{bo} + M_{prop} \quad (2)$$

where

M_{prop} = mass of usable propellant stored in the vehicle, lbM

Also

$$M_{prop} = Vol_{prop} \cdot \rho_{bulk} \quad (3)$$

where

Vol_{prop} = volume available in the vehicle for propellant load, ft^3

and

ρ_{bulk} = bulk density of the propellants, lbM/ft^3

Making the proper substitutions, equation (1) may then be rewritten as:

$$\Delta V = g_c \, Isp_v \ln \left(1 + \frac{Vol_{prop}}{M_{bo}} \cdot \rho_{bulk}\right) \quad (4)$$

Equation (4) now shows that for a constant Vol_{prop}/M_{bo} vehicle, an increase of ρ_{bulk} of propellants also increase delta-V. For a constant Vol_{prop} application then, the objective is to select those propellants whose Isp_v and ρ_{bulk} result in the maximum delta-V.

A propellant combination parameter which strongly influences both Isp_v and ρ_{bulk} is mixture ratio (MR). The optimum MR for a given propellant combination and fixed available propellant volume is that MR which results in the highest vehicle delta-V. This optimum MR does not always correspond to the maximum Isp_v.

One convenient way to evaluate both of these influences simultaneously is to first determine ρ_{bulk} as a function of MR for a particular propellant combination. A plot of Isp_v vs. ρ_{bulk} is then made.

Equation (4) can then be used to make plots of Isp_v vs. ρ_{bulk} based on various values of delta-V and a fixed Vol_{prop}/M_{bo}.

This procedure was followed for several propellant combinations with the resulting plot shown in Figure 63. As is apparent in Figure 63, in some cases, Isp_{ode} data was only available at one mixture ratio (as opposed to the range of mixture ratios considered for many of the propellant combinations). Although these mixture ratios are not necessarily optimum for the Spaceplane vehicle, an approximate SP increment can still be estimated.

One observation, among others, that can be made on the basis of Figure 63 is that vehicles with small propellant volumes (i.e., Vol_{prop}/M_{bo} is low) have higher delta-V capability operating with higher density, lower Isp_v propellants (such as N_2O_4/MMH or ClF_5/N_2H_4) than with lower density, high Isp_v propellants (such as Lox/LH_2). On the other hand, as vehicles possess larger and larger propellant volumes (i.e., Vol_{prop}/M_{bo} is high) the lower density, higher Isp_v propellants (i.e., Lox/LH_2) begin providing vehicle delta-V values closer to those provided by the higher density, lower Isp_v propellants.

The variable labeled as ρ_{veh} in Figure 63 is equal to Vol_{prop}/M_{bo}. This plot shows both the maximum delta-V that may be expected from a particular propellant combination and the MR required to yield that maximum delta-V. also, Figure 63 is based on ODE Isp (i.e., maximum theoretical) values whereas a more rigorous analysis would utilize Isp_v (i.e., delivered Isp values).

There are some secondary influences not considered by this analysis which would normally be evaluated in a detailed design procedure. For example, the effect of propellant combination selection on M_{v2} has not been considered. A cryogenic tank normally weighs considerably more than an equal volume, equal pressure tank containing storable propellants. In the present case only storable propellant combinations are considered.

This proecure was used to identify other propellant combinations (besides N_2O_4/MMH and N_2O_4/PAAB-1) which would improve SPΔV based upon a relatively constant total volume of 18.64 ft^3 in the Spaceplane available for propellants. These other propellant combinations and the approximate SP improvement attributable to each are listed in Table XV.

It should be noted here that a somewhat different propellant selection procedure would be followed if the ratio of Vol_{prop}/M_{bo} were not fixed (e.-g., fixed M_1).

$\Delta V = g\ Isp\ \ln(\rho_B/\rho_{veh} + 1.)$

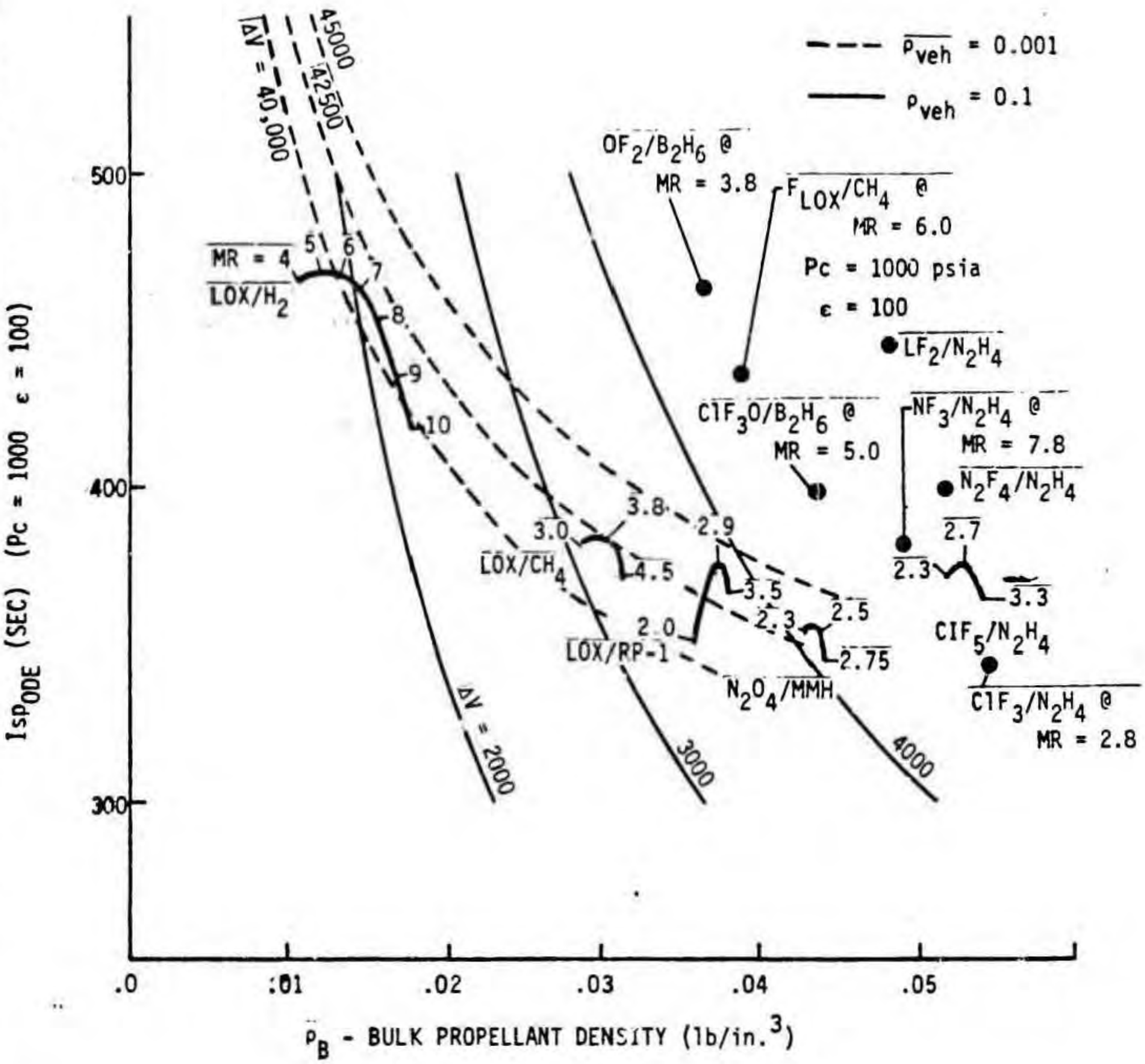


Figure 5-63 Vacuum Delta V Equation Applied to Propellant Performance Properties

TABLE XV

CANDIDATE SPACEPLANE PROPELLANTS

Propellant Combination	ρ_{bulk} (lbM/ft^3)	SPΔv Increment (ft/sec)
LF_2/N_2H_4	83.3	1,550.
N_2F_4/N_2H_4	89.3	1,200.
ClF_3O/MMH-Al (70-30)	99.4	1,140.
BrF_5/MMH-Al (70-30)	121.0	1,130.
ClF_5/PAAB-1	90.9	955.
ClF_3O w/MHF_3, MMH or PAAB-1	92.1 - 93.3	830. - 880.
BrF_5/PAAB-1	115.8	880.
NF_3/N_2H_4	85.0	700.
ClF_3/N_2H_4	93.8	750.
ClF_5 w/MHF_3 or MMH	86.7 - 87.6	680. - 730.
BrF_5/MMH	112.3	680.
OF_2/B_2H_6	62.6	500.
ClF_3O/B_2H_6	75.2	450.
FLox/CH_4	66.5	400.
N_2O_4/PAAB-1	74.6	30.
N_2O_4/MMH	75.2	0

The assumption was also made that cryogens (i.e., LH_2, Lox) or hydrocarbons like propane or methane would not be used aboard the Spaceplane. All possess cryogenic characteristics that make rapid propellant loading required for a military vehicle very difficult.

There are other considerations which also influence propellant selection in addition to those considered in this procedure. These are discussed briefly here.

The propellant combination BrF_5/MMH-Al would yield an impressive SP improvement (1140ft/sec). However, the propellant bulk density of this combination is considerably higher than that of N_2O_4/MMH (121. - 75.2) lbM/ft^3 = 854. lbM. In other words, a maximum vehicle mission start weight may determine the maximum propellant bulk density permissible. An alternative solution would be, if a higher ρ_{bulk} propellant combination were selected, to use smaller tanks to reduce the Spaceplane total mission start weight. However, this would also reduce SP .

The propulsion system weight, which is part of M_{bo}, is strongly influenced by the propellant combination. In general, the lower the propellant bulk density (such as Lox/LH_2 and Lox/Hydrocarbons), the greater the engine weight. For this reason, a higher propellant bulk density can result in a double benefit to the vehicle: (1)possibly higher Isp_v, and (2) lighter engine weight.

The technology status of the various propellant combinations is another important consideration in the selection of propellants. All of the propellant combinations listed in Table XV, except N_2O_4/MMH and N_2O_4/PAAB-1, are advanced technology propellants. In fact, as might be expected, the higher the potential SP improvement, the more advanced the propellant combination. For example, many propellant combinations were considered in the Advanced Spacecraft Deployment System Study (ASDS) recently conducted at ALRC (Ref.2). Part of this study was a technology status evaluation for several of these propellant combinations. This evaluation is summarized in Table XVI. The actual engine systems (including existing, R&D and conceptual systems) on which this technology status determination was made are summarized in Table XVII. (The references cited in Table XVII are listed in the ASDS Final Report noted above).

Any of the propellant combinations in Table XV that have not been specifically mentioned in the foregoing discussion (such as LF_2/N_2H_4, N_2F_4/N_2H_4, NF_3/N_2H_4, etc.) definitely represent new technology. For this reason alone, and assuming a desired near term IOC for the Spaceplane, none of these propellant combinations are recommended for Spaceplane application.

It should also be noted that any aluminum in the rocket exhaust plume is more readily identified by a pronounced IR signature. This may be a major disadvantage for the Spaceplane.

Another important propellant selection consideration is STS Orbiter/propellant compatibility. It is entirely possible that a new propellant combination (i.e., any propellant combination that is not fully qualified) may have to be qualified for storage aboard the Orbiter. This restriction might even apply to the use of N_2O_4/PAAB-1.

TABLE XVI

CANDIDATE PROPELLANTS SELECTION (ASDS)

PROPELLANT COMBINATION	TECHNOLOGY STATUS	SELECTION CRITERIA
SELECTED		
LO_2/LH_2	High	High performing baseline cryogenic (probably used in booster stage)
N_2O_4/MMH	High	Low performing baseline storable
LO_2/RP-1	High	Probably will be used in booster stage (space storable)
LO_2/CH_4	None	May replace RP-1 in booster stage (space storable)
LF_2/LH_2	High	Highest performing bipropellant combination
LF_2/N_2H_4	Low	Being developed for NASA space missions
ClF_3O/B_5H_9	None	Highest performing storable
$ClF_5/LH_2/N_2H_4$	None	Mixed mode tripropellant with high density storable bipropellant
NOT SELECTED		
ClF_5/N_2H_4	Low	Highest bulk density (to be evaluated as tripropellant component)
98% H_2O_2/Al-43	Low	Slightly lower performing than ClF_3O/B_5H_9
LO_2/MMH	None	Not significantly different from LO_2/HC (RP-1, CH_4, etc.)
LO_2/C_3H_8	None	Not significantly different from other RP-1, CH_4, etc. except if subcooled
FLOX/LH_2/CH_4	None	High performing tripropellant -- space storable bipropellant
$LF_2/LH_2/N_2H_4$	None	Highest performing tripropellant -- space storable bipropellant
LO_2/LH_2/RP-1	High	Most likely booster tripropellant -- space storable bipropellant

TABLE XVII

ROCKET ENGINE STATE-OF-THE-ART SUMMARY (ASDS)

PROPELLANTS	Pc (psia)	F (lbF)	ϵ	MR	η_{Is}	YEAR	CONTRACT	REFERENCE	TECHNOLOGY STATUS
LF_2/LH_2	400	15,000	[illegible]	10	.96	1969	NAS3-7991	33	High: Pump-fed, flight-weight RL10A-3-3 (modified) engine testing
	100	2,500	60	≤15	.90	1967	NASw-1229	10	Low: Research & Development (R&D)
	400	8,000	60	12	-	1970	NAS3-7971	18	Low: Chamber Fabrication
LO_2/LH_2	2,000	20,000	400	[illegible]	.98	[illegible]	NAS3-16751 NAS3-20386	22	High: Engine components demonstrated (highest reported engine efficiency)
	400	15,000	57	5	.96	1966	NAS8-5623 NAS8-31008	15	Flight Operational: RL10A-3-3
	300	1,500	40	5.5	.93	1978	NAS3-20107	11	High: Demonstrated > 1200 Thermal Cycles
	300	1,500	40	4.0	.94	1973	NAS3-15850	12,13,14	High: Flightweight Chamber Demonstrated 51,000 pulses, 660 seconds duration
OF_2/B_2H_6	100	1,000	60	3.0	.88	1970	NAS7-100 NAS7-741	7,8	Low: Originally planned for operation in 1980's - abandoned system due to fuel cost and availability
$FLOX/B_2H_6$									
	100	2,500	60	4.0	.93	1969	NASw-1229	9	Low: R & D
	100	100	25	3.0	-	1970	NAS7-660	17	Low: R & D
	100	1,000	60	3.0	-	1971	NAS7-765	29	Low: R & D - Regenerative cooling demonstrated with B_2H_6 and OF_2
	100	1,000	2	3.2	.87	1969	NAS7-659	30	Low: R & D - Film cooling demonstrated
LF_2/N_2H_4	100	800	80	1.6	.89	1979	NAS7-100	5,6	Low: Proposed 1982 operational system utilizes 1960s state-of-the-art injector technology
	100	7,000	35	1.9	.91	1968	F0 4611-67-C-0003	47	Low: Demonstrated 605 seconds duration chamber
$FLOX/CH_4$	250	13,000	40	5.8	.94	1969	NAS3-7950	2	High: Pump-fed, flight-weight RL10A-1 (Modified) engine testing - CH_4 well suited for expander cycle fluid
OF_2/CH_4									
	100	100	-	-	-	1969	NAS3-11215	1	Low: R & D
	500	5,000	60	5.3	-	1971	NAS3-13315	3	Low: R & D - demonstrated 540 seconds with CH_4 regenerative cooling
	100	2,500	60	5.0	.92	1969	NASw-1229	9	Low: R & D
	500	5,000	60	5.3	-	1970	NAS3-12024	26	Low: Engine design
	500	5,000	60	5.0	.95	1971	NAS3-11190	45	Low: R & D - high performance demonstrated
	100	100	40	5.7	-	1970	NAS3-12558	46	Low: R & D

TABLE XVII (Cont.)

ROCKET ENGINE STATE-OF-THE-ART SUMMARY (ASDS)

PROPELLANTS	Pc (psia)	F (lbF)	ε	MR	η_{Is}	YEAR	CONTRACT	REFERENCE	TECHNOLOGY STATUS
FLOX/C_3H_8	100	1,000	-	4.5	.93	1969	NASA/LeRC	31	Low: R & D
	100	150	2	4.5	.92	1969	NASA/LeRC	32	Low: R & D
	100	100	-	4.5	-	1969	NAS3-12059	4	Low: R & D - Film/dump cooling
	100	25	-	4.5	-	1970	NAS3-12058	46	Low: R & D
NF_3/N_2H_4	400	5,000	-	3.0	.93	1962	AF04(611)-7433	24	Low: R & D
NF_3/B_5H_9	400	5,000	-	2.7	.77	1962	AF04(611)-7433	24	Low: R & D
LO_2/RP-1	1,200	16,000	6	2.5	.95	1979	NAS3-21030	19	Low: R & D - high performance demonstrated
	682	80,000	25	2.3	.94	1960	AF04(647)-521	20	Flight operational: Titan I 2nd stage
	705	204,300	8	2.3	.94	1967	NAS7-190	20	Flight operational: H-1
	600	4,500	-	2.4	-	1979	NASA/LeRC	21	Low: R & D
LO_2/CH_4	500	4,500	-	2.4	-	1979	NASA/LeRC	21	Low: R & D
LO_2/exo-THDCP	600	4,500	-	2.4	-	1979	NASA/LeRC	21	Low: R & D - high density fuel
H_2O_2/Al-43	3,000	20,000	10	0.8	.93	1967	AF04(611)-10785	23	Low: 98% and 90% H_2O_2 monopropellant gas-burner demonstrated
ClF_5/N_2H_4	300	500	15	2.5	.94	1963	Rocketdyne	43	Low: R & D
ClF_3/N_2H_4	300	500	15	2.5	.93	1963	Rocketdyne	43	Low: R & D
ClF_3/MHF-3	1,000	1,000	-	3.0	.89	1970	AFRPL	25	Low: R & D
ClF_3O/MHF-3	1,000	1,000	-	2.8	.92	1970	AFRPL	25	Low: R & D
N_2O_4/MHF-3	1,000	1,000	-	2.5	.94	1970	AFRPL	25	Low: R & D
N_2O_4/MMH	260	4,000	-	1.7	.93	1976	F04611-76-C-0066	16	High: Flight-weight engine demonstrated
	150	100	150	1.65	.92	1979	ALRC	51	High: Flight-weight engine demonstrated
	200	600	40	1.6	.89	1974	NAS9-13476	27	High: Beryllium chamber demonstrated [illegible] seconds
	200	600	40	1.6	.89	1974	NAS9-12996	28	High: Demonstrated 10,411 seconds, [illegible] seconds continuous
	152	872	22	1.6	.89	1979	--	49	High: In PFRT R-40A
	125	6,000	55	1.7	.94	1979	NAS9-14000	40	High: IN PFRT - OMS
N_2O_4/N_2H_4	300	3,000	30	1.0	.92	1971	F04611-70-C-0022	34	High: Flight weight PBPS axial engine - 380 seconds chamber demonstration
	300	3,000	30	1.0	.94	1972	AFRPL	35	High: Demonstrated Technology

Sheet 3

TABLE XVII (Cont.)

ROCKET ENGINE STATE-OF-THE-ART SUMMARY (ASDS)

PROPELLANTS	Pc (psia)	F (lbF)	ε	MR	η_{Is}	YEAR	CONTRACT	REFERENCE	TECHNOLOGY STATUS
N_2O_4/A-50	105	8,000	40	2.0	.92	1979	F04701-76-C-0124	39	High: Flight-weight improved transfer demonstrated
↓	827	100,000	49	1.8	.94	1962	AF04(647)-521	48	Flight operational: Titan II upper stage
↓	104	9,850	48	1.6	.91	1966	NAS9-1100	36	Flight operational: Lunar excursion module descent engine
↓	125	9,606	40	1.9	.91	1970	DAC A68-8001	37	Flight operational: Delta AJ10-118F
↓	122	3,500	46	1.6	.92	1969	NAS9-7498	41	Flight operational: Lunar module ascent engine
↓	100	2,200	60	2.0	.90	1962	AF04(647)-767	42	High: PFRT subscale Apollo service module
	2,800	100,000	20	2.4	.92	1967	AF04(611)-10830	50	High: Advanced Development ARES
LF_2/LH_2 + Li	750	2,000	60	1.2	.95	1970	NAS 3-11230	52	Low: R & D (see also Ref. 53)

On the basis of the considerations listed below, it is recommended that N_2O_4/MMH or N_2O_4/PAAB-1 be retained as the baseline Spaceplane internal propulsion propellants. Furthermore, this analysis recognizes both propellant combinations as superior (on a SP basis) to either Lox/LH_2 or Lox/Hydrocarbon propellants.

- o SP potential increment
- o Spaceplane M_I (vehicle start mass) penalty
- o Propulsion system weight effects
- o Technology status
- o IR plume detection
- o STS Orbiter compatibility

5.11.4 Auxiliary Propellant Tanks

Relatively late in the program, ALRC was asked to evaluate the possibility of placing auxiliary propellant tanks in the foreward and aft Spaceplane payload bays. Both of these payload bays can be seen in the H/S Spaceplane vehicle layout drawing in Figure 37. Only a very cursory evaluation was possible with the available time. The results of this evaluation are discussed here.

First, a review of possible auxiliary tank propellant utilization schemes were considered. These included:

- o Depleting, and evacuating main propellant tanks (by fluid interface with deep space vacuum) prior to refilling from auxiliary tanks. This would result in the lowest required ΔP between the main and auxiliary propellant tanks, but would also result in the maximum required onboard He.

- o Pumping directly from auxiliary to main propellant tanks. This method would reduce the total He requirements, but could penalize the Spaceplane on a weight, and hence performance, basis depending on how the pumps were driven.

- o Pumping propellants directly from the auxiliary tanks to the PCE. This, in effect, constitutes a pump-fed propulsion system as opposed to the main pressure-fed system. The turbopumps required (high ΔP, low flowrate) are not state-of-the-art. If the required turbomachinery were available, then the use of all pump-fed propulsion for the Spaceplane would probably be recommended.

- o Including an integral cold He pressurization bottle(s) with each auxiliary propellant tank. This would be a relatively heavy system and, hence result in lower SP . It is relatively simple system, however.

- Operating the PCE in a He blowdown mode, instead of regulated He. In this procedure, the main propellant tanks would be operated to propellant depletion in the blowdown mode. The main tanks would then be pump-fed with propellants from the auxiliary tanks. The main tanks would then be depleted in the blowdown mode using the main tank He pressure. The biggest obstacle is the possibility of PCE module chugging instability if the module inlet pressure dropped too low.

A general observation is that there will always be some He weight penalty for the use of auxiliary propellant tanks if He is to pressurize the auxiliary tanks. The only way to eliminate the He weight penalty, at the expense of incurring an onboard energy supply weight penalty due to the increased energy requirement, is to pressurize the tanks in some other way.

Finally, some preliminary calculations of energy requirements for operating a pump-fed PCE were made. These results are listed below.

- Oxidizer pump shaft horsepower required: 5.3
- Fuel pump shaft horsepower required: 5.3
- Electric power required for each pump: 4.63 Kwatts
- Total pump-fed PCE electric power required: 9.25 Kwatts

The total power requirement (9.25 Kwatts) can be compared to the maximum anticipated aero surface power requirement of 42.0 Kwatts (see Table III). The total energy requirement, on the other hand, is very low (0.308 Kw-Hrs) since the pumps would only operate for approximately 2 minutes. The total energy requirement for the Spaceplane aero surfaces has been estimated to be approximately 7.2 Kwhrs.

5.12 TASK 3.4 VEHICLE INTERFACE

Task 3.4 was performed in parallel with Task 3.2 (Propulsion System Interface), described previously. Whereas Task 3.2 deals with all propulsion system interfaces (i.e., with the pilot, with other Spaceplane subsystems, etc.), this task dealt specifically with the identification of propulsion system/Spaceplane vehicle interfaces. These mechanical interfaces are listed and briefly described below.

- Attachment of PCE modules (16) to Spaceplane vehicle structure
- Attachment of reverse thrusters (2) to Spaceplane aft vehicle structure
- Attachment of RCs thrusters (14) to Spaceplane vehicle structure
- Mating of Spaceplane aft end/PCE to Wide Body Centaur
- Routing and attachment of PCE propellant lines (2) inside of Spaceplane vehicle structure
- Routing and attachment of RCS thruster propellant lines (14 pairs) inside of Spaceplane vehicle structure
- Attachment of propellant tank support structure within Spaceplane vehicle structure (2 tanks with 2 supports/tank)
- Location and attachment of He bottle(s) within Spaceplane vehicle structure
- Routing and attachment of He lines and within Spaceplane vehicle structure
- Location and attachment of He heat exchangers (2) within aft cone of Spaceplane vehicle structure
- Routing and attachment of He lines (2) (from He bottle(s) directly to propellant tank isolation valves) to Spaceplane vehicle structure
- Location and attachment of propellant toroidal manifolds (2) within Spaceplane vehicle structure

The inside diameters for the lines included in this list were identified in Table XI.

The preliminary design of several of these interfaces has been acomplished during the completion of Tasks 3.1 (Propulsion System Installation) and 3.8 (Selected Design). The results are also documented in ALRC Drawing 1195445, which has been reproduced in Figures 51 through 59. In general, the routing of propellant and He lines was a vehicle integration problem addressed by H/S. The mating of the Wide Body Centaur (WBC) to the Spaceplane aft end was also addressed by H/S.

5.13 TASK 3.5 MAINTENANCE AND REFUELING

The objective of this task was to discuss in-situ refurbishment and/or replacement (R&R) of the four major propulsion subsystems (i.e., PCE, RCS, pressurization and tanks). Refueling provisions and auxiliary drop tanks were to be described. Any vehicle requirements thus affected were also to be identified.

One of the major advantages of the PCE concept is its inherent ease of maintenance. Because of its modularity (i.e., multiple PCE modules and multiple He bottles) and its annular geometry, it will be easy to replace selected components.

5.13.1 R&R of Spaceplance Propulsion Major Components

The R&R requirement for Spaceplane propulsion major components has been evaluated on the basis of two important considerations:

- o Ground only R&R vs. ground and on-orbit R&R
- o Postfire R&R vs. prefire R&R (or mission start vs. mission completed R&R)

The major components to be evaluated for R&R activities, on the basis of these considerations, include:

- o PCE modules
- o Propellant tanks and propellant control devices (elastomeric diaphragms or bladders)
- o He bottles and other pressurization subsystem components (i.e., He manifolds and regulators)
- o RCS thrusters

All PCE modules are located around the periphery of the vehicle, which ideally permits easy access to the modules. The propellant lines would then pass through vehicle structure cutouts or ports. In practice, however, the PCE module mounting bracket and valve assemply, which are relatively cool (300-400^{o}F), should be thermally separated from the hot (approx. 2000^{o}F) chamber and nozzle to reduce heat soakback to the valves and vehicle structure at engine shutdown. The impact of this requirement on the ideal PCE configuration is illustrated conceptually in Figure 64.

This lessens, to some degree, the ease of R&R of PCE modules either on-orbit or on the ground. One potential solution to this problem is to place access doors in the vehicle wall close to the PCE mounting structure. For the Spaceplane, however, this is not advisable since the thermal protection system (TPS), consisting of STS-type tiles and/or a coating of silica phenolic ablative, covers the vehicle surface. Making access doors through these materials could be difficult.

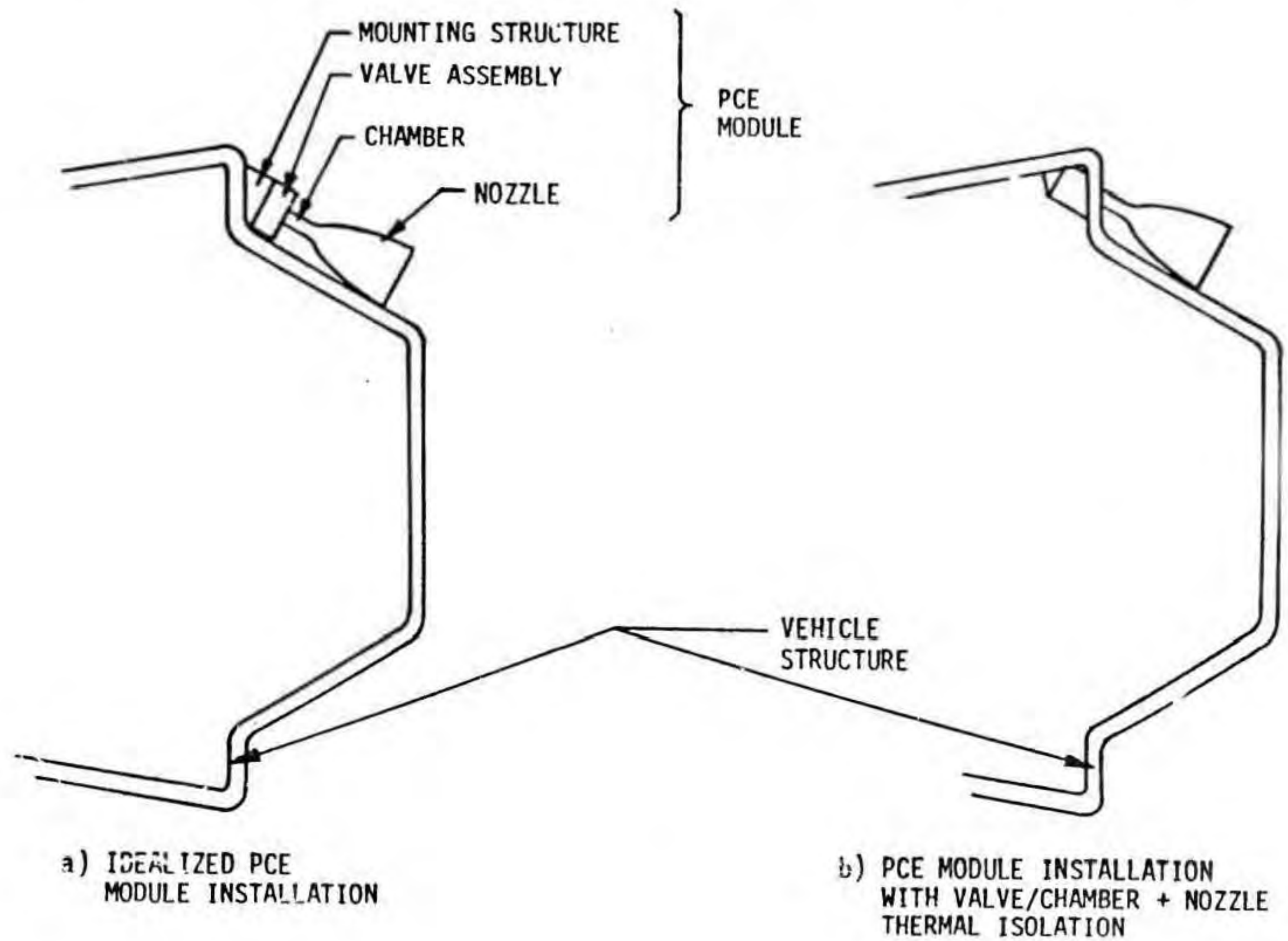


Figure 5-64 Effect of Thermal Isolation of Plug Cluster Engine (PCE) Module Valves on PCE Module Installation

Another alternative is to access the PCE modules from inside the vehicle. The practicality of this alternative depends on at least two other considerations:

- o Can the PCE module nozzle and chamber be removed externally? (The answer is yes if the PCE modules utilize a bolted chamber design.) It is likely that the chamber and nozzle would have to be detached from the valve assembly before the valve assembly could be removed from inside the Spaceplane.

- o Does the arrangement of propellant pressurization bottles (He) and life support equipment behind the pilot allow room to access the PCE modules?

In answer to the first question, it should be noted that another advantage of the bolted chamber design is the fact that rapid removal and replacement of the nozzle and chamber could be important if the total design life of these components is less than that of the Spaceplane design life. The life limitation of the nozzle and chamber is due solely to the eventual degradation and loss of the silicide coating used in the nozzle. This advantage would not be available if an all welded design were used. A further advantage of the bolted flange configuration is the fact that the two flanges, separated by a seal, is an effective conductive heat transfer barrier since the two flanges are not in direct, metal to metal contact. Another alternative would be placement of radiation heat barriers between the valve assembly and nozzle and chamber. These barriers are usually flat reflective mettalic plates placed between the valve assembly and nozzle and chamber so as to reflect the heat being radiated from the hot nozzle back to the nozzle or out to space.

The answer to the second question can only be answered on the basis of the ultimate Spaceplane internal configuration.

In all cases, ease of accessibility to any spaceplane subsystem is more critical in on-orbit R&R activity as opposed to ground R&R activity. This is partly because much on-orbit R&R activity would be done by the pilot or others in spacesuits. Most of the R&R activity (loosening bolts, opening access doors, replacing large and small components, etc.) is much easier on the ground (i.e., not weightless, not wearing bulky gloves, etc.). Some ground R&R activity would be virtually impossible on-orbit, such as working on portions of the Spaceplane interior not big enough to allow for the added volume of a spacesuit.

For these reasons, on-orbit R&R requires the maximum ease or accessibility to those subsystems needing R&R. It should be noted that there are hand tools available, and in development, specifically designed for use in a weightless environment by spacesuited persons. The use of such tools should alleviate some of the difficulties inherent in on-orbit R&R activities.

A final consideration regarding R&R of PCE modules, either on-orbit or the ground, is the fact that the propellant lines and manifolds supplying the PCE modules will always contain N_2O_4, PAAB-1 or MMH. This is a requirement for the quick launch mode of operation discussed in Ref. 1. Ground R&R procedures preclude pre- or postfire examination of propulsion subsystems using these storable propellants without extensive and time-consuming clean and purge operations.

Based on this consideration alone, R&R of PCE modules would only be feasible at ground facilities. Pre-fire R&R, in which the propellant lines and manifolds are not yet filled with propellants, although still hazardous, could be done on the ground or on-orbit. This R&R procedure would not be feasible, however, if rapid launch capability is required. Methods for making on-orbit R&R of PCE modules feasible would probably require additional valves. This added complexity is not warranted by the added capability (i.e., on-orbit R&R of PCE modules). Additionally, the requirement for on-orbit R&R of a faulty PCE module could be eliminated entirely by not using the faulty PCE module or its opposite module during the mission. This would result in a small (1/8) total thrust loss. The faulty PCE module could then be replace on the ground before the next mission.

R&R of full, or partially full, propellant tanks is always very hazardous because of the propellants. Even ground R&R of depleted tanks (residuals and propellant vapors still in tanks) would require extensive drain, purge and clean operations such as those used to desafe a deorbited, land Shuttle orbiter. The extremely close packaging of the various Spaceplane subsystems (i.e., life support, recovery parachute, avionics, propellant pressurization, etc.) makes R&R of the main propellant tanks even more difficult. For these reasons any on-orbit replacement of the main propellant tanks is considered feasible but undesirable. Additionally, the recommended alternative to ground R&R of the tanks is to design them for a useful life at least equal to the vehicle design life. In this way, R&R of the tanks, either on-orbit or on the ground, is never required during their design life.

A requirement for R&R of the baselined propellant control devices (i.e., elastomeric bladders or diaphragms) during the design life of the propellant tanks may necessitate making the propellant tanks easily removed. Again, this procedure would only be done on the ground because of the hazardous nature of the propellants and the close packaging of nearby subsystems. How often such R&R is required depends on:

- The design life of the elastomeric devices compared to the tank design life
- The propellant storage time requirements

Both of these issues still need definition. The ideal solution, of course, would be elastomeric propellant control devices with design lives equal to or greater than that of the propellant tanks. Because these devices are not currently available, ALRC recommends that the development of N_2O_4 - compatible, long life elastomeric diaphragms (or bladders) be considered a technology effort required for successful development of the Spaceplane vehicle. This recommendation is treated separately in the Task 3.7 (Development/Technology) discussion of this report.

The primary considerations relative to R&R of the spherical He pressurization bottles include:

- o The accessibility of the He bottles
- o The He gas storage pressure

In this case, as opposed to propellant tank R&R, the contained medium, He, is relatively benigh. On the other hand, the high He storage pressure (4000 psia) represents a potential hazard during maintenance.

As currently configured, multiple He bottles are connected by one or more high pressure manifolds, which in turn are connected to pressure regulators. Separate He lines also are connected to the pneumatically operated pilot valves on the main propellant valves. For this reason, a single, full He bottle could not be removed without disconnecting all He lines downstream of the associated pressure regulator(s) and pilot valves. This represents a fairly complex procedure, which could be largely avoided by designing He bottles and He distribution system with a design life equal to or greater than the Spaceplane vehicle design life.

R&R of empty He bottle(s), on the other hand, is a much simpler operation, since single bottles can be removed. There appears to be little likelihood that on-orbit R&R of depleted He bottles would ever be required, however, since this could be accomplished on the ground when the Spaceplane would normally return with nearly depleted He bottles.

Any on-orbit R&R of He bottles, or associated hardware, is not recommended. Rather, any indication of pressurization system malfunction (mainly He leakage) during an on-orbit mission start should require a mission abort and return to earth.

On ground R&R of the bottles, on the other hand, is feasible since the pressurization system could be depleted and purged before removing any portions of it. With the storage pressure reduced to atmospheric, single He bottles could be removed, refurbished or replaced without danger. This condition would not likely exist on-orbit since, at mission start, the pressurization system would be charged (to the high storage pressures). Then, at mission end, with a relatively depleted He pressurization system, the Spaceplane could be returned to Earth for He bottle R&R, if required, or otherwise merely refilled for reuse. This scenario would eliminate the need for any on-orbit, full pressure He pressurization subsystem R&R.

Finally, some observations regarding R&R of the RCS thrusters should be made. First of all, the location of the forward RCS thrusters (on the hinge plane bulkhead) as currently configured (see Figure 2), provides easy access with the Spaceplane nose opened up. This makes these RCS thrusters easy to inspect either on the ground or on-orbit. However, as with the PCE modules, R&R of these RCS engines would be hazardous, because the propellant lines and valves supplying the RCS thrusters would always be full of propellants. Of the 8 RCS thrusters in the hinge plane, the forward facing retro thrusters would be the most difficult to access. The other 6 RCS thrusters, which are oriented perpendicular to the vehicle longitudinal axis, are easily accessible. The final configuration of avionics gear in the forward opening could impact this current arrangement.

The RCS thrusters on the aft end of the vehicle (see Figure 4) should also be relatively easy to access. Again, any R&R activity would occur on the ground where a complete purge of the propellants in the RCS thruster supply lines is convenient.

One important observation relative to the above discussion is that many of the on-orbit R&R activities identified as not recommended could become feasible, and in some cases recommended, if a permanent, manned Space Operation Center (SOC) were available for Spaceplane operational support. The present recommendations were based on the availability of Shuttle on-orbit support only.

5.13.2 Refueling

As described in the Task 2.3 (Operation and Control) discussion, both onboard propellant tanks are equipped with fill and drain valves. All He bottles are similarly equipped with fill and vent valves. The location of these valves will determine the relative ease or difficulty of on-orbit refueling. If there is no requirement for on-orbit refueling of either propellant or repressurization of the He bottles, these valves could be located anywhere within the vehicle that is conveninet to propellant and He fill lines. An on-orbit refueling requirement would result in placement of all Spaceplane internal fill, vent and purge ports as close to the vehicle exterior as possible. Additionally, the access door(s) over the fill, vent and purge ports, such as those covering the drogue and secondary parachutes, might be required. These access doors would be located preferably on the aft end of the vehicle to avoid access doors through the TPS.

An additional consideration relative to repressurization of the He bottles is that it must be done slowly to avoid excessive He heating as the He bottle pressure rises. The alternative is rapid filling which would require a He bottle cooling device integrated within the Spaceplane.

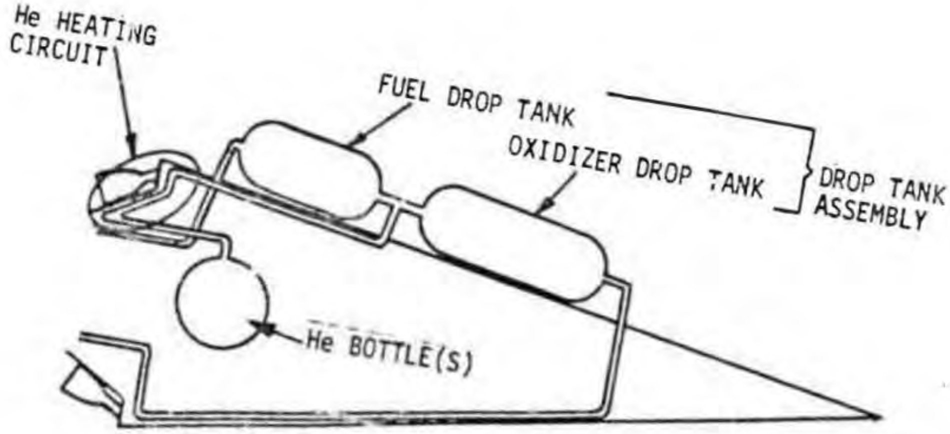
5.13.3 Drop Tanks

Auxiliary drop tanks for added Spaceplane ΔV (SPΔV) would be carried externally to the Spaceplane vehicle. The volume of propellant drop tanks would be determined on the basis of required additional SPΔV.

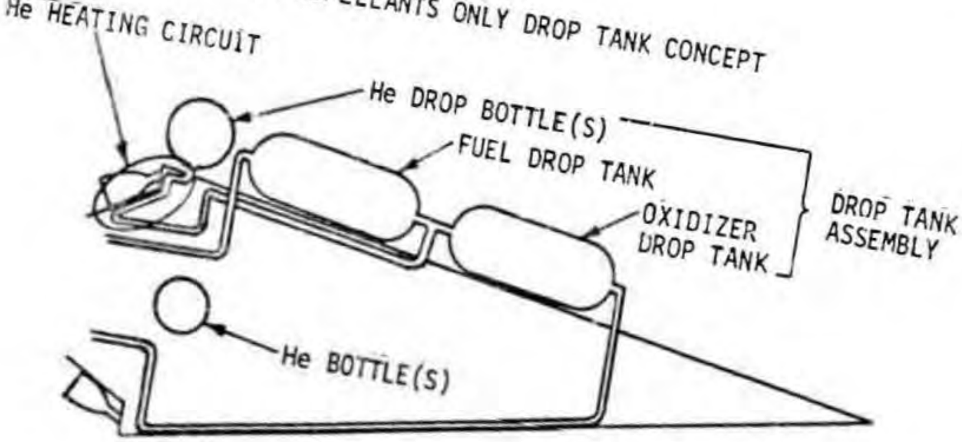
Three alternatives for He pressurization of the drop tank propellants are:

- o Use of Spaceplane internal He bottles
- o Use of separate He bottle(s) integral with the propellant drop tanks and using the Spaceplane He heating circuit
- o Use of separate He bottle(s) integral with the propellant drop tanks (no He heating)

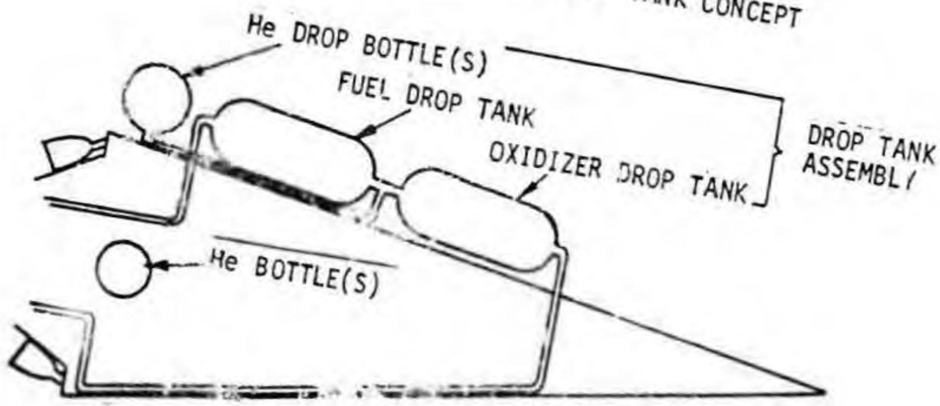
These alternatives are illustrated conceptually in Figure 65.



a) PROPELLANTS ONLY DROP TANK CONCEPT



b) PROPELLANTS & WARMED He DROP TANK CONCEPT



c) PROPELLANTS & COLD He DROP TANK CONCEPT

Figure 5-65 Drop Tank Assembly Propellant He Pressurization Concepts

The first alternative could, depending on the drop tank propellant load, result in severe Spaceplane pressurization subsystem weight penalties due to the increase He requirement. This weight penalty would, in turn, result in some SPΔV penalties. Perhaps more importantly, the already tight packaging constraints of the various Spaceplane subsystems may not allow use of an enlarge He bottle. A resulting lack of drop tank size flexibility would result from selection of this alternative.

The second drop tank assembly concept has an integral He supply, which means the Spaceplane onbaord He requirements are unaffected. However, this concept is somewhat more complex because it carries three fluids (He, fuel and oxidizer) instead of just two. It also requires four drop tank-to-vehicle interfaces (i.e., (1) fuel, (2) oxidizer, (3) cold He, and (4) warmed He), instead of three.

Like the second concept, the third concept required drop tanks to carry fuel, oxidizer and He. By using cold He pressurization, however, the drop tank-to-vehicle He interface, required to provide warmed He, is eliminated. This concept, therefore, requires only two drop tank-tovehicle interfaces: (1) fuel, and (2) oxidizer. There is a weight penalty associated with the use of cold He, however, as opposed to warmed He. This SPΔV penalty is indicated in Table XVIII, which shows SPΔV increment as a functionof drop tank propellant load and use of warmed or cold He propellant pressurization.

The sperical tank radii shown in Table XVIII are included only as a way of presenting the approximate drop tank size. The actual propellant tank portion of the drop tank assemplies would probably be cylindrical. The SP values shown are also based on the use and depletion of the drop tank propellants before any Spaceplane internally stored propellants are used. The current PCE configuration was also assumed (N_2O_4/MMH; Isp = 312.89 seconds; tank pressure = 284 psia; He temp in propellant tanks = 70°F).

The weights of drop tank assemblies using cold or warmed He, as a function of resulting SPΔV increment, have also been plotted in Figure 66.

As both Table XVIII and Figure 66 show, the SPΔV penalty resulting from use of cold He pressurization is small.

All of the drop tank assembly concepts considered would require a drop tank-to-vehicle electrical interface to activate and control valves (propellant and/or He).

This third drop tank assembly concept (propellant + cold He) is recommended as the preferred configuration on the basis of these considerations:

- Only two drop tank-to-vehicle fluid interfaces
- Spaceplane internal He requirement unaffected
- Minor SPΔV penalty for not utilizing Spaceplane He heating circuit

TABLE XVIII

DROP TANK PERFORMANCE SUMMARY

Drop Tank Configuration	Drop Tank Assembly Weights: Dry Weight (lbm)	Wet Weight (lbm)	Propellant Weight (lbm)	Spherical Tank Diameters: N_2O_4 Tank (in.)	MMH Tank (in.)	He Tank (in.)	Space Plane Δv Increment (ft/sec)
Propellant + Warm He	19.0	119.0	100.	13.1	13.1	6.3	169.4
Propellant + Warm He	38.0	238.0	200.	16.5	16.6	7.8	335.0
Propellant + Warm He	114.0	714.0	600.	23.8	23.9	11.3	961.6
Propellant + Warm He	189.9	1,189.9	1,000.	28.3	28.3	13.4	1,537.0
Propellant + Warm He	265.9	1,665.9	1,400.	31.6	31.7	15.0	2,068.2
Propellant + Warm He	341.8	2,141.8	1,800.	34.4	34.4	16.3	2,560.8
Propellant + Warm He	398.8	2,498.8	2,100.	36.2	36.3	17.2	2,907.6
Propellant + Cold He	34.0	134.0	100.	13.1	13.1	10.3	169.0
Propellant + Cold He	68.1	268.1	200.	16.5	16.6	13.0	333.3
Propellant + Cold He	204.1	804.1	600.	23.8	23.9	18.7	947.9
Propellant + Cold He	340.5	1,340.5	1,000.	28.3	28.3	22.1	1,502.4
Propellant + Cold He	476.2	1,876.2	1,400.	31.6	31.7	24.8	2,006.1
Propellant + Cold He	612.9	2,412.9	1,800.	34.4	34.4	26.9	2,465.8
Propellant + Cold He	715.0	2,815.0	2,100.	36.2	36.3	28.4	2,785.6

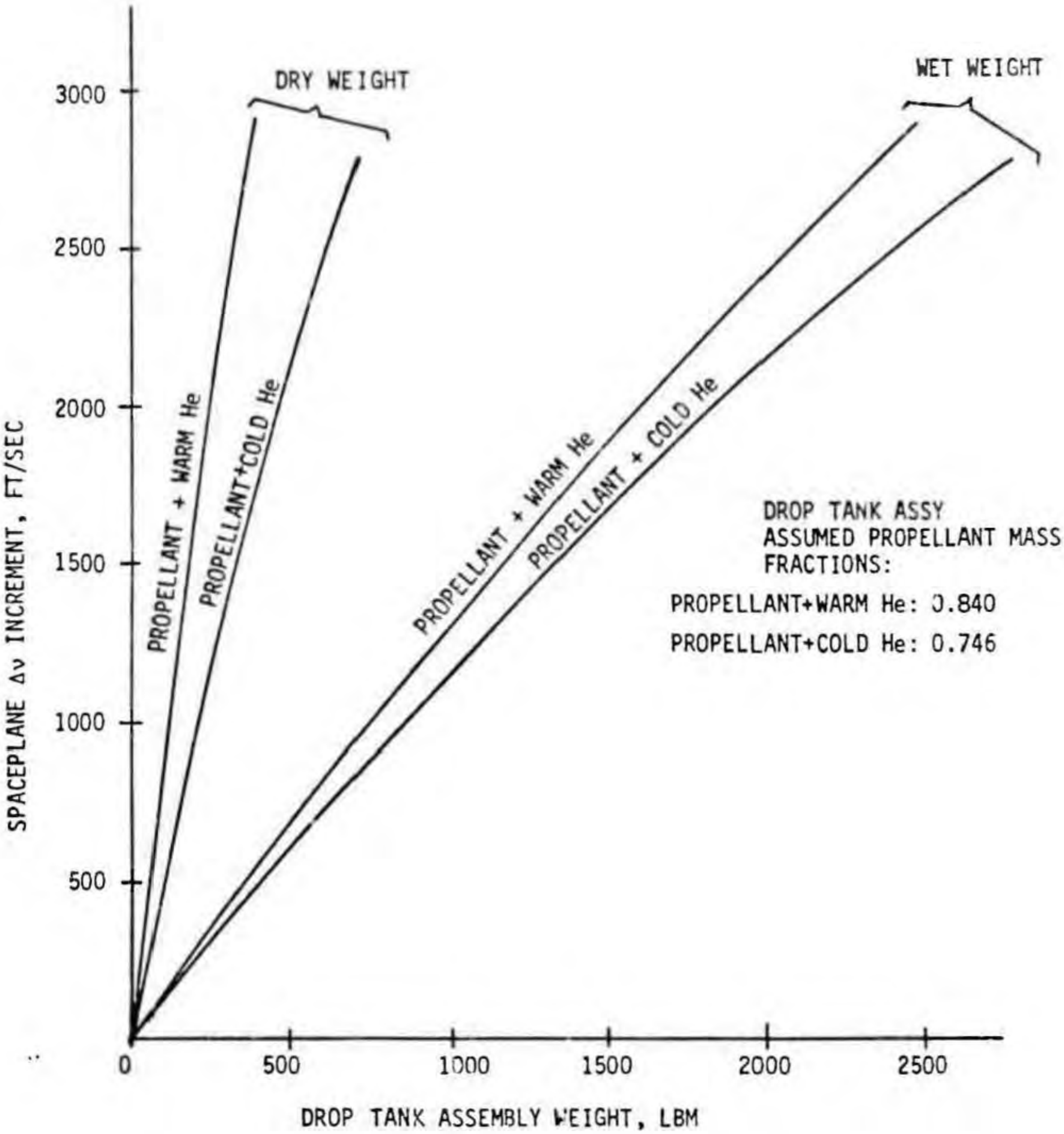


Figure 5-66 Spaceplane Δv Increment vs Drop Tank Assembly Weight

The possibility of other propellant pressurization concepts, such as monopropellant hot gas generators, could be considered. Several of these pressurization concepts and propellant control subsystems are described in References 2 and 3. The drop tank assembly weight could be further reduced, if required, by heating the drop tank He in other ways such as:

- Using a drop tank integral radiant He heat exchanger

- Using a drop tank integral electrical He heater running off of the Spaceplane onboard power supply

All Spaceplane drop tank assembly concepts, in addition to the propellant, electrical or He pressurant gas interfaces, will require: (1) some modification to the Spaceplane internal structure to support the drop tank assembly weight while on the ground, and (2) Spaceplane vehicle structural attach points. This second requirement again represents the problem of attachment to the Spaceplane vehicle through the TPS layers. An evaluation of the STS Orbiter-External Tank (ET) structural interface would be helpful, since it represents a similar situation (i.e., mechanical attachment to a TPS-protected re-entry vehicle).

5.14 TASK 3.6 SAFETY ISSUES

The primary objective of Task 3.6 was to perform a preliminary failure mode analysis of each propulsion subsystem. The completion of this task was assigned to ALRC's Quality Assurance (QA) department. A complete description of the results of their analysis follows.

Safety for the pilot and vehicle is the overriding consideration for the Spaceplane internal propulsion system. In general, specific safety considerations must include adequate strength margins for all system components (including tanks), three mechanically series independent flow control devices to prevent premature firing of any thrust chamber assembly, dual venting paths, redundancy in all critical components where feasible, separation of components, where feasible, to reduce effects of explosions or fire, and the elimination of the possibility of propellant leakage.

The overall purpose of the preliminary hazard analysis described here is to provide a systematic means for early identification of potential safety problem areas, to support design reviews, and to establish a documented baseline to facilitate subsequent design changes.

The specific objective of the analysis was to identify RCS and PCE single point failures which could result in a critical or catastrophic hazard. A critical failure is defined as any single point failure which would result in (1) damage to the equipment or (2) the use of contingency or (3) the use of emergency procedures. A catastrophic failure is defined as any single point failure which would (1) result in personnel injury and/or loss of life, or (2) make the Spaceplane unable to de-orbit (re-enter) or return to the orbiter.

The important assumptions on which this analysis were based are outline below.

- o Categorization of a component by the "worst case" direct effect of failure will define the components criticality.
- o Failure modes that could propagate to other subsystems or the vehicle have been identified.
- o Failure detectability, assuming the availability of telemetry or the crewman responding to monitored displays, is not covered in this study.
- o Structural failures of lines (gross external leaking) are included in this analysis. In most cases, line structural failures can be relate dto upstream/downstream external leakage effects.
- o Redundancy for roll, yaw and pitch has been assumed to be provided by the forward and aft RCS thrusters.

- o PCE module-out capability during on-orbit maneuvers has been assumed. PCE module-out capability during synergistic plane change maneuvers and re-entry is TBD.

- o The analysis considers only those single failure points having a significant potential impact on pilot/vehicle safety.

The results of this preliminary hazard analysis are presented in Table XIX, which identifies the subsystem component, failure mode(s), and classification and the effects on the pilot/vehicle.

Another objective of this task was to evaluate the inherent safety of the baseline RCS and PCE when the Spaceplane is carried as a payload in the Orbiter payload bay. A NASA/Headquarters Document (NHB 1700.7) titled: "Safety Policy and Requirements for Payloads Using the Space Transportation System" (Ref. 6) was reviewed for this purpose. As a result of this review, the following technical requirements extracted from the referenced document, are considered mandatory to meet the payload requirements for the Spaceplane while in the Orbit payload bay.

A few portions of the document extract, which are not considered applicable, have been enclosed in parentheses.

Functions Resulting in Critical Hazards. Any function that could result in a critical hazard must be controlled by two independent inhibits. Monitoring of these inhibits and the requirement to return to a safe condition will be determined on a case-by-case basis; when required, these inhibits may be monitored by the Orbiter flight crew in real time or by the ground in near-real time.

Functions Resulting in Catastrophic Hazards. Any function that could result in a catastrophic hazard must be controlled by three independent inhibits. In-flight monitoring of these inhibits and the capability to return to a safe condition shall be available to the flight crew in the Orbiter in real-time until the function is intentially performed or, in the case of a free-flying payload, until final separation of the payload from the Orbiter. When necessary to assure safe ground operations, monitoring of the inhibits shall be available at the launch site.

Liquid Propellant Propulsion Systems. The premature firing of a liquid propellant propulsion system is a catastrophic hazard. Each propellant delivery system must contain three mechanically independent propellant flow control devices in series that remain closed during all ground and flight phases (except ground servicing) until the deployed payload has reached a safe distance from the Orbiter. For bipropellant systems, these devices must prevent mixing of or any contact between the fuel and oxidizer, as well as prevent expulsion of either or both propellants through the thrust chamber. For monopropellant systems, these devices must prevent propellant expulsion through the thrust chamber. In addition, the operation of a propellant delivery system by opening the mechanical devices shall be controlled by circuitry that contains at least three electrical inhibits. (These electrical inhibits shall be monitored as stated in Paragraph 202.2) The need for monitoring of the mechanical devices will be determined through the safety review process.

TABLE XIX

PRELIMINARY HAZARD ANALYSIS (SPACEPLANE SINGLE POINT FAILURE SUMMARY)

Subsystem	Component	Failure Mode	Failure Classification	Effects
Helium Pressurization	Helium storage tank	Structural failure (Rupture)	Catastrophic	Vehicle damage. Probable loss of pilot/vehicle
	Helium check valve	Structural failure	Catastrophic	Loss of propellant tanks, pressurization supply. Pilot/vehicle stranded in orbit.
	Helium pressure regulator	Fails to open	Casastrophic	Loss of propellant tank pressurization. Pilot/vehicle stranded in orbit.
	Fuel tank shutoff valve	Fails to open	Catastrophic	Unable to pressurize the fuel tank. Stranded pilot/vehicle in orbit.
	Fuel tank check valve	Fails to open	Catastrophic	Unable to pressurize the fuel tank. Possible pilot/vehicle stranded in orbit.
	Oxidizer tank shut-off valve	Same as fuel tank check valve	Catastrophic	
	Oxidizer tank check valve	Same as fuel tank check valve	Catastrophic	
	Heat exchanger	Fails to warm the helium to 70°F	Critical	The rapid expansion of GHe during tank blowdown would result in temperatures as low as 200°F. GHe flowrate would be excessive to maintain the desired regulated pressure with the possibility of stranding the pilot/vehicle in orbit.

TABLE XIX (Cont.)

PRELIMINARY HAZARD ANALYSIS
(SPACEPLANE SINGLE POINT FAILURE SUMMARY)

Subsystem	Component	Failure Mode	Failure Classification	Effects
Helium Pressurization (cont.)	Subsystem including lines separable joints, QD's, etc.	External GHe leakage	Critical	Excessive use of GHe. Loss of propellant tank pressurization, possible stranding of pilot/vehicle in orbit.
Propellant Feed	Propellant tanks	Structural failure	Catastrophic	Fragmentation of propellant tank(s) and damage to Spaceplane. Possible fire or explosion with loss of pilot/vehicle.
	Oxidizer and fuel isolation valves	Fail to open	Catastrophic	Loss of oxidizer or fuel to the plug cluster engine modules. Pilot/vehicle stranded in Orbit.
	Bipropellant Valves	Fail to open	Critical	PCE module redundancy is available for Spaceplane orbit maneuvers but is TBD for re-entry and synergistic plane changes.
	Thrust Chamber assembly	External hot gas leak	Catastrophic	Safety hazard. Loss of TCA or possible damage to adjacent TCA's and/or vehicle. Pilot/vehicle stranded in orbit. Limited redundancy available.
	Subsystem includes lines, separable joints, QD's, etc.	External fuel and/or oxidizer leakage	Catastrophic	Possible pilot/vehicle loss if leak rate is excessive. Fire or explosive hazard possible.
Reverse Thrusters		No thrust (biprop-valve fails to open) or fails to shutdown (biprovalve fails to close)	Critical	Contingency/emergency procedure required.
ACS Thrusters			Catastrophic	Same as thrust chamber assembly. Loss of translation is TBD.

RETRIEVAL OF PAYLOADS. Deployable and free-flying payloads that are to be retrieved shall have the capability to return hazardous systems to a safe condition and shall provide verification to the Orbiter or the ground that safing has been accomplished prior to retrieval while still a safe distance from the Orbiter.

HAZARD DETECTION AND SAFING. The need for hazard detection and safing by the flight crew shall be minimized and will be implemented only when an alternate means of reduction or control of hazardous conditions is not available. When implemented, these functions shall be capable of being tested for propef operations during both ground and flight phases.

EXTENDIBLE PAYLOADS. Any payload which may be operable in a fashion which could prevent closure of the payload bay doors shall be provided with primary and backup methods of clearing the payload bay door envelope. The primary method may be either retraction or jettison and shall be controlled by remote initiation. When the primary method is retraction, the backup technique must be either remote jettison or removal by EVA. (For EVA, see Paragraph 217.)

FAILURE PROPAGATION. The design shall preclude propagation of failures from the payload to the environment outside the payload.

REDUNDANCY SEPARATION. Safety-critical redundant subsystems shall be arranged so that the probability of propagation of failure of one to the other is minimized.

STRUCTURAL.

Structural Design. The structural design shall provide ultimate factors of safety equal to or greater than 1.40 for all normal STS mission phases. Where required to avoid a catastrophic hazard, primary structural and support bracketry shall be designed to preclude failure caused by propagation of pre-existing flaws based on fracture control.

Stress Corrosion. The selection of materials used in the design of payload structures, support bracketry, and mounting hardware shall comply with the stress corrosion requirements of MSFC-SPEC-522. For those applications in which MSFC-SPEC-522 requires the submittal of a materials usage agreement, the data shall be submitted as a waiver request in accordance with JSC 13830.

Pressure Vessels. Pressure vessels shall meet the ASME Boiler and Pressure Vessel Code, Section VIII, Divisions 1 and 2 or MIL-STD-1522. Where weight limitations prohibit meeting the above standards for flight vessels, NSS/HP 1740.1 shall be used as the standard with an ultimate safety factor of 1.5 or greater. Pressure vessels using MIL-STD-1522 or NSS/HP 1740.1 shall also be qualification tested to demonstrate no failure at the design burst pressure level. Pressure vessels using the ASME Code or MIL-STD-1522 shall also be qualification tested to demonstrate a life cycle capability of at least twice the maximum predicted number of operating cycles. Upon request for waiver, pressure vessels of payloads previously flown on expendable launch vehicles which do not meet the above requirements will be considered for single use on the STS provided that they have complied with applicable expendable launch vehicle ground safety requirements without waivers and that certification to this effect is presented. Particular attention shall be given to ensure compatibility of fluids used in cleaning, test, and operation with pressure vessels.

Pressurized Lines and Fittings. Pressurized lines and fittings shall have an ultimate factor of safety equal to or greater than 4.0.

Decompression. Payloads located within manned pressurized volumes designed to withstand decompression or subsequent repressurization shall be capable of tolerating the differential pressure without resulting in a hazard.

MATERIALS.

Hazardous Materials.

General. Hazardous materials shall not be released or ejected in on near the Orbiter. Hazardous fluid systems must contain the fluids after exposure to all STS environments including normal and emergency landing loads. Alternate methods to the above, including use of the Orbiter vent and dump capability, must be negotiated with the STS operator. Particular attention shall be given to ensure storage and use compatibility of metallic and non-metallic materials with hazardous fluids. Guidelines for material selection to avoid incompatibilities are available in JSC 11123, JSC 09604, and JSC 02681.

ELECTRICAL SYSTEMS. Electrical power distribution circuitry shall be designed so that faults internal to the payload do not damage STS circuitry and do not create ignition sources. Where lightning protection is required to avoid a catastrophic hazard, document JSC 07636 shall be used as a guide for payload design.

On the basis of the preliminary hazard analysis and safety evaluation of the Spaceplane as an STS payload, it has been concluded that critical and catastrophic single failure points exist for the failure to open, rupture, excessive external leakage and failure to close biprop valves. However, all of these modes of failure can be circumvented by redundancy or reduced to an "Acceptable Risk Level".

The following recommendations are also made on the basis of the preliminary hazard analysis and safety evaluation of the Spaceplane as an STS payload:

- Rational for acceptability of all single failure points should be established as an extension of this preliminary safety study. This would provide guidelines for designers during the detail design phase of the program. Two important parts of this study would include a redundance study; and a failure detectability analysis to determine single failure point classifications. This classification would establish the following items:

(a) Items for which design action is recommended.

(b) Items for which additional in-flight and ground procedures are recommended.

(c) Items for which (1) analysis and historical test data supports acceptability and (2) adequate procedures exist to minimize or eliminate the effect of a failure mode occurrence.

(d) Items which (1) possess passive equipment characteristics, (2) possess adequate safety margins and (3) are acceptable on the basis of historical test results.

- The next phase of the hazard analysis should also include the safety problems associated with:

  (a) Delivery of the Spaceplane to low earth orbit using the Space Transportation System.

  (b) Delivery of the Spaceplane to low earth orbit using an expendable launch vehicle.

  (c) Return to earth (re-entry) of the Spaceplane under its own power.

- Plan and provide a program organized and conducted to provide the detailed information required to fully access the safety of the Spaceplane as an STS payload.

- Identify the Spaceplane participant(s) responsible for providing the required safety inhibits.

- Define the instrumentation required for verification to the Orbiter or the ground that Spaceplane safing has been accomplished.

It is recognized that the RCS and PCE conceptual designs do not currently provide the detail required to fully assess the safety of the Spaceplane as an STS payload or during launch, synergistic plane change or de-orbit maneuvers. However, several improvements to the baseline controls configuration (Ref. 4) were suggested as a result of the preliminary hazard analysis. These suggested changes enhance both the safety aspects of the Spaceplane internal propulsion system as well as the operational aspects (system weight, complexity, etc.).

- o Use single flowmeters instead of redundant flowmeters (currently the NII launch vehicle uses single flowmeters).

- o Replace the FVV (Fuel Vent Valve) and OVV (Ox Vent Valve) with a single pressure relief valve downstream of the He regulator(s).

- o Use a single fill and vent valve manifolded to all of the He bottles. A quick disconnect would still be located in series with the fill and vent valve providing redundancy.

- o Use a more reliable solenoid and pressure switch combination in place of pressure regulating valve for a very small weight penalty (the propellant tanks must be capable of withstanding peak modulated pressure).

Several other suggestions relate directly to the Spaceplane electrical control system which (1) must be manrated, and (2) incorporates elements of the onboard propulsion system. These suggestions are listed below.

- o Electrical connectors, plugs and receptacles should be mechanically keyed to prevent incorrect connection.

- o Electrical subsystems should include checkout test points which will permit normal subsystem checkout tests to be made without disconnecting electrical connectors which are normally connected in flight.

- o The control system should be designed to sustain a failure of a single item without loss of life or vehicle.

- o Redundant wiring should not be routed in the same bundle or through the same connector.

- o Electrical circuits should not be routed through adjacent pins of an electrical connector if a short circuit between them would constitute a single failure that would cause loss of the pilot.

Finally, one important observation, relative to Spaceplane safety in general, should be reemphasized: the reliability/safety requirements for a reusable, manned military spacecraft boosted by an expendable launch vehicle could, and most likely will, be much different than the corresponding requirements for a Shuttle boosted, reusable, manned military spacecraft. This fact should be recognized early in the total Spaceplane program, especially where plans are developed for designing both a test vehicle(s) and operational vehicles, and selecting a launch mode.

The preliminary safety analysis reported herein, although dealing primarily with only the Spaceplane internal propulsion subsystem, has identified many components and subsystems whose design will be influenced strongly by the reliability/safety requirements imposed on the Spaceplane. The Air Force should begin defining specific reliability/safety requirements for the Spaceplane to avoid potentially severe design problems that could otherwise occur later in a development program.

5.15 TASK 3.7 DEVELOPMENT/TECHNOLOGY ISSUES

The objective of this task was to identify any development and/or technology requirements, with justification, necessary for the successful development of the baseline Spaceplane onboard propulsion system. Three such development and/or technology requirements and the associated risks and reasons necessary are described below.

5.15.1 Long-Life N_2O_4 - Compatible Elastomeric Diaphragm (Bladder)

A long-life (50 to 100 missions x 24 hour/mission) N_2O_4 - compatible, elastometric diaphragm, or bladder, is required for the Spaceplane internal propulsion N_2O_4 spherical propellant tank.

MMH (and PAAB-1) compatible elastometric bladders and diaphragms are considered state-of-the-art and thus do not require any significant technology development. Some examples of propulsion systems using long-life (months to years), MMH (and PAAB-1) compatible elastometric bladders or diaphragms would include:

- ERTS orbit-adjust subsystem (OAS)
- Atmospheric Explorer OAS
- Intelsat III
- Pioneer 10 and 11
- Gemini/OAMS
- Mariner '69/RCS
- Titan III Transtage/RCS
- Apollo LEM/RCS

The elastomeric diaphragm, or bladder, concept has been recommended primarily because all of the state-of-the-art (SOA) alternatives are excessively heavy and/or large and/or the wrong shape for the Spaceplane vehicle. For example, piston tanks are heavy and require cylindrical tanks. Other SOA alternatives cannot be used because they do not provide the propellant control that is required to provide (1) smooth PCE firing (i.e., liquid propellants, instead of propellant vapors, at the PCE module inlet) and/or (2) elimination of propellant sloshing and resulting vehicle movement during precise rendezvous and docking maneuvers. SOA alternatives in this category include:

- Screen assemblies
- Start tanks
- Hoppers

The elastomeric diaphragm concept will ideally overcome all of the deficiencies inherent in the SOA alternatives. The primary goal of technology program would be to develop (or find if already developed) an elastromeric material which would be usable as a diaphragm or bladder and, more importantly, be compatible with N_2O_4 for long term use. Several potential elastomers have already been identified by ALRC. One or more of these materials should now be selected as the subject of technology program to determine whether it is, or is not, adequate for the Spaceplane application.

5.15.2 Modification of ALRC 100 LBF RCS Thruster To Provide 188 LBF Thrust

This requirement is a development issue and not a technology issue. In other words, the risk involved in making this modification is considered very low for the reasons outlined here.

This modification is required to provide a Spaceplane total thrust of approximately 3000 lbF with 16 PCE modules (i.e., 16 x 188 lbF = approx. 3000 lbF). The chamber pressure of 142 psia was selected on the basis of the PCE optimization which is described in the Task 3.8 (Selected Design) discussion.

First, the needed thruster chamber pressure is the same or less than that of the existing ALRC 100 lbF RCS thruster. In general, a lower P_c and /or a greater thrust level will result in a less stringent chamber cooling requirement. In this case, less film cooling will be required.

Secondly, ALRC's platelet injector manufacturing technology allows the rapid and low cost enlargement of the existing 100lbF thruster injector. This enlargement process, including the photographic enlargement/reduction and cropping techniques, was discussed at the Spaceplane Kickoff Meeting held at Aerospace Corporation on 23 and 24 July 1981.

Thirdly, the existing valve assemblies for the 100lbF RCS thruster can accommodate the flowrates required to provide 188 lbF with no modification.

Finally, the manufacture of the chamber and nozzle, with either a bolted or all welded design, for a thrust level of 188 lbF presents no new technology requirements. In fact, the increased size of these components could benefit their manufacture.

5.15.3 Modification of ALRC's 100LBF RCS Thruster or 5 LBF RCS Thruster to Provide 15 LBF Thrust

This required modification is a result of the RCS definition task, also described in the Task 2.4 (RCS Definition) discussion, which identified an RCS module of 15 lbF thrust. This modification, like that of the ALRC 100 lbF RCS thruster, is a development requirement and not a new technology requirement. The reasons for considering it a development requirement, rather than a technology requirement, are outlined below.

First, the P_c is less than the ALRC 5 lbF RCS thruster which results, as with the 100 lbF RCS thruster modification, in a lower film cooling requirement. A more severe film cooling requirement would result, however, if the 100 lbF RCS thruster were modified to provide only 15 lbF of thrust, since the P_c would be constant. This consideration indicates that the ALRC 5 lbF RCS thruster should be scaled up (to a higher thrust and lower P_c) rather than scaling down the ALRC 100 lbF RCS thruster (to a low thrust at same P_c).

Secondly, the platelet injectors, utilized in both the 5 lbF and 100 lbF RCS thruster are, as before, easily enlarged or reduced/cropped as required.

Thirdly, the existing valve assembly of the ALRC 5 lbF RCS thruster can accommodate the flowrates required to provide 15 lbF with no modifications. The 100 lbF RCS thruster valve assembly may not provide adequate delta-P at the lower flowrates. This is another reason to modify the 5 lbF RCS thruster to provide 15 lbF of thrust rather than modify the 100 lbF RCS thruster.

Finally, as with modification of the 100 lbF RCS thruster, the manufacture of the chmaber and nozzle, with either a bolted or all-welded design, for a 15 lbF thrust level presents no new technology requirements.

5.16 TASK 3.8 SELECTED DESIGN

The primary objective of this task was to select, from parametric propulsion system data, an optimum Spaceplane onboard propulsion system design. The justification (optimization procedure) for this design selection was to be described. A layout drawing of this design was to be made. A written description (preliminary engine specification) of this design, including performance and operations information, was to be provided.

5.16.1 Propulsion System Optimization Procedure

The onboard propulsion system optimization procedure used was based on the assumption that Spaceplane delta-V (SP delta-V) was to be maximized by varying these 4 PCE parameters:

- e_m = PCE module nozzle area ratio
- F_m = PCE module thrust
- N = Number of PCE modules
- P_c = PCE module chamber pressure

The computer program SP delta-V (for Spaceplane Version 1) was ideal for performing this optimization because these 4 major PCE design parameters are program inputs, while SP delta-V is a program output. The optimization task then was to vary these 4 PCE design parameters for each SPV1 execution to find the combination of PCE design parameters which resulted in the maximum SP delta-V. This somewhat iterative procedure was particularly appropriate, since SPV1 is a terminal-operated, interactive computer program.

Superimposed on the resulting PCE parametric envelope data were the PCE diameter and length constraints of 43 and 12 inches, respectively. These values were the diameter and length of the PCE selected for the initial Spaceplane vehicle integration effort in Task 2.1 (Performance). It was assumed that although the total thrust level, or number of PCE modules, could change, that the overall PCE envelope (i.e., length and diameter) would not change from these initial values established in Task 2.1. Other constraints, such as thrust level per PCE module, were also included in the optimization procedure.

Some explanation regarding the optimum chamber pressure for any pressure-fed propulsion system, including the Spaceplane onboard propulsion system, will also be helpful. In general, the optimum pressure-fed propulsion system is one which results in the highest vehicle delta-V (or payload for a given gross vehicle weight). Specifically, this was the assumed criteria for optimization of the Spaceplane onboard propulsion system. The two important trends that determine the chamber pressure (P_c) of this optimum pressure-fed system are:

1. For a given thrust level and nozzle area ratio, as P_c increase, Isp also increases. Since

   $$\text{delta-V} = g_c \text{ Isp } \ln (W_{IGN}/W_{BO} \qquad (1)$$

   where

   g_c = acceleration of gravity, 32.174 fr/sec^2,
   W_{IGN} = gross vehicle weight (i.e., loaded with propellants) and
   W_{BO} = vehicle burnout weight (i.e., empty of propellants,

   therefore, delta-V is directly proportional to delivered Isp. Therefore, not considering other effects, delta-V tends to increase with increasing P_c.

2. As P_c increases, the weight of the propellant pressurization subsystem (he, He bottles, etc.) and propellant tanks also increases. Since these subsystem weights increase both W_{IGN} and W_{BO} by the same amount, the ratio of W_{IGN}/W_BO decreases. In this case equation (1) shows that, not considering other effects, delta-V tends to decrease with increasing P_c.

The effect of these two trends is to define an optimum P_c which results in the maximum vehicle delta-V. The computer program SPV1 is again ideal for performing this P_c optimization because it takes into account both of these trends.

Finally, it should be noted that the results of this optimization procedure can be altered if other constraints are imposed, such as a limited W_{IGN}, limited vehicle envelope, high gloss trajectory, etc.

5.16.2 N_2O_4/MMH Propulsion System Optimization

Some of the important assumptions made to perform this optimization are listed below.

- The propellant tank pressure was defined as twice the PCE module chamber pressure. This assumption was based on comparison with existing pressure-fed, storable, man-rated rocket engines (e.g., OME).

- A mixture ratio of 1.65 for the N_2O_4/MMH propellant combination was assumed based on the results of the MR optimization described previously (Task 2.1 Performance).

- o The total Spaceplane burnout (or dry) weight, not including the weight of the PCE, RCS, PS, and propellant tanks, was 4500 lbM. (This was a conservative value compared to the 4155 lbM value recommended subsequently by H/S. However, this assumption did not affect the results of the optimization. For the definition of the baseline N_2O_4/MMh propulsion system SPV1 was rerun at the optimum PCE configuration with the 4155 lbM value to determine the predicted SP delta-V.)

- o Spherical propellant tanks were assumed.

- o 4000 psia HE storage pressure was assumed.

- o 1400 lbM total propellant load was assumed.

- o The HE was assumed to enter the propellant tanks at approximately 70°F. This, in turn, presupposes the use of a heat exchanger somewhere in the HE circuit.

- o The parametric values of the four major PCE design parameters were:
  - N (number of PCE modules): 8, 12, 16, and 20
  - Fm (PCE module thrust, lbF): 375, 250, 188, and 150
  - e_m (PCE module nozzle area ratio): 40, 100, 150, and 200
  - P_c (PCE module chamber pressure, psia): 142, 190, 250, and 300

  N and Fm were always considered a pair of parameters which, multiplied, gave a total PCE thrust of approximately 3000 lbF.

- o A total PCE outside diameter (D_{PCE}) constraint of 43.5 in. (corresponding to the previous baseline PCE) was assumed.

- o A total PCE length (L_{PCE}) constraint of 12 to 14 in. was also assumed. (This corresponds closely to the previous baseline PCE length of 12.3 in.)

The resulting propulsion system parametric weight, envelope and performance (SP delta-V) data is shown in Figures 67 through 74. Figures 67 throught 70 show SP delta-V vs. D_{PCE}; Figure 71 through 74 show SP delta-V vs. L_{PCE}. The D_{PCE} and L_{PCE} constraints (43.5 in. and 12 to 14 in., respectively) are superimposed on these figures. Again, it noted that the SP delta-V values shown in these figures are for a vehicle burnout weight (excluding the PCE, RCS, PS and propellant tanks) of 4500 lbM. For the perferred H/S value of 4155 lbM, all of these SP delta-V values should be increased by about 200 ft/sec. The important trends resulting from this optimization can be summarized as follows:

- o SP delta-V improves with decreasing P_c (at least down to P_c = 142 psia)

- o SP delta-V improves with increasing Fm (decreasing N)
- o SP delta-V improves with increasing e_M
- o SP delta-V improves with increasing L_{PCE}
- o SP delta-V improves with increasing D_{PCE}

Another practical constraint, in addition to the D_{PCE} and L_{PCE} limitations, was that 375 and 250 lbF for Fm (i.e. N = 8 and 12) really represented new PCE modules. Fm values of 150 or 188 lbF, on the other hand, represent only minor modifications to ALRC's existing 100 lbF RCS thruster discussed previously. Therefore, the N = 8 and N = 12 cases were eliminated. Since the N = 16 cases provided higher SP delta-V values than the N = 20 cases, N = 16 was selected as the new baseline value.

Figure 75 shows L_{PCE} vs. D_{PCE} for the N = 16 cases. It illustrates that the L_{PCE} and D_{PCE} constraints could not be met simultaneously. The choice was to have either an excessive D_{PCE} or an excessive L_{PCE}. Since the D_{PCE} constraint was considered more crucial than the L_{PCE} constraint, a slightly longer (14.5 in. vs. 12 in. previously new PCE baseline was chosen.

Figure 76, a plot of L_{PCE} and D_{PCE} vs. e_m, was used to determine an approximate e_m optimum since the actual e_m optimum was an intermediate value of those used in the opttimization (i.e., 40, 100, 150, and 200). This value, as shown in Figure 80, was approximately 52.

The resulting preliminary optimum N_2O_4/MMH onboard propulsion system had the following basic characteristics:

o	N	= 16 (vs. N = 20 previously)
o	Fm	= 188 lbF (vs. Fm = 150 lbF previously)
o	e_m	= approximately 52
o	P_c	= 142 psia
o	D_{PCE}	= 43.26 in. (vs. 43.5 in. previously)
o	L_{PCE}	= 14.15 in. (vs. 12.3 in. previously)

A final task of the N_2O_4/MMH onboard propulsion system optimization was to determine if the actual optimum P_c could be less than 142 psia, since the parametric P_c range considered was 142 psia to 300 psia. To do this, SPV1 was run at several design points corresponding to P_c values of 100, 80, and 40 psia. The PCE module nozzle area ratios were selected to result in PCE diameters of approximately 43.3 inches. At the low PCE module chamber pressures (40 and 80 psia) this corresponded to nozzle area ratios as low as 10 and 20. A resulting plot of SP delta-V vs. PCE diameter for the different P_c's is shown in Figure 77, with the PCE diameter constraint superimposed. A plot of PCE diameter vs. PCE module nozzle area ratio, with the PCE diameter superimposed, is shown in Figure 78. From these two figures a cross plot showing SP delta-V vs. PCE module chamber pressure, t the PCE diameter of 43.3 inches, was made. This plot is shown in Figure 79. This figure shows that a P_c of 142 psia is optimum for the N_2O_4/MMH onboard propulsion system. It was assumed that the N_2O_4/MMH RCS would also operate at this pressure (142 psia).

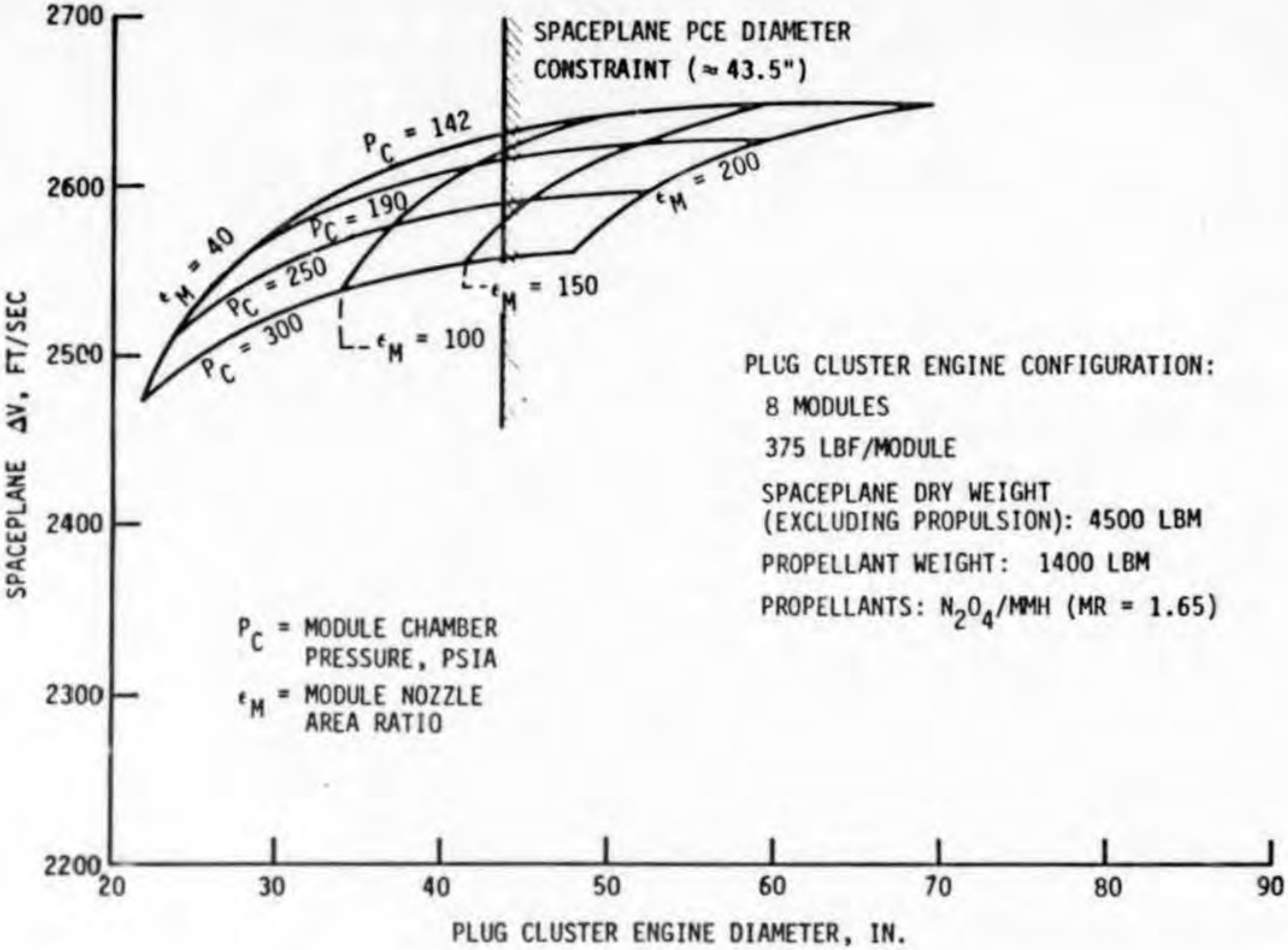


Figure 5-67 Spaceplane Δv vs Plug Cluster Engine Diameter (N = 8)

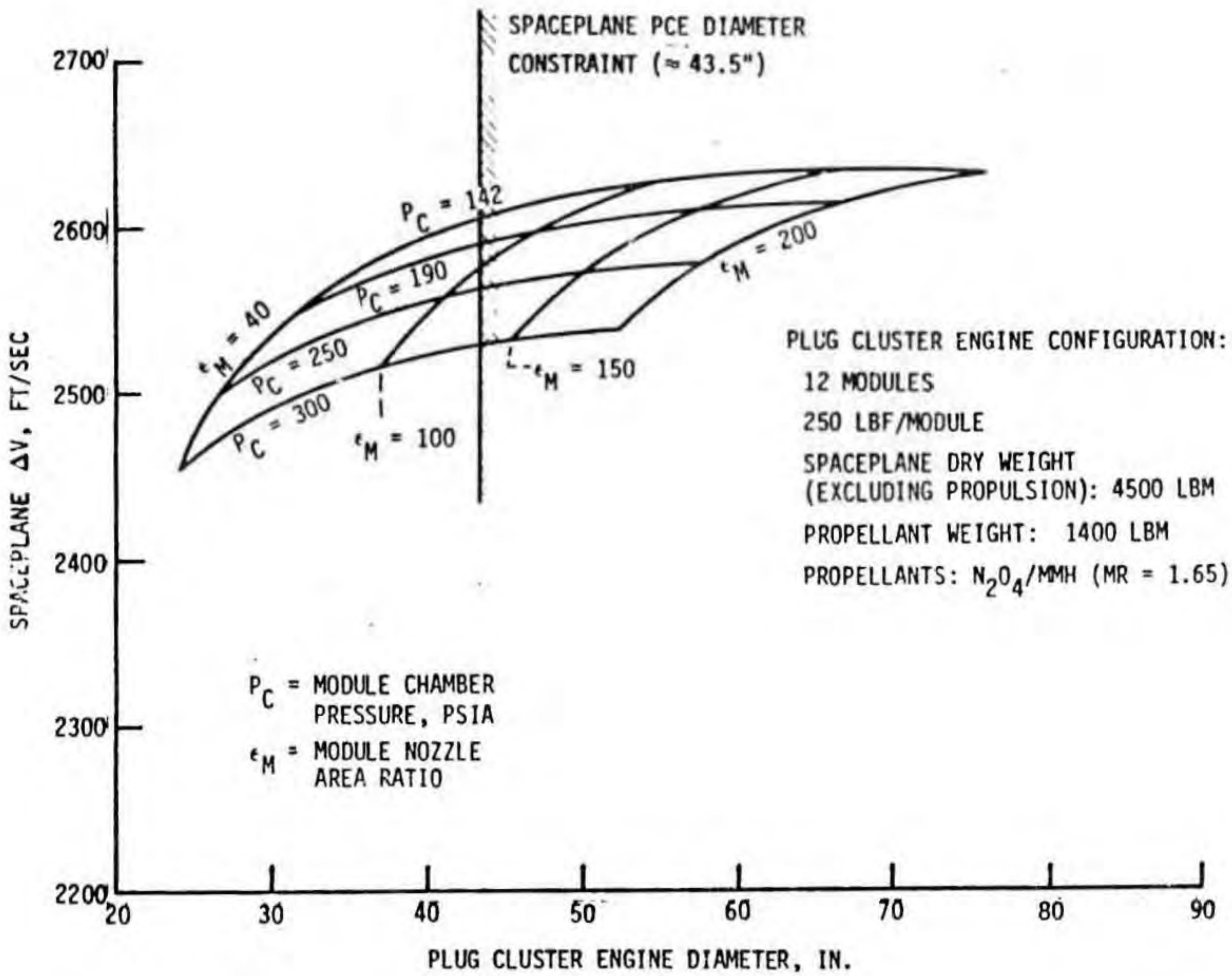


Figure 5-68 Spaceplane Δv vs Plug Cluster Engine Diameter (N = 12)

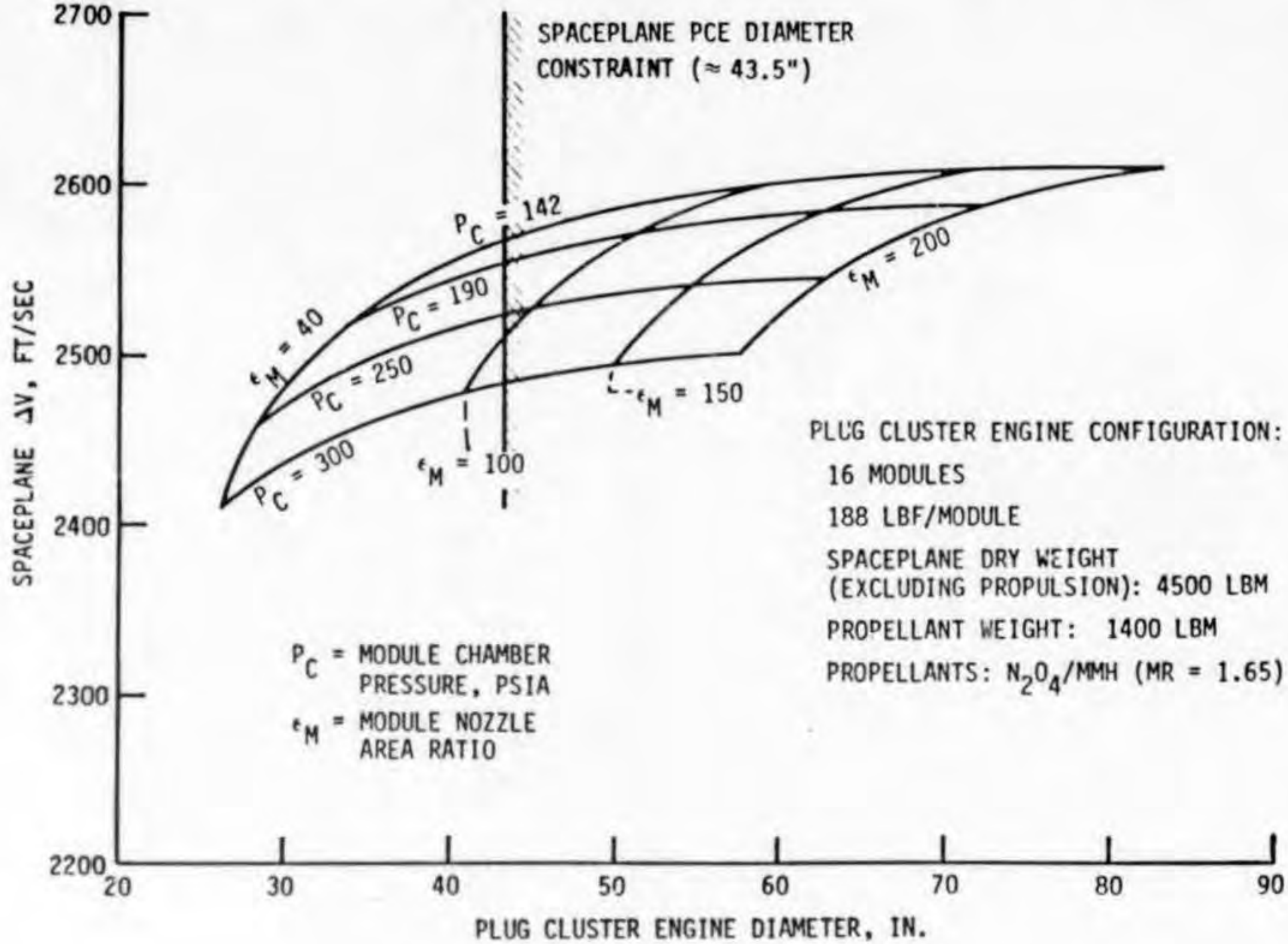


Figure 5-69 Spaceplane Δv vs Plug Cluster Engine Diameter (N = 16)

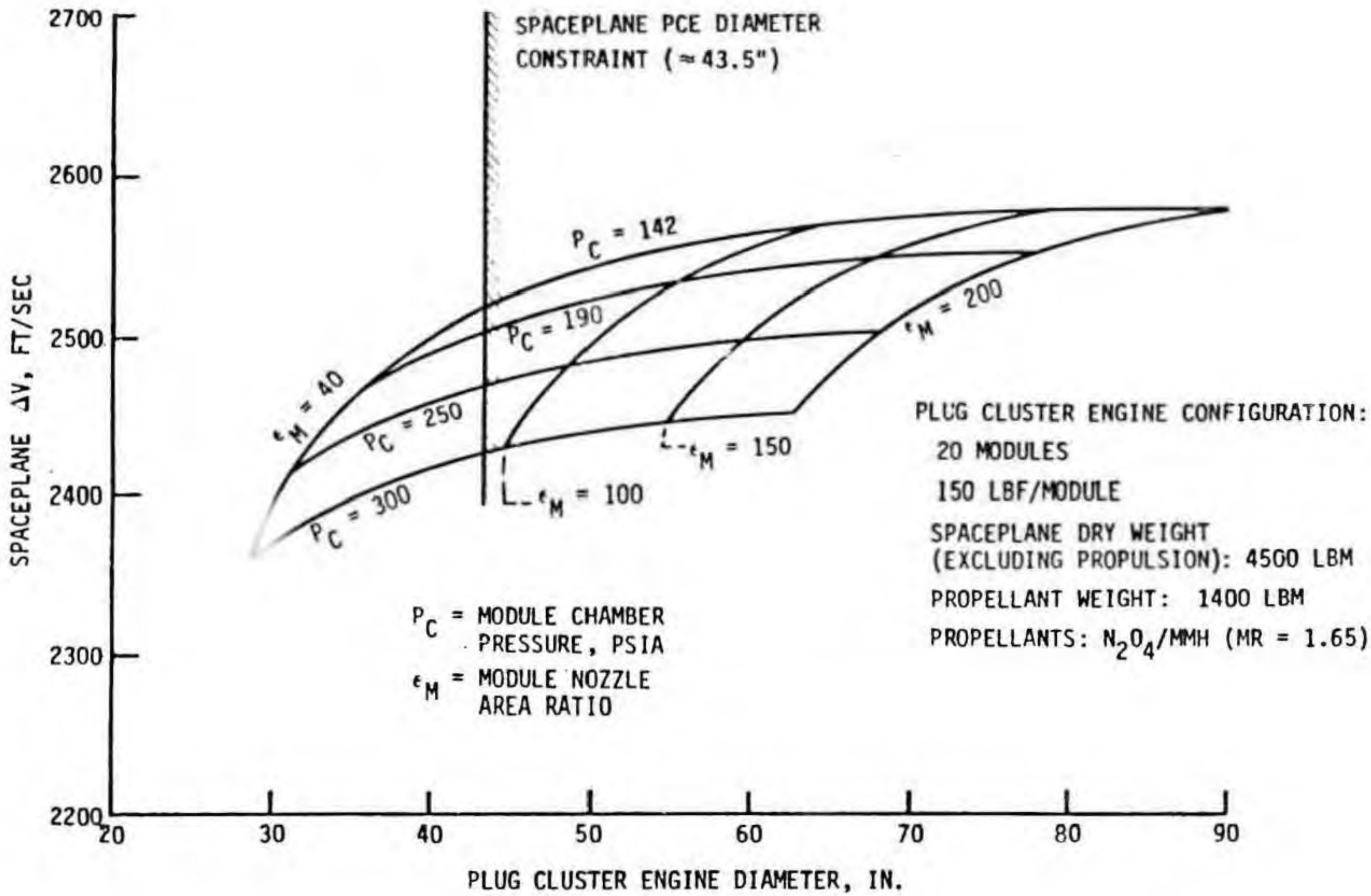


Figure 5-70 **Spaceplane Δν vs Plug Cluster Engine Diameter (N = 20)**

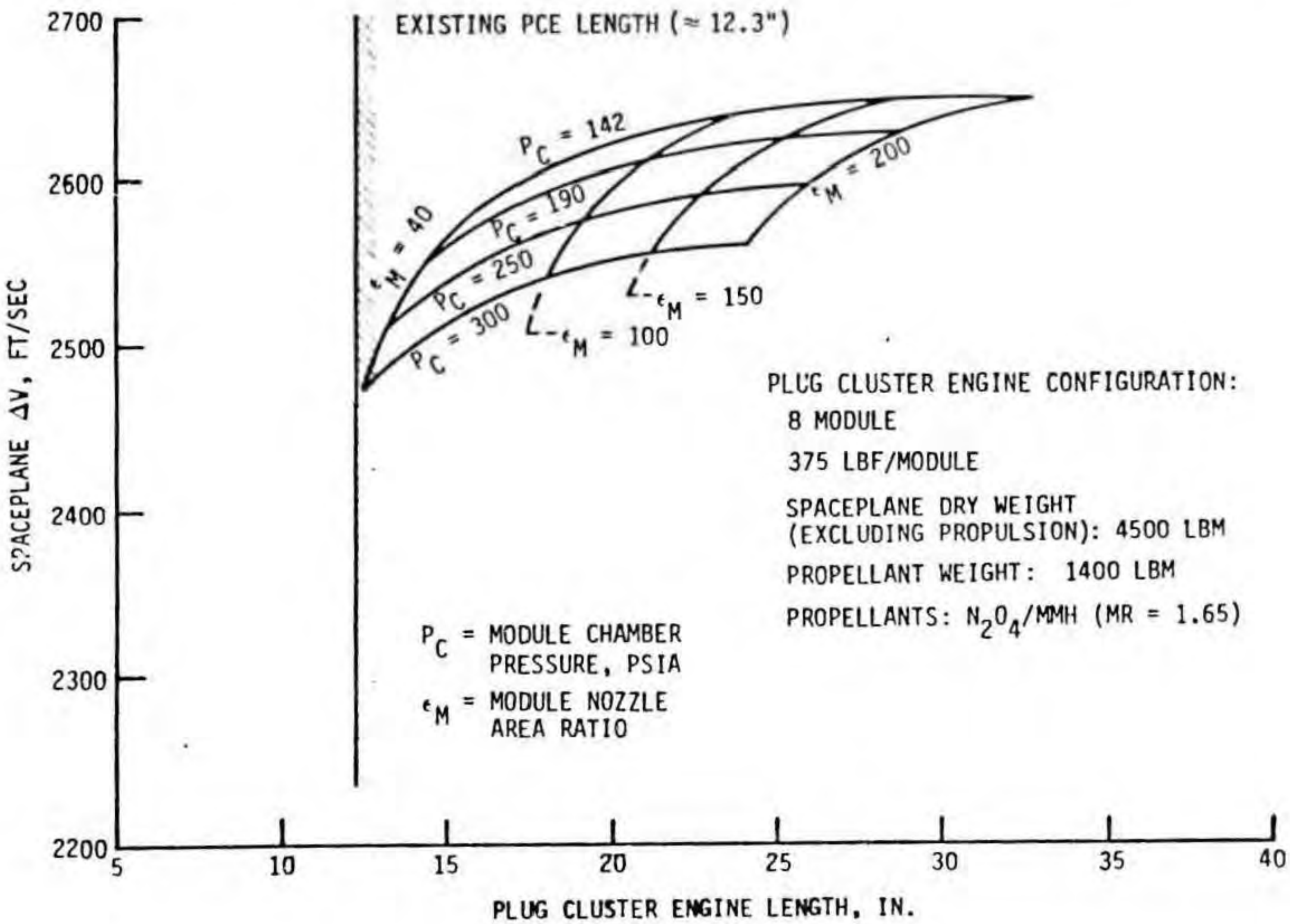


Figure 5-71 Spaceplane Δv vs Plug Cluster Engine Length (N = 8)

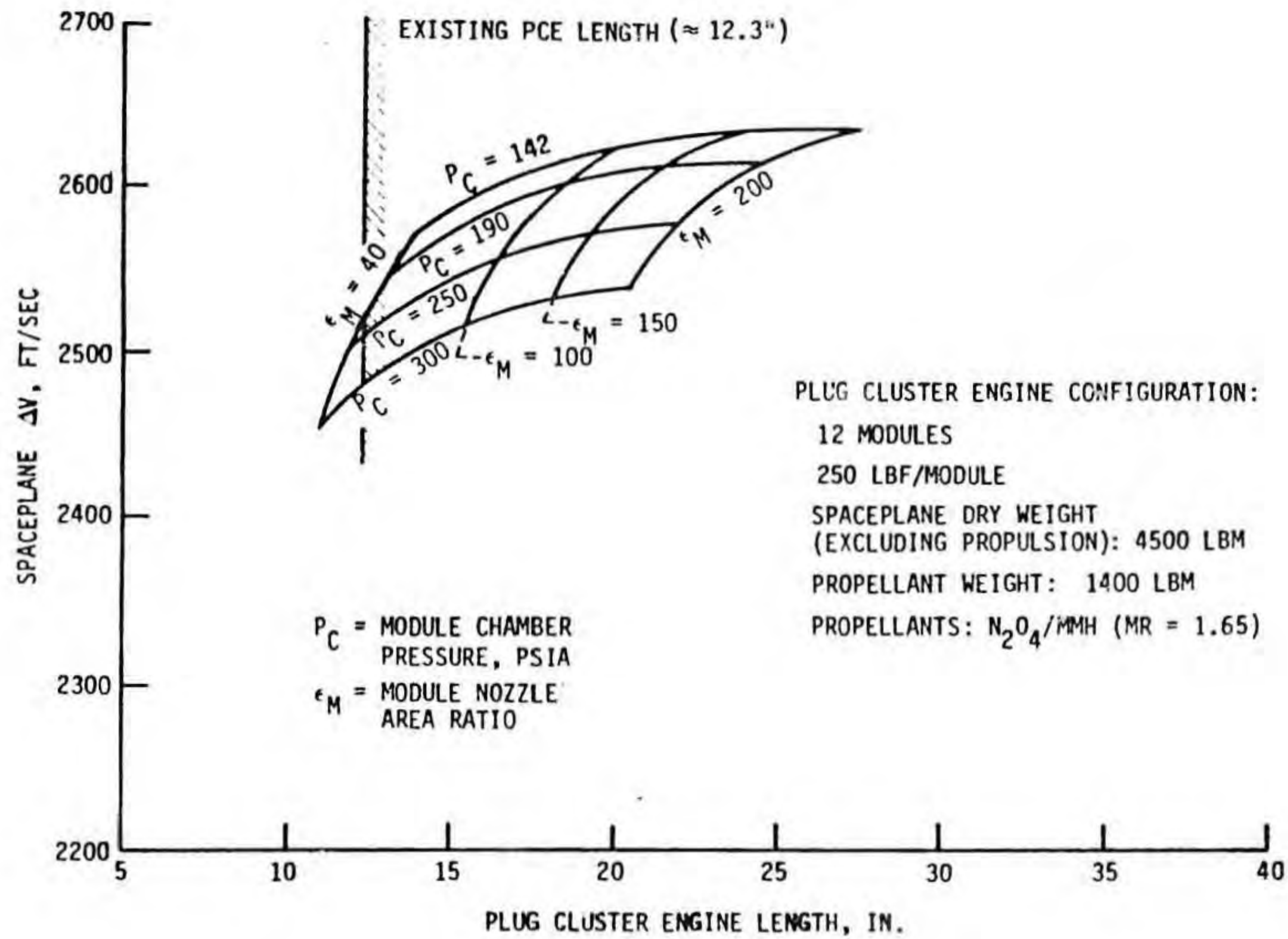


Figure 5-72 Spaceplane Δv vs Plug Cluster Engine Length (N = 12)

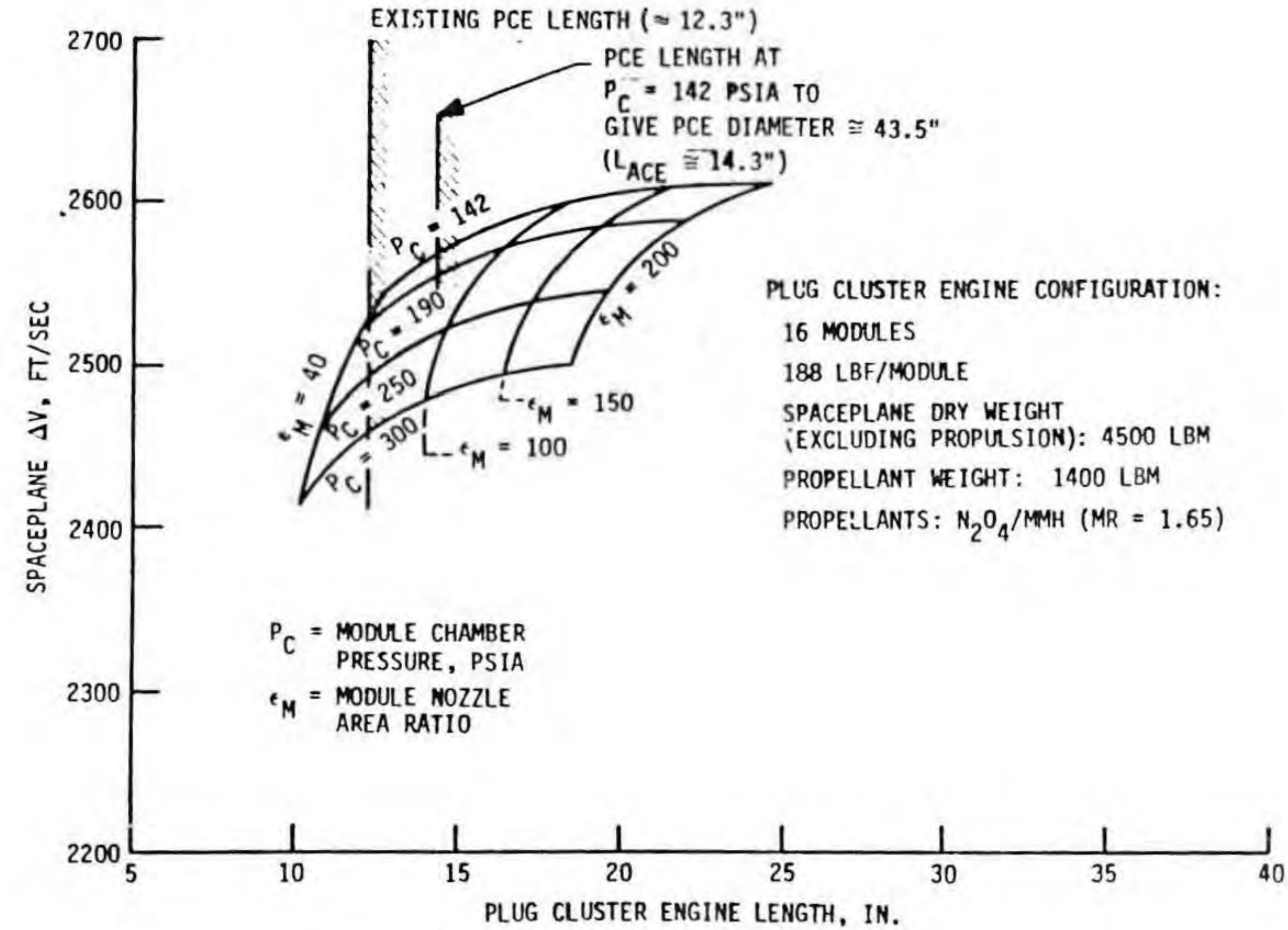


Figure 5-73 Spaceplane Δv vs Plug Cluster Engine Length (N = 16)

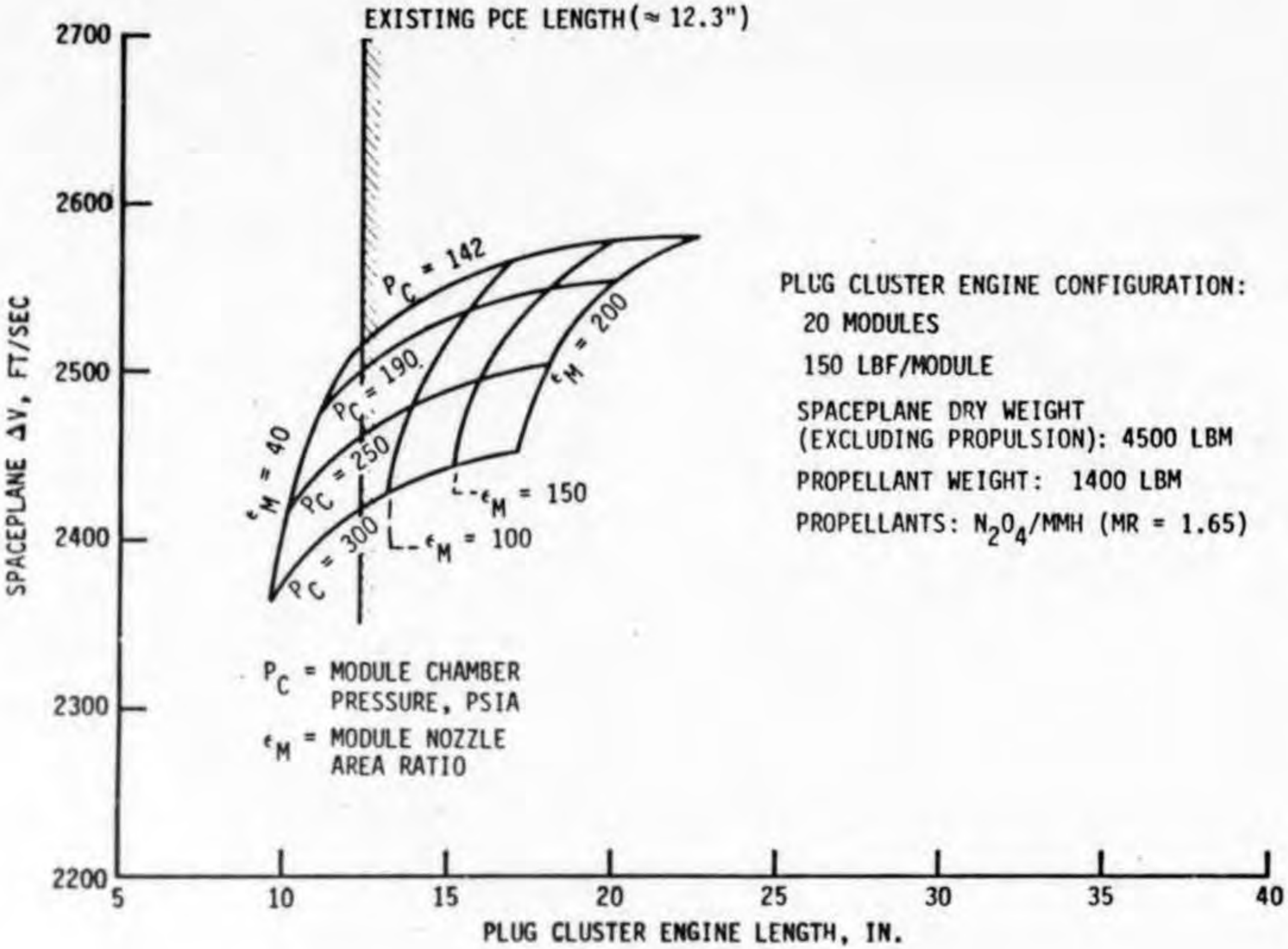


Figure 5-74 Spaceplane Δv vs Plug Cluster Engine Length (= 20)

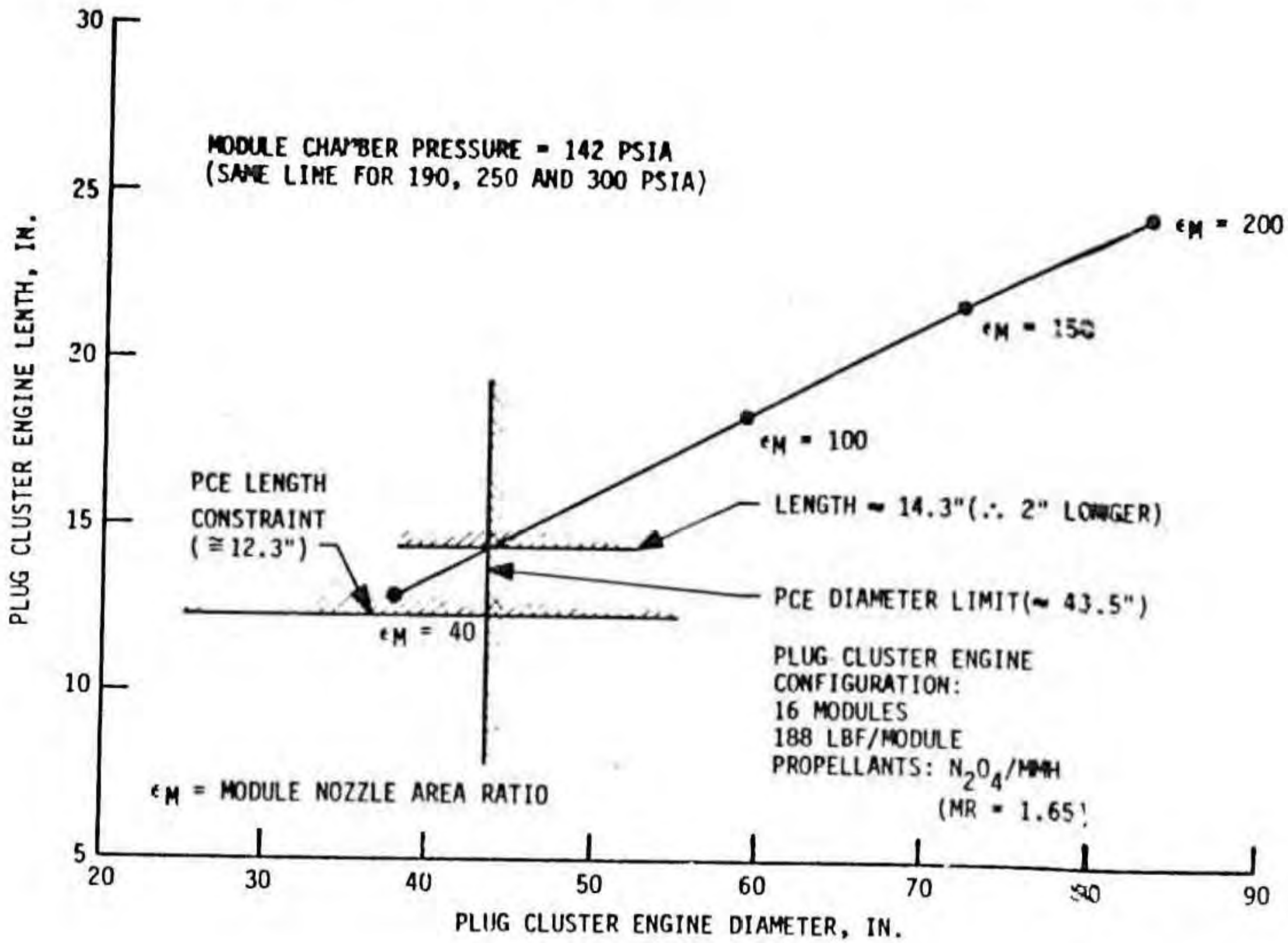


Figure 5-75 Plug Cluster Engine Length vs Plug Cluster Engine Diameter (N = 16)

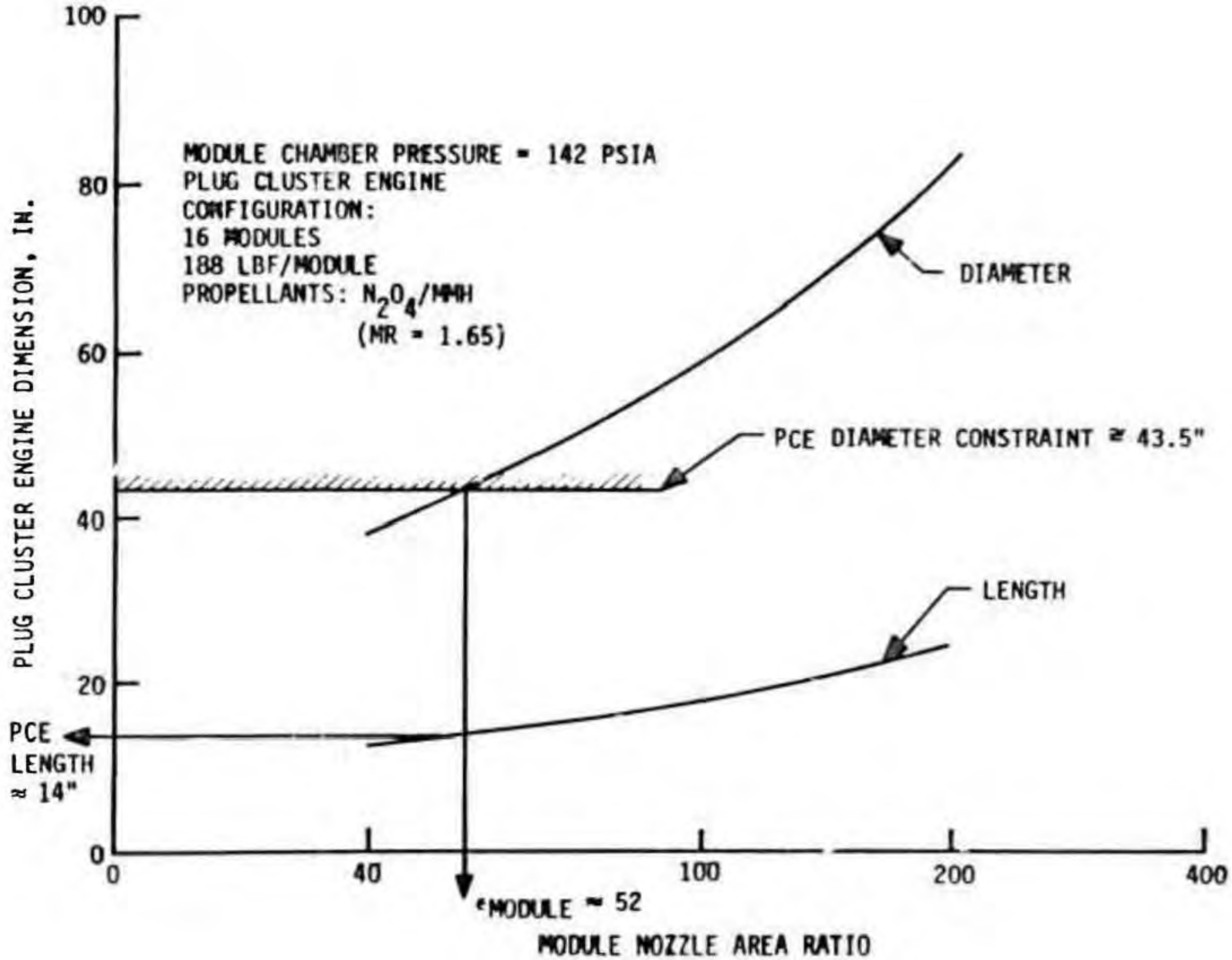


Figure 5-76 Plug-Cluster Engine Dimension vs Module Nozzle Area Ratio (N=16)

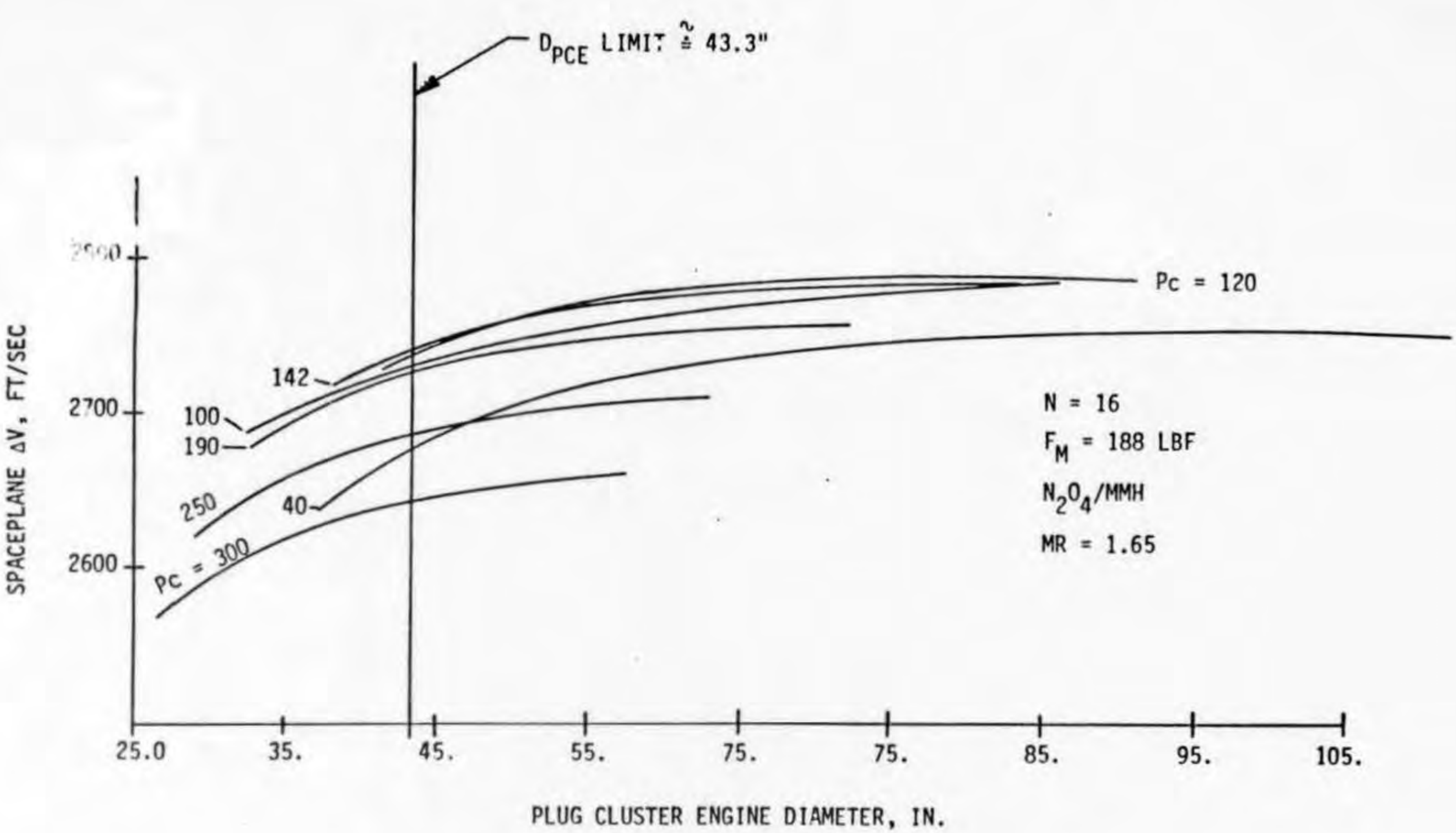


Figure 5-77 SP ΔV vs Plug Cluster Engine Diameter (N_2O_4/MMH)

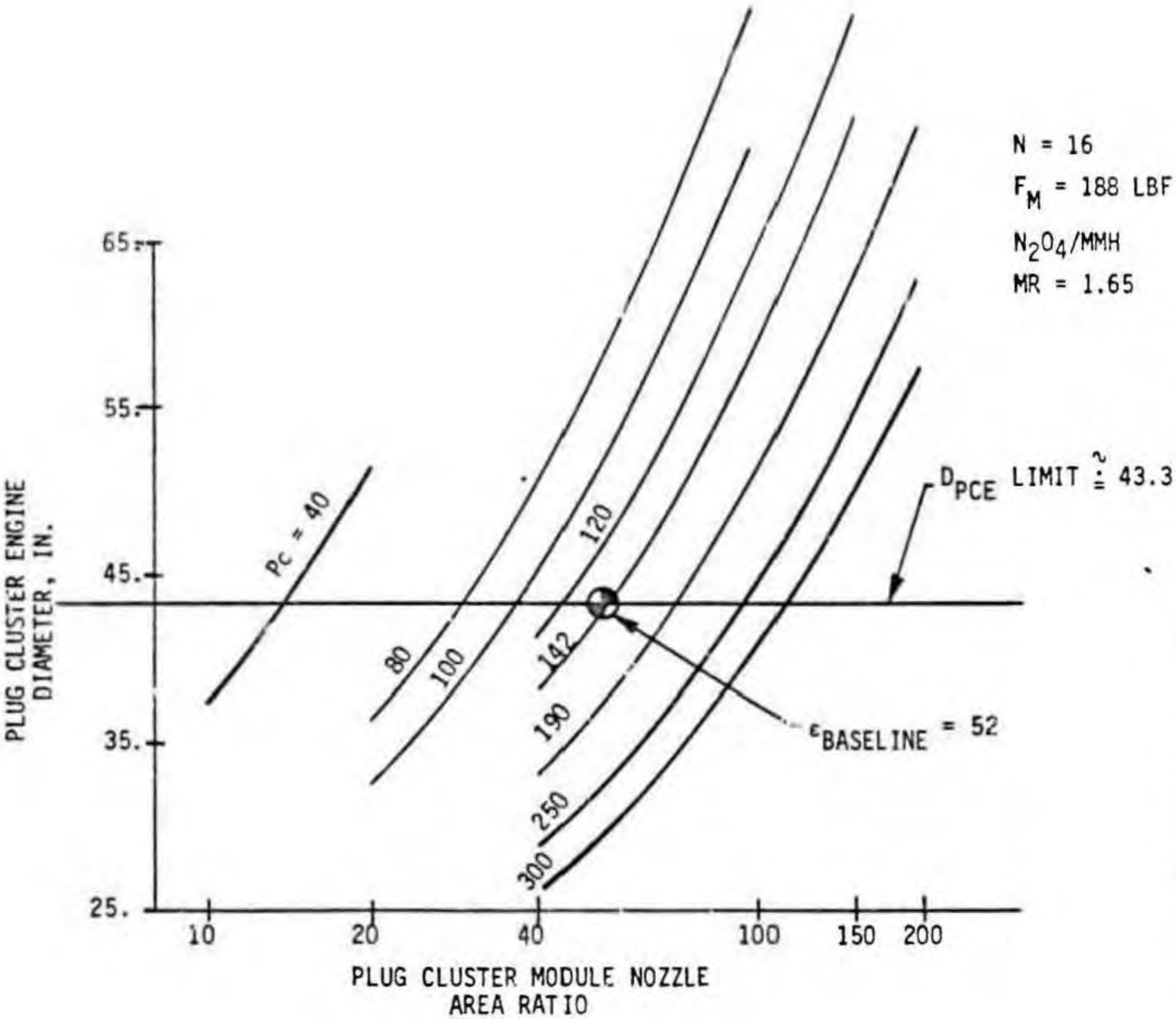


Figure 5-78 Plug Cluster Engine (PCE) Diameter vs PCE Module Nozzle Area Ratio (N_2O_4/MMH)

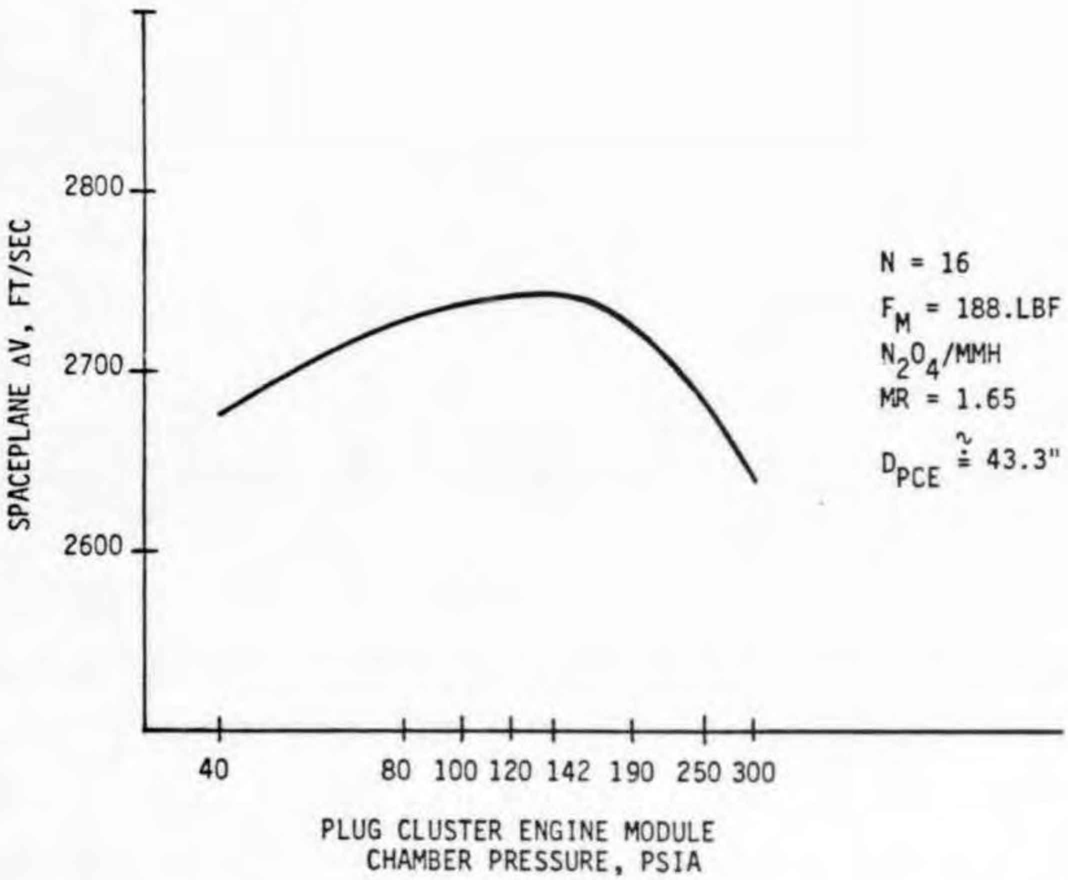


Figure 5-79 SPΔv vs Plug Cluster Engine Module Chamber Pressure (N_2O_4/MMH)

5.16.3 N_2O_4/MMH Propulsion System Selected Design

To generate the N_2O_4/MMH onboard propulsion system operating specification, SPV1 was executed at the design point identified in the optimization procedure (i.e., N = 16, Fm = 188 lbF, e_m = 52 and P_c = 142 psia). This SPV1 run also incorporated the H/S Spaceplane burnout weight (excluding the PCE, RCS, PS, and propellant tanks) of 4155 lbM.

A complete listing of both the inputs and outputs for this baseline system shown in Figures 80 and 81. The baseline design is also shown in Figures 82 through 85, which were taken directly from ALRC Drawing 1195445. All four of these figures, except Figure 58 for the propellant tank configuration, were shown previously in Figures 56, 57 and 59, which illustrated the current RCS. The baseline PCE modules are arranged such that the vehicle X and Y axes pass between pairs of PCE modules instead of through the center line of PCE modules. This was done so that the RCS valve assemblies could be located directly on the vehicle major axes, X and Y, between the PCE modules.

5.16.4 N_2O_4/PAAB-1 Optimization Propulsionn System

Some of the important assumptions made to perform this optimization are listed below.

- o The propellant tank pressure was defined as twice the PCE module chamber pressure. This assumption was based on comparison with existing pressure-fed, storable, man-rated rocket engines (e.g., OME).
- o As determined previously (Task 2.1 Performance), a mixture ration of 1.20 for the N_2O_4/PAAB-1 propellant combination was assumed.
- o The total Spaceplane burnout (or dry) weight, not including the weight of the PCE, RCS, PS, and propellant tanks, was 4155 lbM, as recommended by H/S.
- o Spherical propellant tanks were assumed.
- o 4000 psia HE storage pressure was assumed.
- o 1400 lbM total propellant load was assumed.
- o The HE was assumed to enter the propellant tank at approximately 70°F. This again, in turn, presupposes the use of a heat exchanger somewhere in the HE circuit.
- o The ranges of four major PCE design parameters were:
  - N (number of PCE modules): 16
  - Fm (PCE module thrust, lbF): 188
  - e_m (PCE module nozzle area rattio): 40, 100, 150, and 200

- P_c (PCE module chamber pressure, psia): 40, 100, 120, 142, 190, 250, and 300

16 PCE modules of 188 lbF each were selected as optimum on the basis of the N_2O_4/MMH propulsion system optimization results. The reasons for eliminating N = 8, 12 aand 20 (N = 8 and 12 represent new PCE module development; N = 20 results in lower SP delta-V compared to N = 16) are also valid for N_2O_4/PAAB-1.

- A total PCE outside diameter (D_{PCE}) constraint of 43.5 in. (corresponding to the preliminary baseline PCE) was again assumed.
- A total PCE length (L_{PCE}) constraint of 12 to 14 in. was also assumed.

The resulting propulsion system parametric weight, eenvelope and performance (SP delta-V) data is shown in Figures 86, 87, and 88. Figure 86 is a plot of SP delta-V vs. PCE diameter, with the PCE diameter constraint (43.3 inches) again superimposed. The plot of PCE diameter vs. PCE module nozzle area ratio with the PCE diameter constraint superimposed is shown in Figure 87. These two plots were then cross plotted to show SP delta-V as a function of PCE module chamber pressure for a fixed PCE diamter of 43.3 inches. This plot is shown in Figure 88.

The important trends and results of this optimization study are:

- SP delta-V improves with increasing e_m
- SP delta-V improves with increasing L_{PCE}
- SP delta-V improves with increasing D_{PCE}
- SP delta-V is a maximum at P_c = approx. 100 psia.

The PCE module nozzle area ratio at this point is 37.0. As a check, SPV1 was run at each P_c value again, with the PCE module nozzle area ratio required to yield a PCE diameter of 43.3 inches. The results of these runs confirmed the initial assumption that a PCE module P_C of 100 psia and PCE module nozzle area ratio of 37.0 are optimum.

The resulting optimum N_2O_4/PAAB-1 on-board propulsion system had the following basic characteristics:

- N = 16
- Fm = 188 lbF
- e_m = approx. 37
- P_c = 100 psia
- D_{PCE} = 43.3 inches
- L_{PCE} = 13.9 inches

```
 1.NUMBER OF PCE THRUSTERS=                              16.00
 2.PCE THRUSTER THRUST(LBF)=                            133.00
 3.PROPELLANTS (1 FOR N204/MMH, 2 FOR
   N204/PAAB-3(OR R-20))=                                 1.00
 4.PCE THRUSTER NOZZLE AREA RATIO=                       52.00
 5.PROPELLANT TANK PRESSURE(PSIA)=                      284.00
 6.PCE THRUSTER CHAMBER PRESSURE(PSIA)=                 142.00
 7.PCE THRUSTER MIXTURE RATIO=                           1.650
 8.SPACEPLANE VEHICLE WEIGHT(NOT
   INCLUDING PROPELLANTS, PROPELLANT OR
   PRESSURANT TANKS OR PCE(LBM)=                       4155.00
 9.OXIDIZER DENSITY(LBM/FT3)=                            90.90
10.FUEL DENSITY(LBM/FT3)=                                54.85
11.OXIDIZER TANK MATERIAL
   DESIGN SAFETY FACTOR=                                  4.00
12.FUEL TANK MATERIAL
   DESIGN SAFETY FACTOR=                                  4.00
13.OXIDIZER TANK MATERIAL DENSITY(LBM/IN3)=                .16
14.FUEL TANK MATERIAL DENSITY(LBM/IN3)=                    .16
15.OXIDIZER TANK MATERIAL
   ULTIMATE STRENGTH(PSIA)=                          130000.00
16.FUEL TANK MATERIAL
   ULTIMATE STRENGTH(PSIA)=                          130000.00
17.OXIDIZER TANK TYPE=                                    1.00
   (=1 FOR SPHERICAL)
   (=2 FOR CYLINDRICAL)
   (=3 FOR CONFORMAL)
   (=4 FOR TOROIDAL)
   (=5 FOR INTEGRAL FUEL/OX)
18.FUEL TANK TYPE=                                        1.00
   (SAME OPTIONS, EXCEPT 5, AS OX TANK TYPE)
19.X STATION LOCATION (IN.)
   OF OXIDIZER TANK C.G. (OR FOREWARD BULKHEAD IF
   CONFORMAL TANK)=                                     202.00
20.X STATION LOCATION (IN.) OF FUEL TANK C.G.
   (OR FOREWARD BULKHEAD IF CONFORMAL TANK)=            233.80
21.TOTAL PROPELLANT WEIGHT(LBM)=                       1400.00
22.HE STORAGE PRESSURE(PSIA)=                          4000.00
23.HE TEMP AT PROP DEPLETION(R)=                        530.00
24.TOROIDAL HE TANK SMALL RAD(IN.)=                       6.00
```

Figure 5-80 SPV1 Inputs for N_2O_4/MMH Spaceplane On Board Propulsion System

AEROJET LIQUID ROCKET COMPANY
SPACEPLANE INTERNAL PROPULSION DESIGN

PCE MODULE PERFORMANCE

1.ISP ODE(SEC)=	334.35
2.DIV EFF=	.993
3.BL LOSS(LBF)=	2.80
4.KIN EFF=	.975
5.ERE=	.990
6.COL EFF=	.976
7.ISP DEL(SEC)=	308.30
8.FUEL FLOW(LBM/SEC)=	.23
9.OX FLOW(LBM/SEC)=	.38
10.TOTAL FLOW(LBM/SEC)=	.61
11.CSTAR(FT/SEC)=	5509.13

PLUG CLUSTER ENGINE(PCE)
ENVELOPE PARAMETERS

1.MOD. THROAT AREA(IN2)	.753
2.THROAT DIAM(IN)	.979
3.THROAT RAD.(IN.)	.490
4.MOD. EXIT DIA(IN.)	7.062
5.PCE DIA(IN.)	43.26
6.BASE AREA(IN2)	728.69
7.PCE LENG(IN.)	14.15

PLUG CLUSTER ENGINE
PERFORMANCE PARAMETERS

1.	
2.PCE MACH NO.	5.88
3.PCE PRANDTL	119.48
4.MOD MACH NO.	4.99
5.MOD PRANDTL ANG	105.30
6.THETA(DEG)	14.189
7.PCE AREA RATIO	120.51
8.BASE PRESS(PSIA)	.187
9.BASE THRUST(LBF)	136.50
10.PCE THRUST(LBF)	3052.73
11.PCE ISP DEL.(SEC)	312.89

PLUG CLUSTER ENGINE(PCE)
WEIGHT BREAKDOWN(LBMS)

1.PLUG BASE	1.28
2.LINES	18.12
3.VALVES & ACT	27.16
4.INJECTORS	5.38
5.COMB. CHAMBERS	9.74
6.NOZZLES	20.95
7.TOTAL PCE	82.63
8.RCS WT(LBM)	45.00

SPACEPLANE VEH. PARAMETERS

1.BURN TIME(SEC)	143.49
2.FUEL WEIGHT(LBM)	528.30
3.OX WEIGHT(LBM)	871.70
4.FUEL VOL(FT3)	9.63
5.OX VOL(FT3)	9.59
6.PROP. C.G.(IN.)	214.00
7.PCE CG (IN)	224.67
8.GLOW(LBM)	5872.53
9.BURNOUT WT(LBM)	4472.53
10.DELTA V(FT/SEC)	2741.53

Figure 5-81 SPV1 Outputs for N_2O_4/MMH Spaceplane On Board Propulsion System (Sheet 1 of 2)

OXIDIZER TANK
(SPHERICAL)

1.RADIUS AVAILABLE(IN.)	15.48
2.RADIUS REQUIRED(IN.)	15.82
3.WALL THICKNESS(IN.)	.069
4.TANK WEIGHT(LBM)	34.75
5.X FOREWARD STATION(IN.)	186.12
6.X AFT STATION(IN.)	217.88
7.PROPELLANT C.G. STATION(IN.)	202.00

(TYPE RETURN KEY TO CONTINUE)

FUEL TANK
(SPHERICAL)

1.RADIUS AVAILABLE(IN.)	18.39
2.RADIUS REQUIRED(IN.)	15.84
3.WALL THICKNESS(IN.)	.069
4.TANK WEIGHT(LBM)	34.91
5.X FOREWARD STATION(IN.)	217.89
6.X AFT STATION(IN.)	249.71
7.PROPELLANT C.G. STATION(IN.)	233.80

PRESSURIZATION SYSTEM DATA

1.WT. OF HE FOR FUEL TANK(LBM)=	1.91
2.WT. OF HE FOR OX TANK(LBM)=	1.90
3.WT. OF RESIDUAL HE (LBM)=	1.18
4.TOTAL WT. OF HE REQD(LBM)=	4.98
5.TOTAL VOL(FT3) OF HE REQD=	2.05

SINGLE SPHERICAL HE TANK:

1.TANK RADIUS(IN.)=	9.46
2.TANK WALL THICKNESS(IN.)=	.641
3.TANK WT.(LBM)=	115.26

Figure 81. SPVI Outputs for N_2O_4/MMH Spaceplane On Board Propulsion System (Sheet 2 of 2)

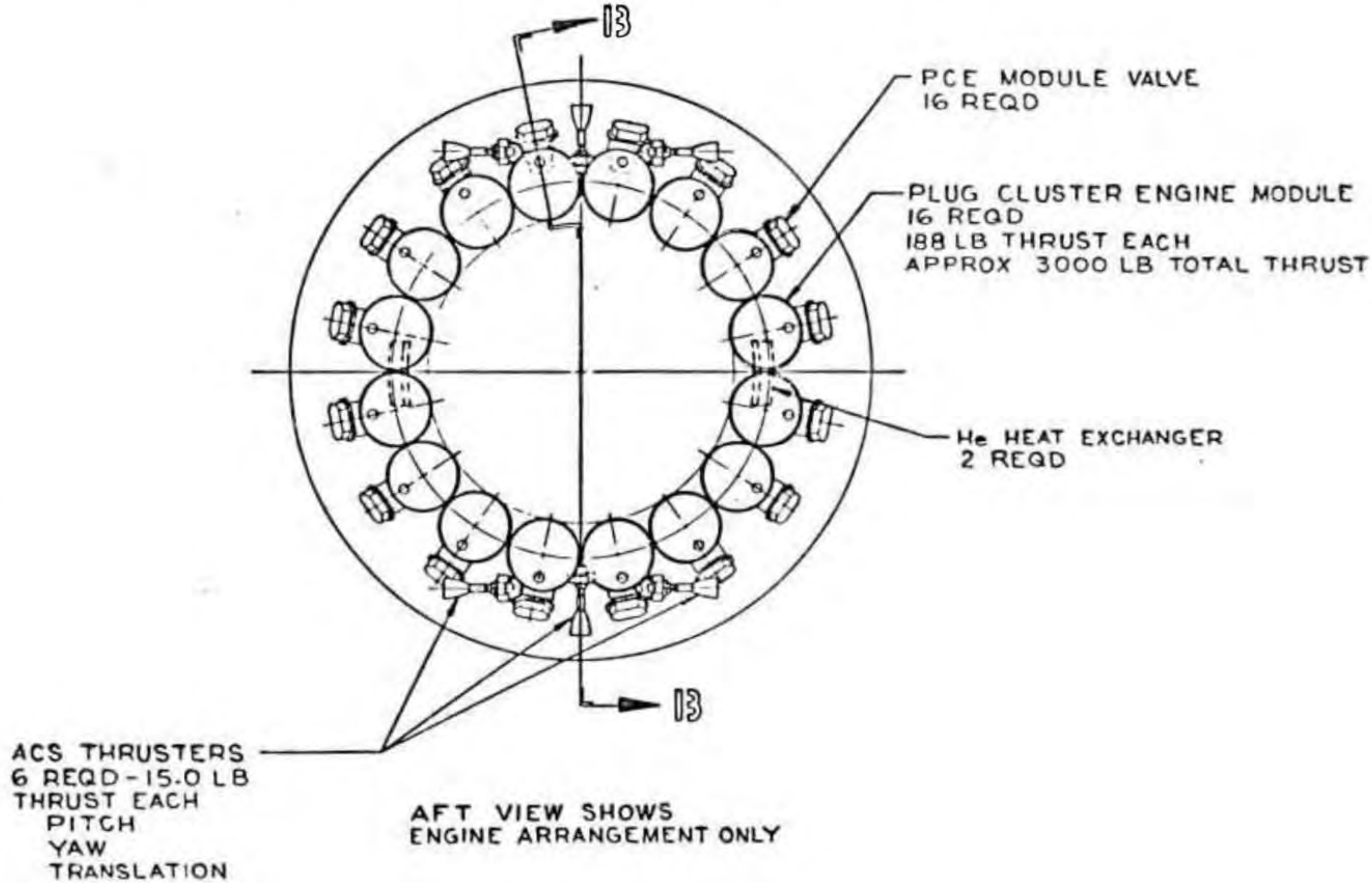


Figure 5-82 N_2O_4/MMH Plug Cluster Engine (Rear View)

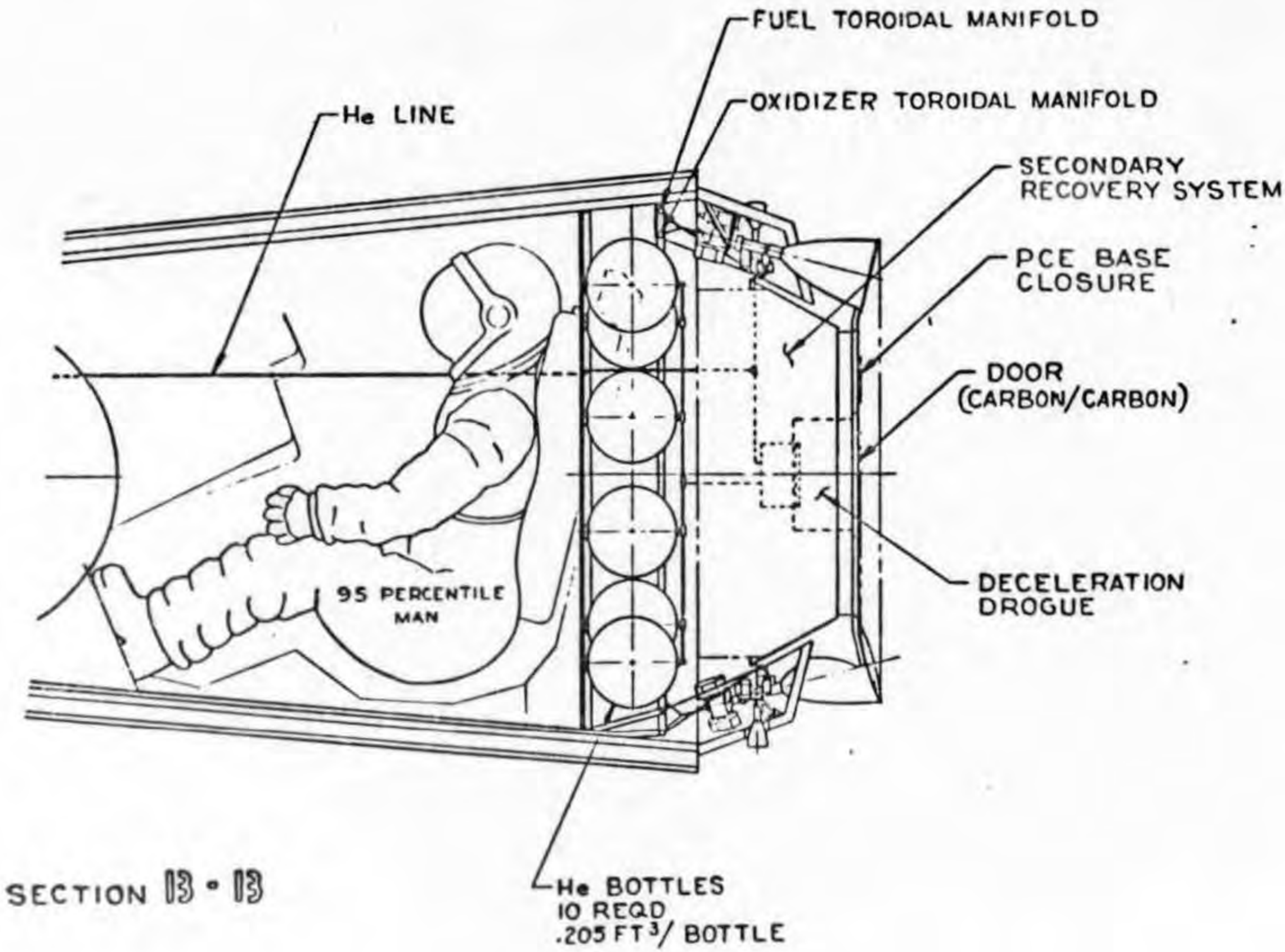


Figure 5-83 N_2O_4/MMH Plug Cluster Engine (Side View)

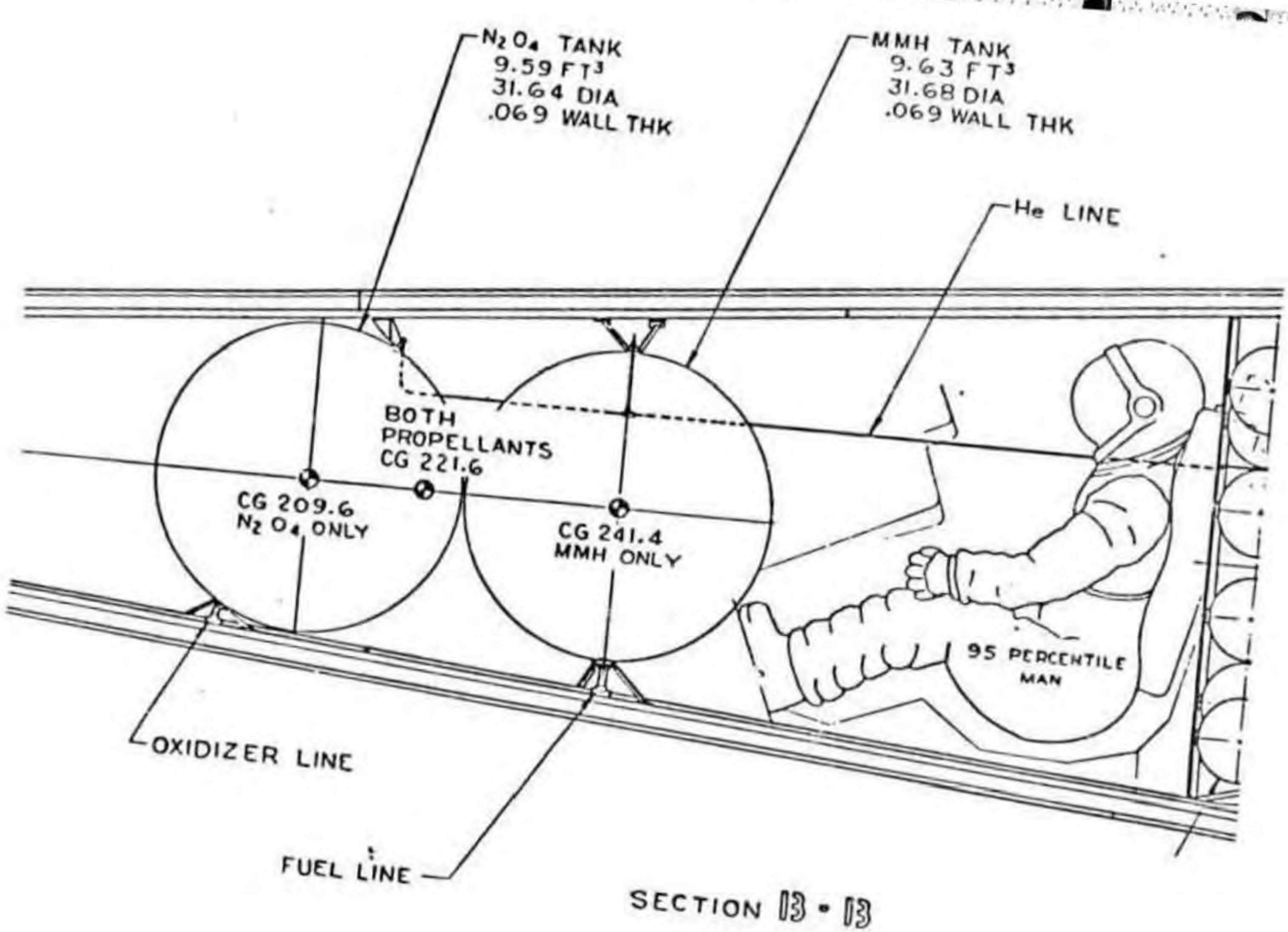


Figure 5-84 N_2O_4/MMH Propellant Tanks

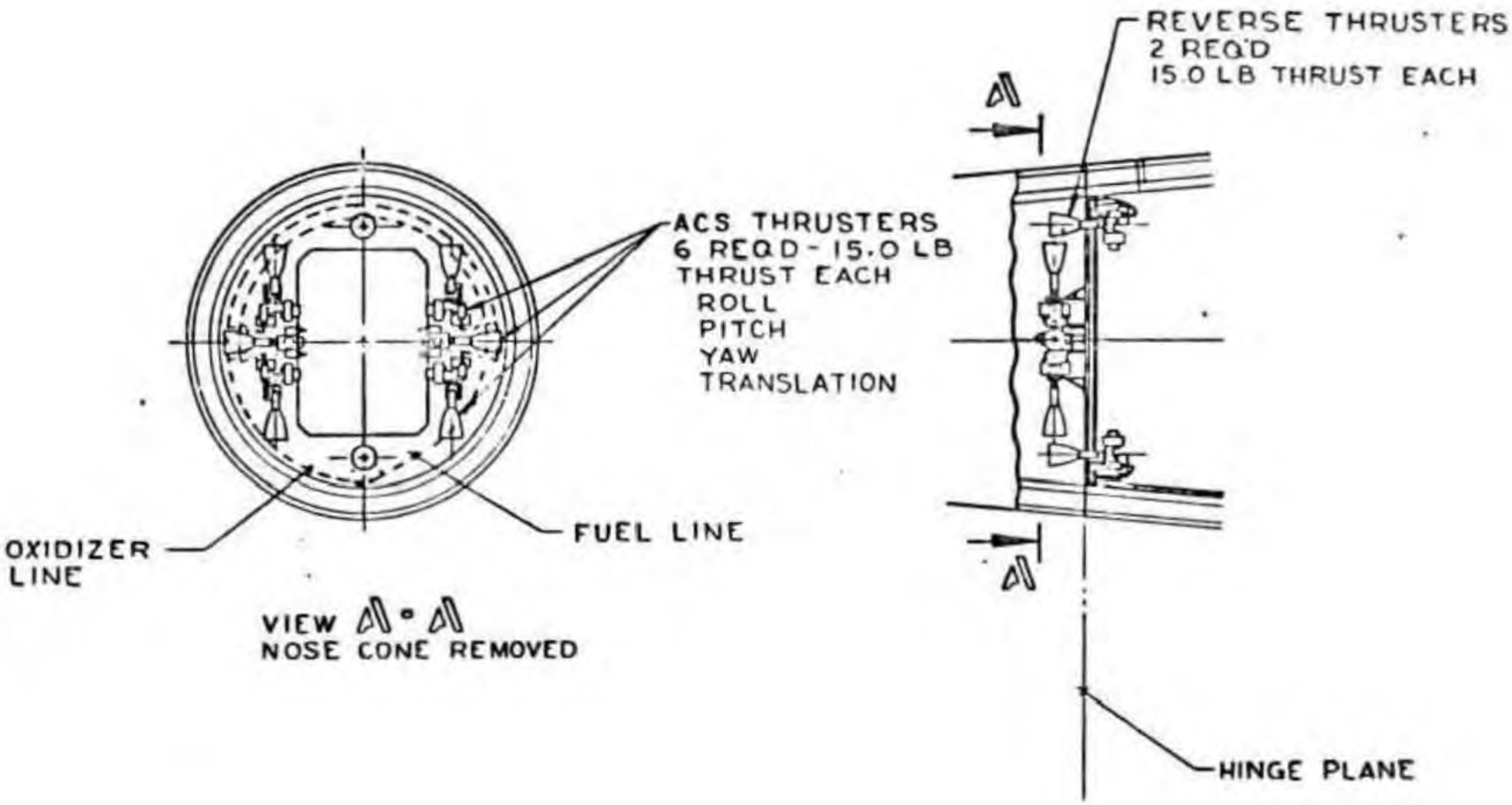


Figure 5-85 N_2O_4/MMH Foreward RCS Thrusters

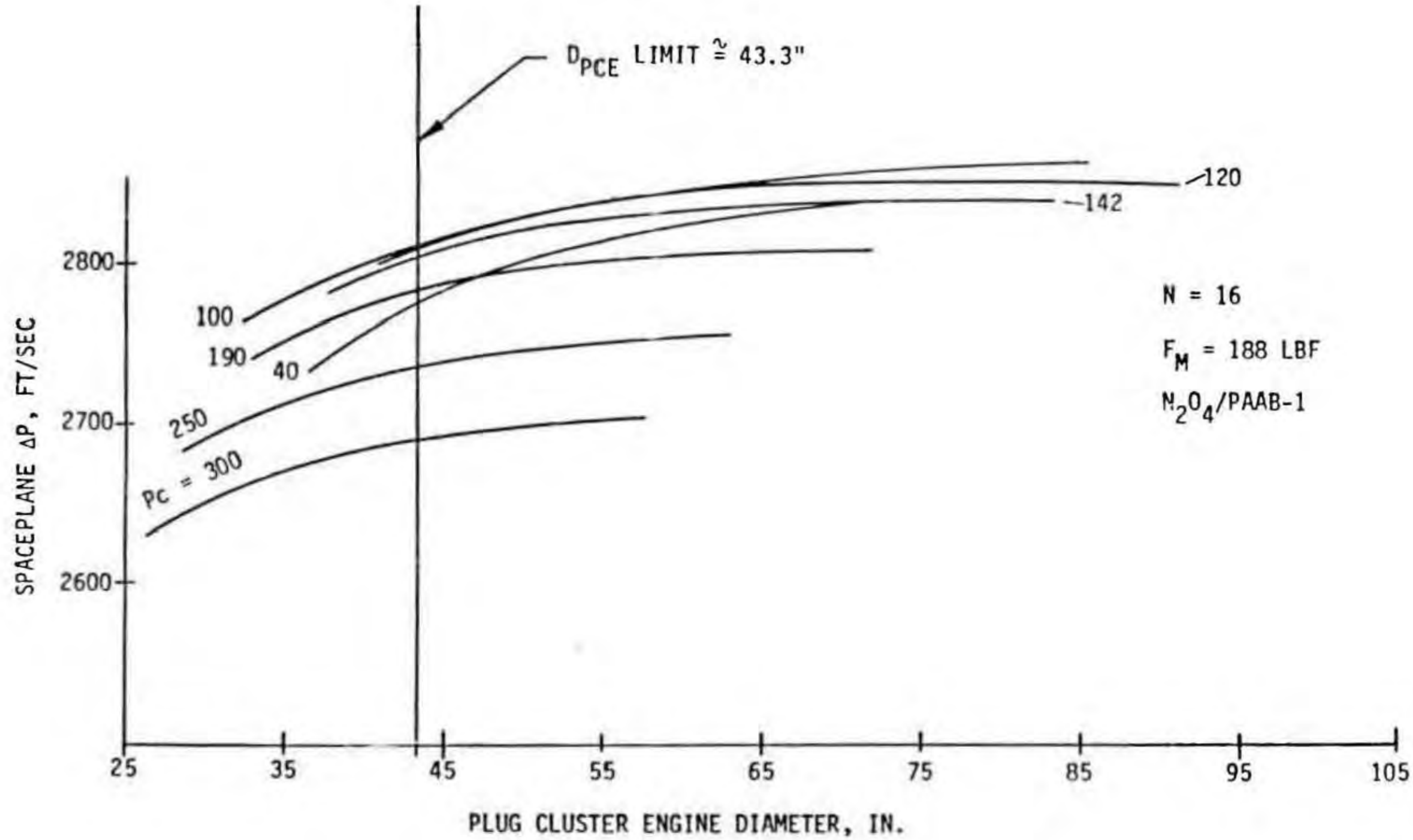


Figure 5-86 SPΔv vs Plug Cluster Engine Diameter (N_2O_4/PAAB-1)

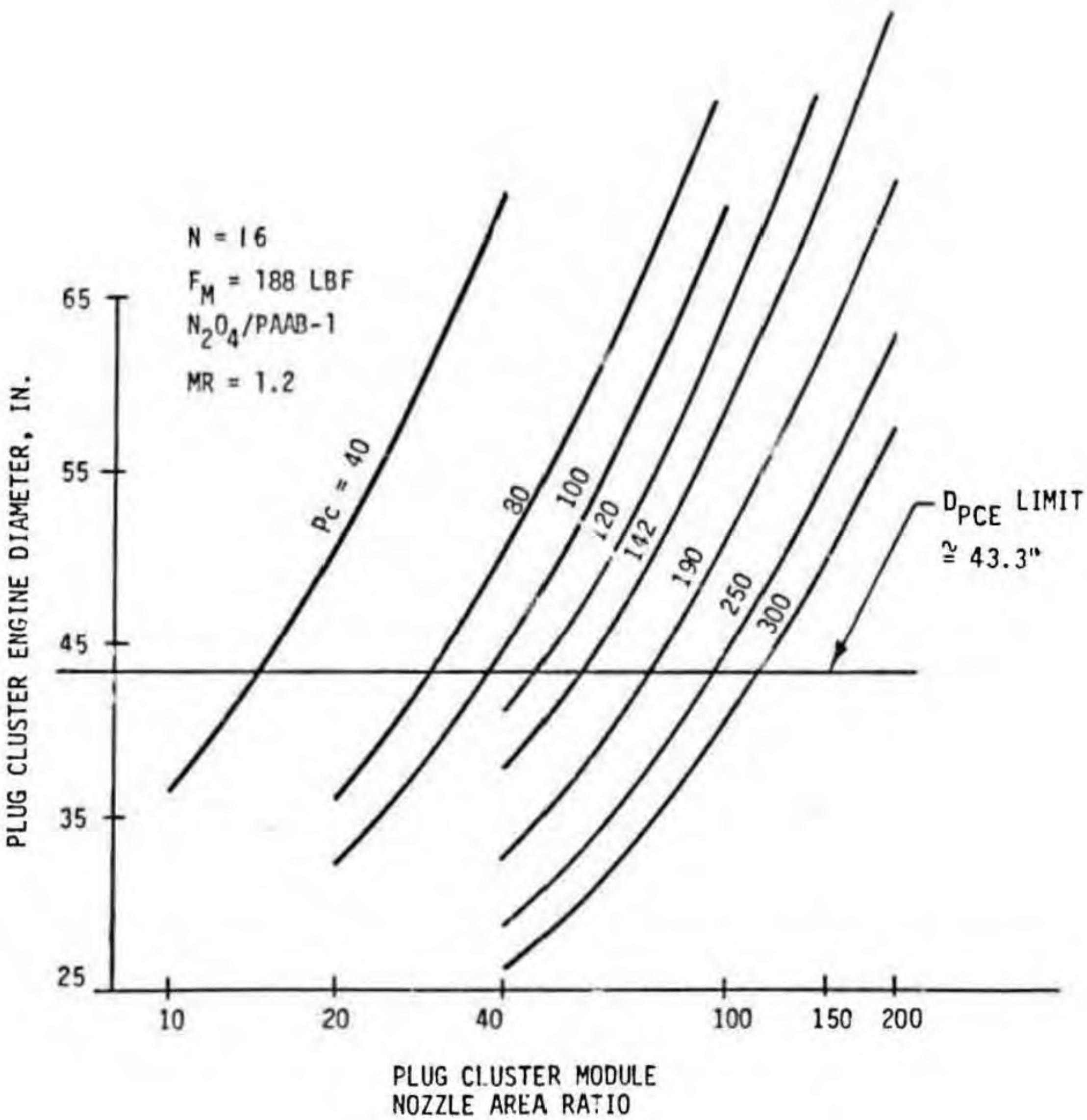


Figure 5-87 Plug Cluster Engine (PCE) Diameter vs PCE Module Nozzle Area Ratio (N_2O_4/PAAB-1)

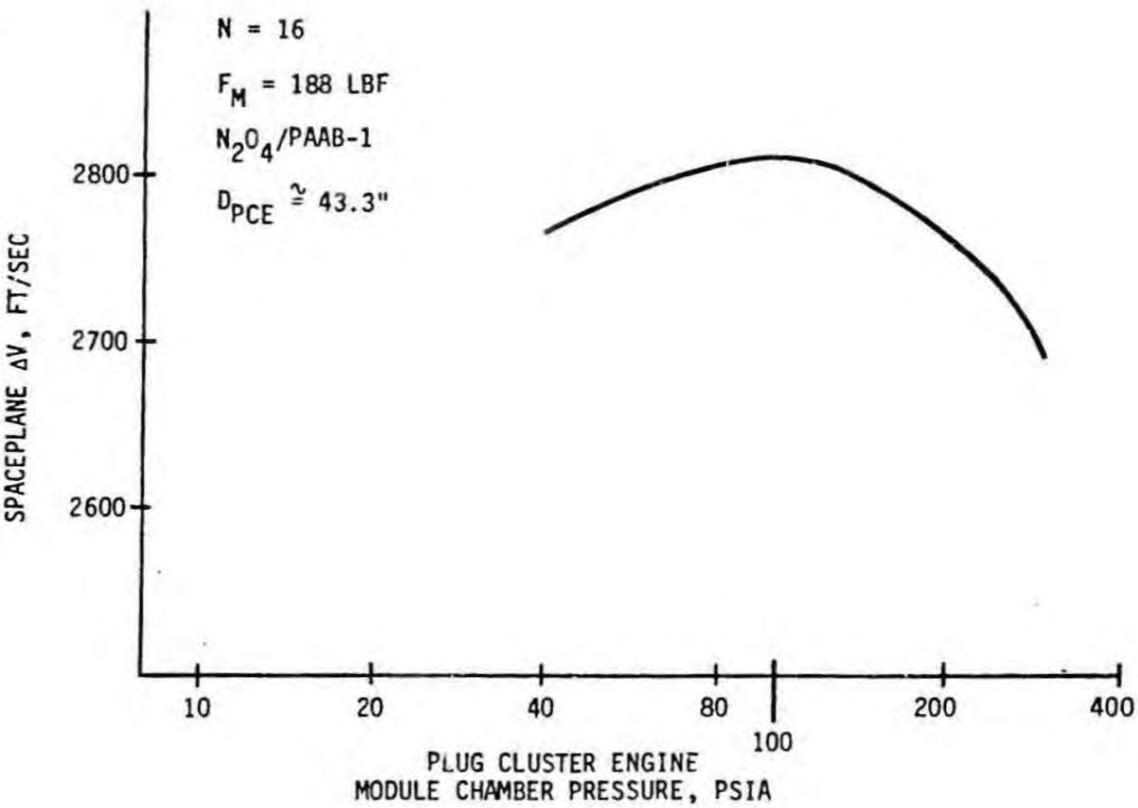


Figure 5-88 SP Δv vs Plug Cluster Engine Module Chamber Pressure (N_2O_4/PAAB-1)

It should be emphasized that the P_c = 100 psia case does not correspond to the maximum PCE vacuum delivered Isp case. For example, at a PCE module chamber pressure of 300 psia and PCE module nozzle ratio of 112, the total PCE vacuum delivered Isp is 322.2 seconds, as opposed to the optimum PCE value of 316.85 seconds. SP delta-V is actually higher (by about 110 ft/sec) at the lower PCE Isp value because the higher PCE Isp value (322.2 seconds) corresponds to the heaviest propellant pressurization subsystem and propellant tank weights. These heavier subsystem weights more than compensate for the increased SP delta-V one would expect with a high Isp. It was also assumed that the N_2O_4/PAAB-1 RCS would operate at a P_c of 100 psia.

5.16.5 N_2O_4/PAAB-1 Propulsion System Selected Design

To generate the N_2O_4/PAAB-1 onboard propulsion system operating specification, SPV1 was executed at the design point identified in the optimzation procedure (i.e., N = 16, fm = 188 lbF, e_m = 37 and P_c = 100 psia). The results of this execution, with both input and output, are shown respectively in Figures 89 and 90. SPV1 was also used to locate the propellant tank c.g.'s such that the resulting "propellants only" c.g. is at Station 221.6.

The baseline design is illustrated in Figure 91 through 95, all of which were taken directly from ALRC Drawing 1195445. The baseline PCE modules are arranged such that the vehicle X and Y axes pass between pairs of PCE modules instead of through the center line of PCE modules. This was done so that the RCS valve assemblies could be located directly on the vehicle major axes, X and Y, between the PCE modules.

Other aspects of the N_2O_4/PAAB-1 baseline onboard propulsion system are now discussed. These include:

- o Control panel requirements
- o TVC capability
- o Propellant tank evaluation

It should be noted that these observations are also valid for the N_2O_4/MMH baseline onboard propulsion system.

1. Control Panel Requirements

   The Spaceplane internal propulsion system parameters to be monitored from the cockpit control panel were identified in April. The selection of parameters was based on an evaluation of other manned aerospace vehicles including:

   - o Space Shuttle Orbiter
   - o Lunar Module (Descent Propulsion System)

   The Lunar Module is more similar to the Spaceplane than the Space Shuttle Orbiter because the descent propulsion system, like the Spaceplane internal propulsion, is a storable, pressure-fed system.

1.NUMBER OF PCE THRUSTERS=	16.00
2.PCE THRUSTER THRUST(LBF)=	188.00
3.PROPELLANTS (1 FOR N2O4/MMH, 2 FOR N2O4/PAAB-1(OR M-20))=	2.00
4.PCE THRUSTER NOZZLE AREA RATIO=	37.00
5.PROPELLANT TANK PRESSURE(PSIA)=	200.00
6.PCE THRUSTER CHAMBER PRESSURE(PSIA)=	100.00
7.PCE THRUSTER MIXTURE RATIO=	1.200
8.SPACEPLANE VEHICLE WEIGHT(NOT INCLUDING PROPELLANTS, PROPELLANT OR PRESSURANT TANKS OR PCE(LBM)=	4155.00
9.OXIDIZER DENSITY(LBM/FT3)=	90.90
10.FUEL DENSITY(LBM/FT3)=	61.39
11.OXIDIZER TANK MATERIAL DESIGN SAFETY FACTOR=	4.00
12.FUEL TANK MATERIAL DESIGN SAFETY FACTOR=	4.00
13.OXIDIZER TANK MATERIAL DENSITY(LBM/IN3)=	.16
14.FUEL TANK MATERIAL DENSITY(LBM/IN3)=	.16
15.OXIDIZER TANK MATERIAL ULTIMATE STRENGTH(PSIA)=	130000.00
16.FUEL TANK MATERIAL ULTIMATE STRENGTH(PSIA)=	130000.00
17.OXIDIZER TANK TYPE= (=1 FOR SPHERICAL) (=2 FOR CYLINDRICAL) (=3 FOR CONFORMAL) (=4 FOR TOROIDAL) (=5 FOR INTEGRAL FUEL/OX)	1.00
18.FUEL TANK TYPE= (SAME OPTIONS, EXCEPT 5, AS OX TANK TYPE)	1.00
19.X STATION LOCATION (IN.) OF OXIDIZER TANK C.G. (OR FOREWARD BULKHEAD IF CONFORMAL TANK)=	207.15
20.X STATION LOCATION (IN.) OF FUEL TANK C.G. (OR FOREWARD BULKHEAD IF CONFORMAL TANK)=	238.91
21.TOTAL PROPELLANT WEIGHT(LBM)=	1400.00
22.HE STORAGE PRESSURE(PSIA)=	4000.00
23.HE TEMP AT PROP DEPLETION(R)=	530.00
24.TOROIDAL HE TANK SMALL RAD(IN.)=	6.00

Figure 5-89 SPV1 Inputs for N_2O_4/PAAB-1 Spaceplane On Board Propulsion System

AEROJET LIQUID ROCKET COMPANY
SPACEPLANE INTERNAL PROPULSION DESIGN

PCE MODULE PERFORMANCE

1.ISP ODE(SEC)=	335.00
2.DIV EFF=	.992
3.BL LOSS(LBF)=	2.71
4.KIN EFF=	.979
5.ERE=	.990
6.COL EFF=	.981
7.ISP DEL(SEC)=	311.66
8.FUEL FLOW(LBM/SEC)=	.27
9.OX FLOW(LBM/SEC)=	.33
10.TOTAL FLOW(LBM/SEC)=	.60
11.CSTAR(FT/SEC)=	5564.38

PLUG CLUSTER ENGINE(PCE)
ENVELOPE PARAMETERS

1.MOD. THROAT AREA(IN2)	1.063
2.THROAT DIAM(IN)	1.163
3.THROAT RAD.(IN.)	.582
4.MOD. EXIT DIA(IN.)	7.077
5.PCE DIA(IN.)	43.35
6.BASE AREA(IN2)	731.75
7.PCE LENG(IN.)	13.88

PLUG CLUSTER ENGINE
PERFORMANCE PARAMETERS

1.	
2.PCE MACH NO.	5.46
3.PCE PRANDTL	113.89
4.MOD MACH NO.	4.55
5.MOD PRANDTL ANG	99.26
6.THETA(DEG)	14.630
7.PCE AREA RATIO	85.69
8.BASE PRESS(PSIA)	.202
9.BASE THRUST(LBF)	147.64
10.PCE THRUST(LBF)	3058.12
11.PCE ISP DEL.(SEC)	316.85

PLUG CLUSTER ENGINE(PCE)
WEIGHT BREAKDOWN(LBMS)

1.PLUG BASE	1.34
2.LINES	19.01
3.VALVES & ACT	26.26
4.INJECTORS	6.37
5.COMB. CHAMBERS	11.57
6.NOZZLES	20.58
7.TOTAL PCE	85.13
8.RCS WT(LBM)	45.00

SPACEPLANE VEH. PARAMETERS

1.BURN TIME(SEC)	145.05
2.FUEL WEIGHT(LBM)	636.36
3.OX WEIGHT(LBM)	763.64
4.FUEL VOL(FT3)	10.37
5.OX VOL(FT3)	8.40
6.PROP. C.G.(IN.)	221.59
7.PCE CG (IN)	225.31
8.GLOW(LBM)	5811.04
9.BURNOUT WT(LBM)	4411.04
10.DELTA V(FT/SEC)	2810.03

Figure 5-90 SPV1 Outputs for N_2O_4/PAAB-1 Spaceplane On Board Propulsion System (Sheet 1 of 2)

OXIDIZER TANK
(SPHERICAL)

1.RADIUS AVAILABLE(IN.)	15.96
2.RADIUS REQUIRED(IN.)	15.13
3.WALL THICKNESS(IN.)	.047
4.TANK WEIGHT(LBM)	21.44
5.X FOREWARD STATION(IN.)	191.97
6.X AFT STATION(IN.)	222.33
7.PROPELLANT C.G. STATION(IN.)	207.15

FUEL TANK
(SPHERICAL)

1.RADIUS AVAILABLE(IN.)	18.86
2.RADIUS REQUIRED(IN.)	16.23
3.WALL THICKNESS(IN.)	.050
4.TANK WEIGHT(LBM)	26.46
5.X FOREWARD STATION(IN.)	222.63
6.X AFT STATION(IN.)	255.19
7.PROPELLANT C.G. STATION(IN.)	238.91

PRESSURIZATION SYSTEM DATA

1.WT. OF HE FOR FUEL TANK(LBM)=	1.45
2.WT. OF HE FOR OX TANK(LBM)=	1.17
3.WT. OF RESIDUAL HE (LBM)=	.61
4.TOTAL WT. OF HE REQD(LBM)=	3.23
5.TOTAL VOL(FT3) OF HE REQD=	1.33

SINGLE SPHERICAL HE TANK:

1.TANK RADIUS(IN.)=	8.19
2.TANK WALL THICKNESS(IN.)=	.555
3.TANK WT.(LBM)=	74.78

Figure 90. SPV1 Outputs for N_2O_4/PAAB-1 Spaceplane on Board Propulsion System (Sheet 2 of 2)

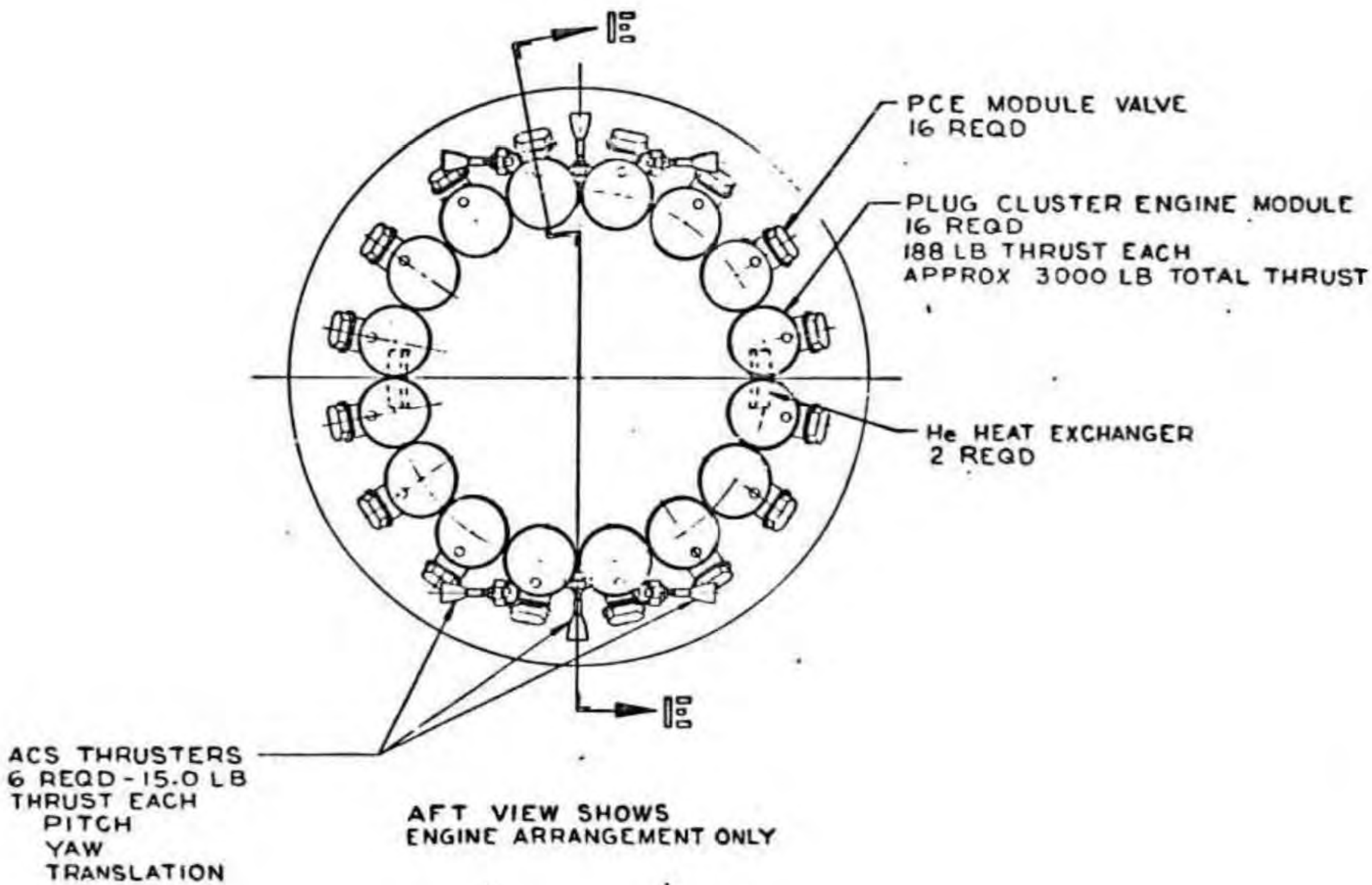


Figure 5-91 N_2O_4/PAAB-1 Plug Cluster Engine (Rear View)

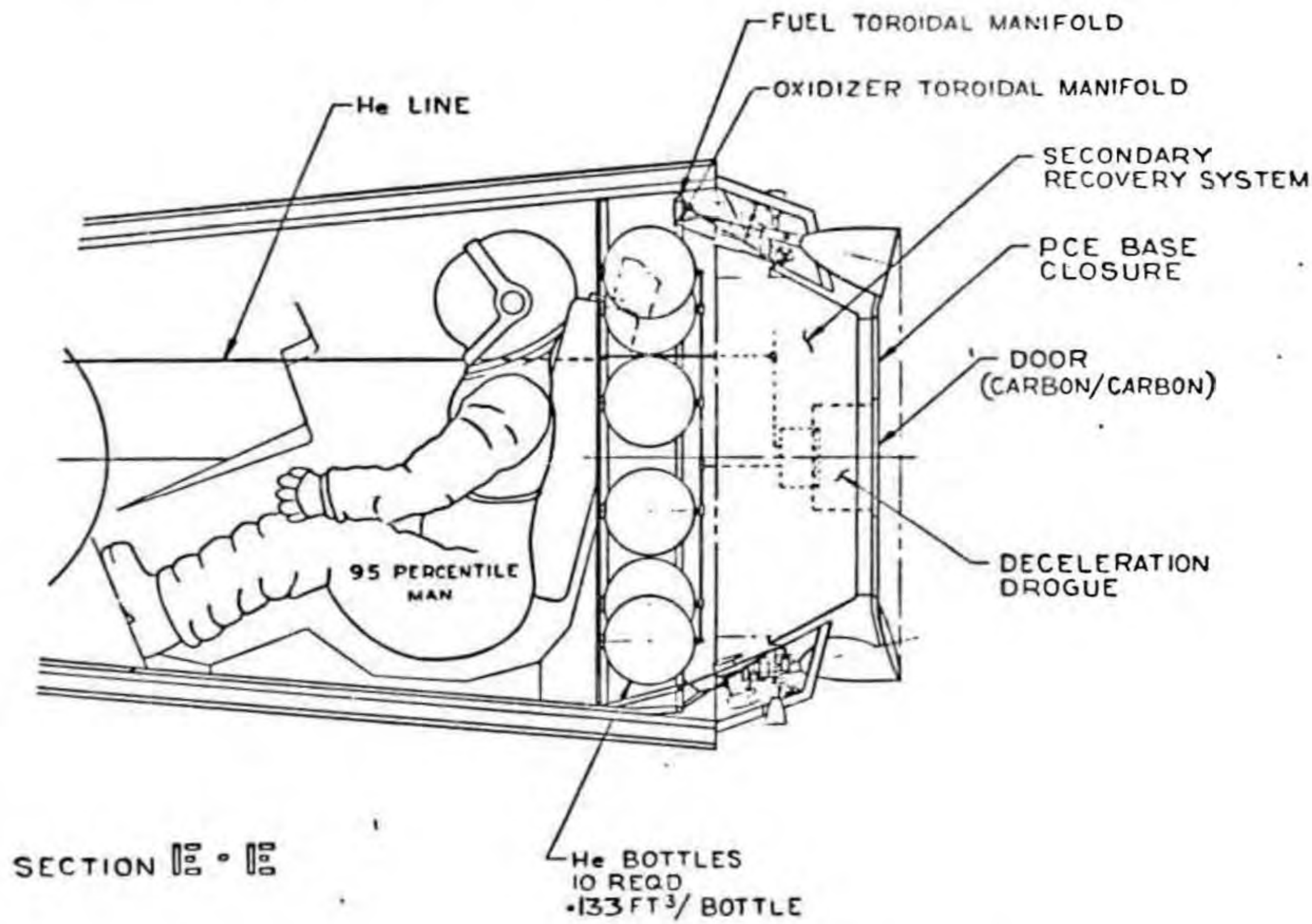


Figure 5 92 N_2O_4/PAAB-1 Plug Cluster Engine (Side View)

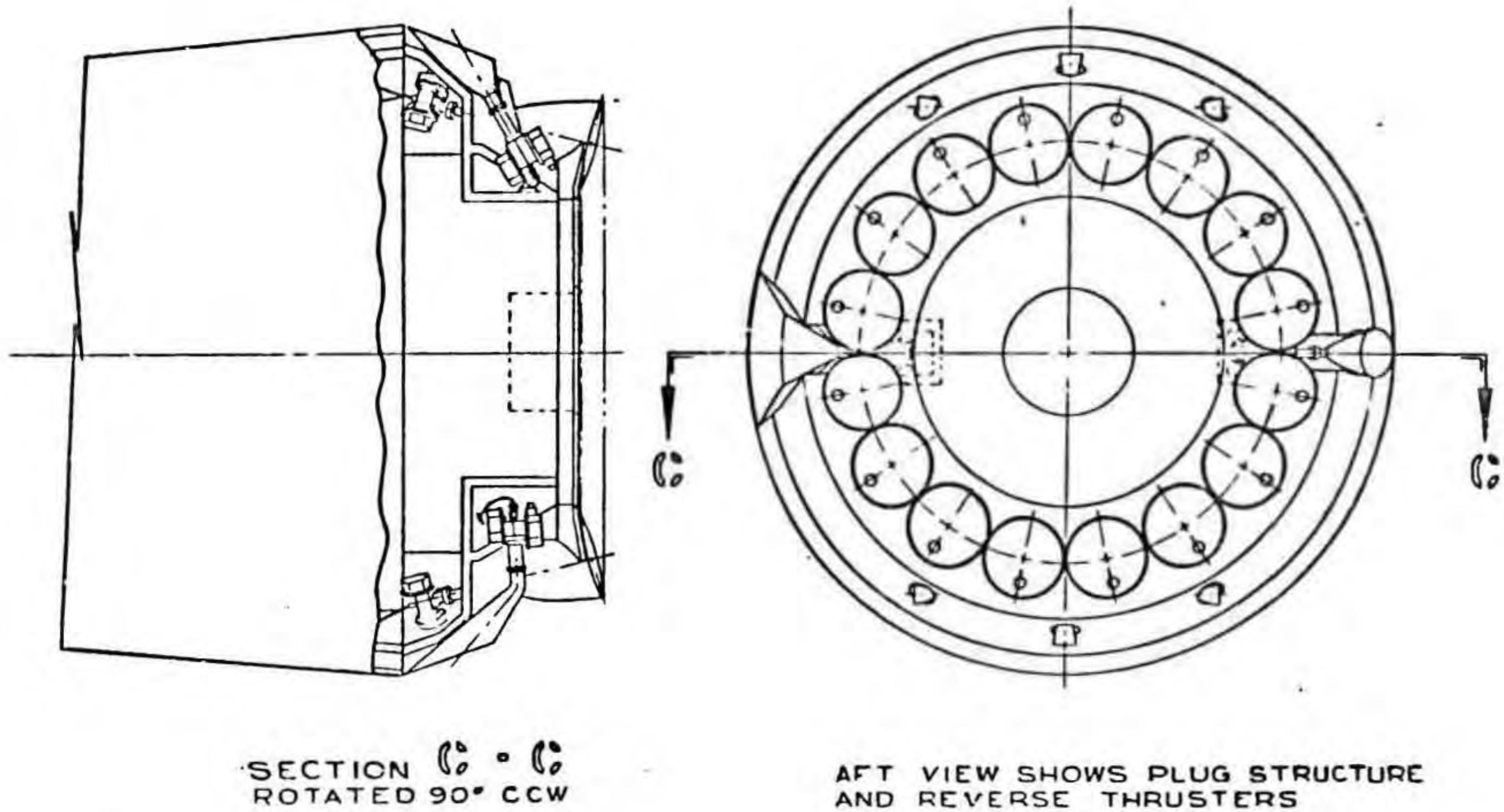


Figure 5-93 N_2O_4/PAAB-1 Retro Thrusters

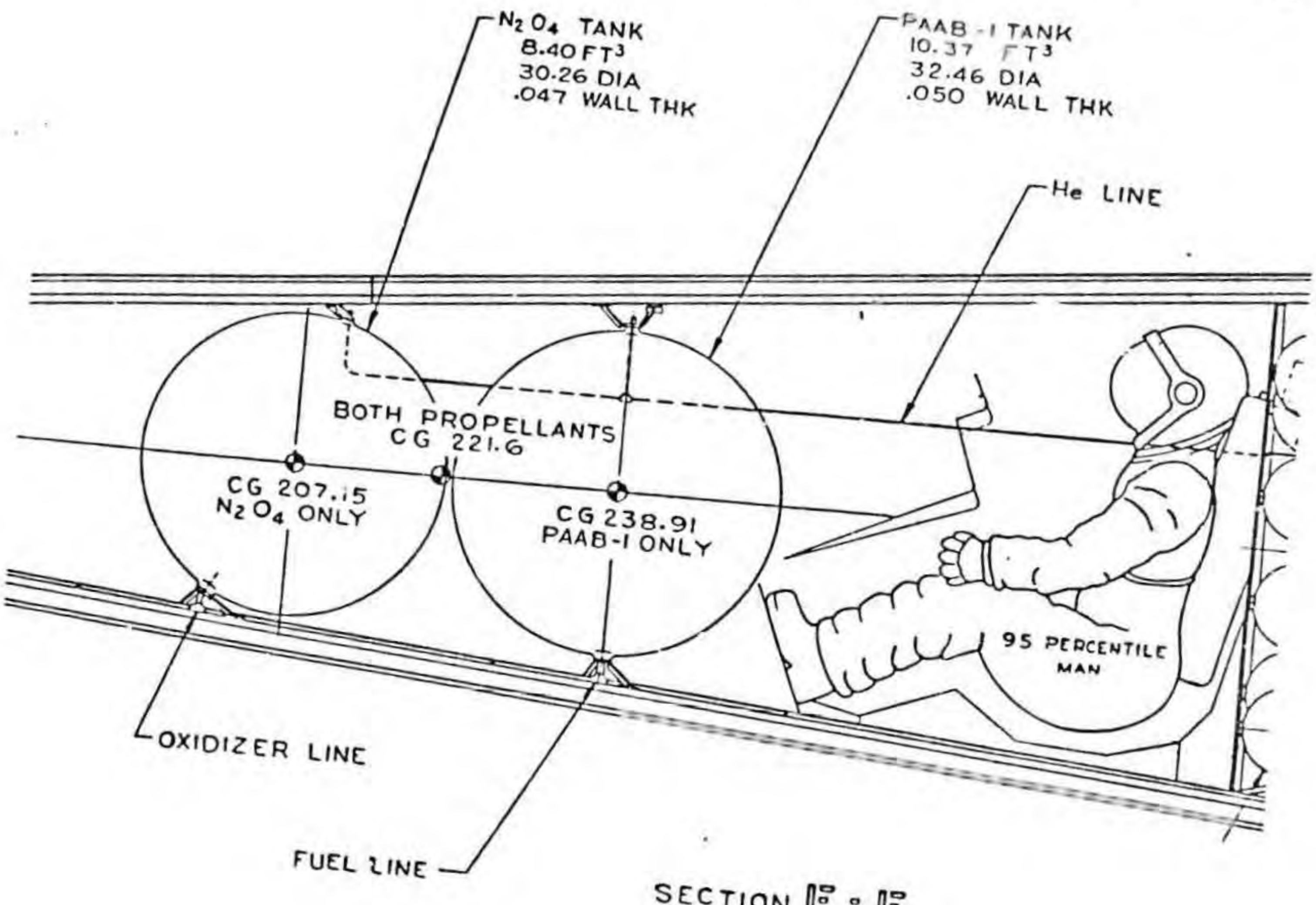


Figure 5-94 N_2O_4/PAAB-1 Propellant Tanks

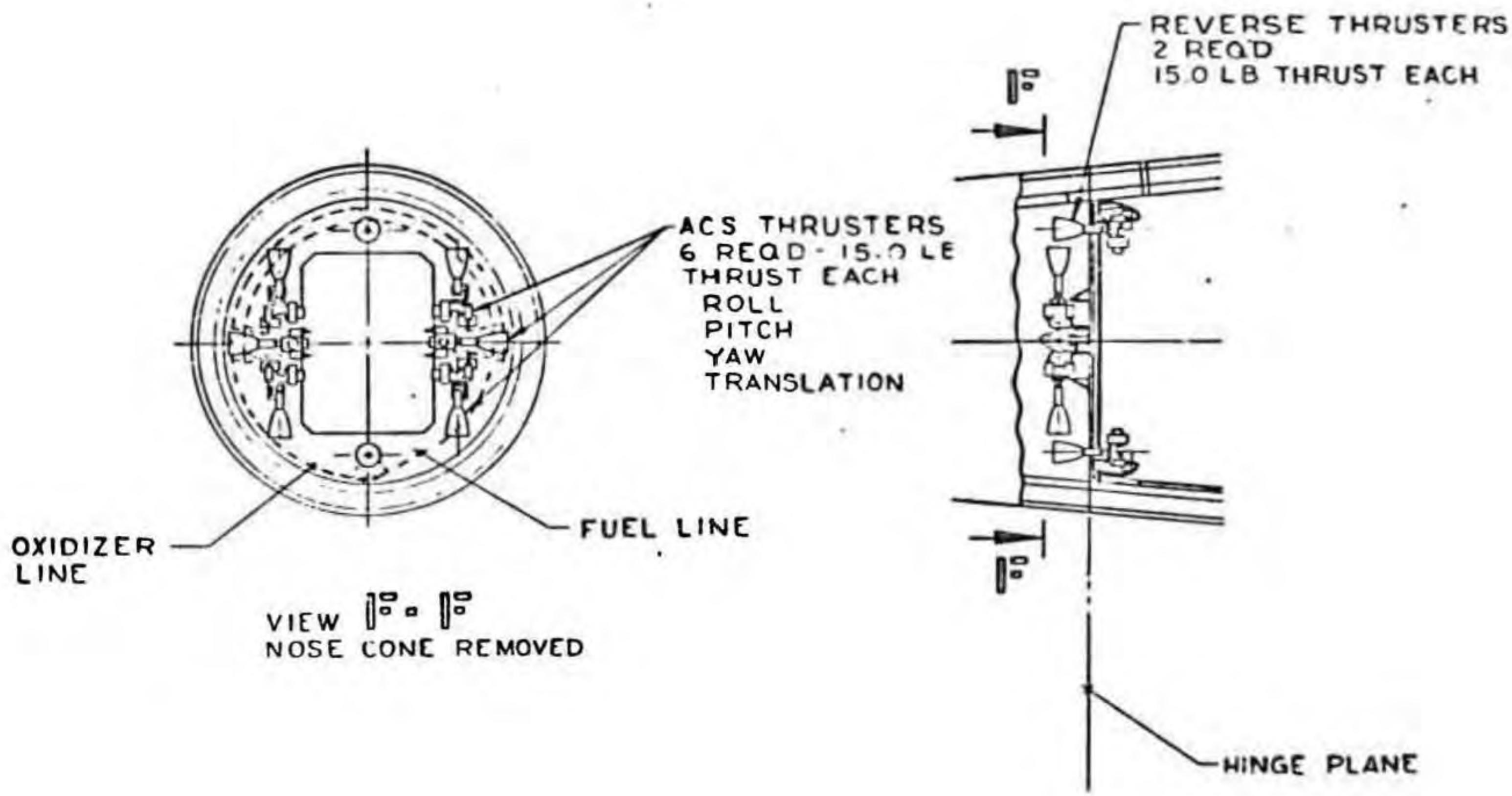


Figure 5-95 N_2O_4//PAAB-1 Foreward RCS Thrusters

Several modes of propulsion system parameter display were also identified. These included:

- o Always visible
- o Available upon pilot action
- o Caution and/or warning (C/W) signal
- o Combinations of the above

It should be pointed out that in most cases the C/W signal can and should be integral with both the "always visible" display and "pilot action" display. The resulting two display classifications are used in Table XX to list the Spaceplane internal propulsion system parameters to be displayed.

It should also be noted that the display requirements, as described in Table XX, are maximum requirements. For the Spaceplane application, several, or possibly many, of these displays may be unneccessary.

TABLE XX

SPACEPLANE INTERNAL PROPULSION PARAMETERS FOR COCKPIT DISPLAY

Always Visible with Internal Caution/Warning (dial, meter, gauge, digital, etc.)	Available (Pilot Action Required) with Integral Caution/Warning)
1. Fuel Quantity (% or lbM)	F or P_c of individual PCE modules (16)
2. Ox Quantity (% or lbM)	Temp of individual PCE modules (16)
3. Fuel Temp	Indication of firing PCE modules (16)
4. Ox Temp	Fuel flowrate (PCE)
5. Fuel Pressure	Ox flowrate (PCE)
6. Ox Pressure	Indication of firing RCS thrusters
7. He Temp in Fuel Tank	Temp of individual RCS thrusters
8. He Temp in Ox Tank	F or P_c of individual RCS thrusters
9. He Pressure in Fuel Tank	
10. He Pressure in Ox Tank	
11. He Bottle Pressure	
12. He Bottle Temp	
13. PCE Thrust Level	

2. TVC

A thrust vector control (TVC) analysis of the baseline PCE was conducted. The major results and trends of this analysis are described here. For the purposes of this analysis, the effective gimbal angle is defined as the angle between the SPaceplane vehicle longitudinal axis (+X axis) through the c.g. and the direction of PCE thrust acting through the vehicle c.g.

- o The maximum effective gimbal angle (14.2^{o}) decreases with the number of PCE modules firing at one time. The maximum value occurs with only 1 PCE module firing. The minimum gimbal angle is 0.966^{o} which occurs when all but one module is firing. (There is no effective gimbaling when all of the PCE modules are firing.)

- o The greatest vehicle moment results when the given number of firing PCE modules are sequentially located adjacent to each other.

- o For an even number of firing PCE modules, the resulting total minimum moment possible (in addition to the translational acceleration) is always zero. For an uneven number of firing PCE modules, the resulting total minimum moment possible is always some fixed value (beginning at 9357.7 lbF-in. with one PCE module firing, down to a value less than 2000 lbF-in. with more than one PCE module firing).

- o An even number of firing PCE modules can provide pure pitch or yaw moments in addition to vehicle translational acceleration. An uneven number of firing PCE modules cannot provide pure pitch or yaw in addition to vehicle translational acceleration.

- o The total vehicle translational acceleration is a strong function of the numberof firing PCE modules but not of the position if those modules.

All of the above trends are documented in Table XXI, which consists of a brief summary of the total capability of the PCE as a function of number of firing PCE modules.

TABLE XXI

N_2O_4/PAAB-1 PLUG CLUSTER ENGINE TVC CAPABILITY

N (Number of PCE Modules Firing)	$A_{CG\ Max}$ (Max Linear Acceleration of Spaceplane CG, ft/sec^2)	M_{Max} (Max Moment, lbF-in.)	α_{Max} (Max Angular Acceleration About Spaceplane CG, °/sec^2)	Θ_{Max} (Direction of Linear CG Acceleration with Respect to Vehicle Longitudinal [X] Axis)
0	0	0	0	0
1	1.008	9,357.7	9.215	14.2
2	2.014	18,355.8	18.08	13.93
3	3.016	26,649.0	26.24	13.50
4	4.01	33,917.3	33.4	12.91
5	5.00	39,884.0	39.28	12.16
6	5.982	44,315.2	43.64	11.29
7	6.955	47,046.1	46.33	10.29
8	7.922	47,966.7	47.23	9.203
9	8.885	47,046.1	46.33	8.04
10	9.845	44,315.2	43.64	6.829
11	10.805	39,884.0	39.28	5.596
12	11.76	33,917.3	33.4	4.368
13	12.73	26,649.0	26.24	3.171
14	13.69	18,355.8	18.08	2.029
15	14.67	9,357.7	9.215	0.966
16	15.641	0	0	0

3. Propellant Tank Evaluation

A review of existing aerospace vehicle tankage was made to determine if the current Spaceplane propellant tanks and HE bottles have acceptable D/t ratios (D = tank diameter, t = tank wall thickness). Some of the systems evaluated were:

- Mariner '71 N_2O_4 Tank
- Agena Propulsion He Bottle
- Atlas Pneumatic System He Bottle
- Gemini RCS N_2O_4 Tank
- Lunar Module Ascent Engine Tanks (A-50 or N_2O_4)
- Saturn II Pneumatic System He Bottle
- X-15 H_2O_2 Tank
- Agena GN_2 Bottles

The three important results of this evaluation were:

- The Spaceplane tank (propellant tanks and He bottles) D/t values are somewhat conservative (i.e., low).
- The Spaceplane tank weights are consistent with existing systems of the same size, volume and storage pressure.
- From these two observations it can be implied that the actual Spaceplane tank wall thickness will be smaller (makes D/t larger) than predicted by SPV1, whereas the total predicted tank weights are accurate.

5.17 REPORTING AND PRESENTATIONS

The total program required a monthly progress report at the end of each month, in addition to a final report. There were, as a result, eight monthly reports written, as well as the final report.

In addition to these reports, two technical presentations were made at TI/TD (Technical Information/Technical Direction) meetings held at Aerospace Corporation in January and May of 1982. Other contractors and government agencies in attendance at these meetings included:

- o SRI (Stanford Research Institute)
- o SNLA Sandia National Laboratory at Alber erque
- o H/S (Hamilton Standard)
- o Lincom
- o Honeywell
- o Pratt & Whitney
- o USAF Space Division personnel
- o Aerospace Corporation
- o Lockheed

A third technical presentation/final briefing is anticipated f r August or September 1982.

5.18 CONCLUSIONS AND RECOMMENDATIONS

5.18.1 Conclusions

The major conclusions resulting from this study are:

- o The Spaceplane concept is viable from a propulsion viewpoint.
- o Demonstrated ALRC 100 lbF and 5 lbF bipropellant engines, with minor modifications, meet the Spaceplane internal primary and secoondary propulsion requirements.
- o The selected Spaceplane internal propulsion system is a flexible concept which can be optimized for many conditions (e.g., various launch modes).
- o Extensive Spaceplane internal propulsion system modeling capability (prediction of propulsion system weight, c.g., envelope, performance, etc.) exists at ALRC.

5.18.2 Recommendations

The major recommendations resulting from this study are:

- o Establish firm vehicle and propulsion system design requirements.
- o Prepare Spaceplane preliminary propulsion system design, based on above requirements.
- o Begin development of a long-live N_2O_4-compatible, elastometric tank diaphragm.
- o Begin design modifications to ALRC 5 and 100 lbF thrusters for Spaceplae PCE and RCS application.

Several specific recommendations, which support the general recommendations listed above, are outlined below.

- o Technology for constructing the Spaceplane skin to incorporate active, liquid re-entry cooling exists at ALRC. Both regenerative and transpiration cooling appear applicable to the Spaceplane re-entry conditions. It is recommended that a study be made to determine the weight advantage of replacing the currently base-lined thermal protection system (tiles or ablattive or combination) with active, liquid cooling.
- o Evaluate the performance of PCE module scarfed nozzles. The use of scarfed nozzles could impact

PCE delivered Isp (especially at lower altitudes), PCE structural configuration and/or weight, etc.

- Perform a radiation heat transfer analysis of hot PCE module nozzles radiating to the Spaceplane vehicle structure. This may be important in determining the total heat load to the Spaceplane.

- Perform a more rigorous design and analysis of an electrical He heater to refine the preliminary electrical power requirements.

- Evaluate an advanced technology Spaceplane vehicle/propulsion system. Such a vehicle might incorporate, for example:

  - all graphite/polyimide structure
  - LF_2/N_2O_4 propellants
  - active cooling using EC/LSS byproducts (water)
  - high pressure, pump-fed propulsion concepts
  - integral tank/structure/TPS materials

- Perform a rigorous propulsion system controls analysis to better define a valve power requirements.

- Conduct a PCE test program to verify the performance prediction methodology currently in use.

- Investigate the use of existing electric motor-driven, storable propellant pumps for PCE application. These motor/pump sets, although designed for a different aerospace application, may be ideal for use on the Spaceplane.

- Perform a Spaceplane propulsion optimization based on the use of non-spherical tanks. Although non-spherical tanks will always be heavier than any equal volume, equal internal pressure spherical tanks, the resulting SP delta-V penalty of an optimized propulsion system using non-spherical tanks may be quite low.

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4. O'Brien, C.J., Unconventional Nozzle Tradeoff Study, Final Report, Contract NAS 3-20109, NASA CR-159520, ALRC, July 1979.

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6. Safety Policy and Requirements for Payloads Using the Space Transportation System, NHB1700.7.

APPENDIX A

MISCELLANEOUS TASKS

Spaceplane Heat Balance/Interfaces

From conversations with Hamilton Standard, it was concluded that about 100 lb of ECC/LSS weight might be saved by eliminating (1) the sublimator and (2) the water required to reject a maximum of 4,000 Btu/hr from the system (caused by biological and instrument heat generation). Quick calculations showed that the internal propulsion system propellant heat capacity was sufficient to receive this amount of heat for one day. However, subsequent calculations revealed that the major heat capacity of the propellants existed at temperatures above those of the heat sources. Further, it was seen that a double heat exchanger would be needed to provide redundant protection for the pilot from propellant vapors in the event of a single heat exchanger crack or leak.

Two solutions to this problem were considered. One was to use a small heat pump to drive the low temperature cockpit heat into the equal or higher temperature propellants. Rotating machinery is already being used in the ECC/LSS to remove heat. A second approach was to determine the thermal history of a non-insulated, orbiting spacecraft to see if heat removal is always a problem. A preliminary study was made of a conical vehicle in a 100-minute orbit about Earth, such that one side sees Earth, while the other side sees the sun at various angles of 60% of the time and shade (deep space) 40% of the time. Table A-1 summarizes the results of the study. A low emissivity/absorptivity (/) Spaceplane exterior results in little improvement; only about 600 Btu/hr of the 4000 Btu/hr internal heat generation is lost from the spacecraft during one Earth orbit. Conversely, a "grey" surface, such as ablative, grey tiles, etc., with high (/), results in excessive heat loss from the spacecraft during Earth orbit. In this latter case, the propellants freeze within about 6 hours, even with internal heating.

At least two solutions exist to this critical heat balance problem. In one solution, a proper amount of insulation used with the Spaceplane skin (maybe tiles alone) could provide enough net heat loss to eliminate the ECC/LSS heat rejection system, without freezing propellants and pilot. The second solution, selection of a proper Spaceplane surface (/), could provide the same result. Considerable Spaceplane volumetric savings could be made if surface insulation could be avoided.

For example, more than 10% of the Spaceplane volume would be occupied by an internal insulation layer only 1 inch thick.

Radiation Heat Transfer Analysis

The primary purpose of this task was to perform a preliminary radiation heat transfer analysis of the hot (operating) PCE module nozzles. A first step in this analysis was determination of PCE and RCS thruster exhaust gas temperatures. Figure A-1 is a plot of exhaust gas temperature as a function of propellant combination and nozzle area ratio. The N_2O_4/MMH line in the plot corresponds to a mixture of ratio (MR) of 1.65 and chamber pressure (P_c) of 142 psia which are the current N_2O_4/MMH PCE design values. The N_2O_4/PAAB-1 line in the plot corresponds to a MR of 1.20 and P_c of 100 psia. These are the current N_2O_4/PAAB-1 PCE design values.

It should be noted that the maximum chamber wall temperature experienced by the N_2O_4/MMH PCE modules will be approximately 2960°R (2500°F) because of the use of film cooling. Temperatures of the nozzle wall downstream of this maximum temperature point (usually near the throat) will be less than this maximum value. The major assumptions and results of this analysis are described below.

The major assumptions were:

- o Emissivity of the nozzle silicide coating is 0.8. This value is consistent with data for several different types of silicide.
- o A nozzle temperature of 2000°F was assumed. 4000° is an average value of free stream combustion gas temperatures (PCE module throat (T = approx. 5000°R); PCE module exit (T = approx. 3000°R)). This 4000°R value was then biased by 2000°R to account for the use of fuel film cooling. This thruster hot fire test data, which shows that the difference between combustion gas free stream temperature and nozzle material temperatures at any nozzle location is approximately 2000°R.
- o Each nozzle weighs approximately 1.31 lbM.
- o C_p of the nozzle material (Columbium) is 0.065 Btu/lbM°R.
- o The actual nozzle radiation surface was assumed to be a flat circular shape with an area equal to the PCE module nozzle exit area.
- o A solar flux of 442.2 Btu/hr.ft^2 (1.395 KW/M^2) at the Spaceplane aft end was assumed.
- o The nozzle coating surface was assumed to be grey (i.e., the emissivity is equal to the absorptivity).

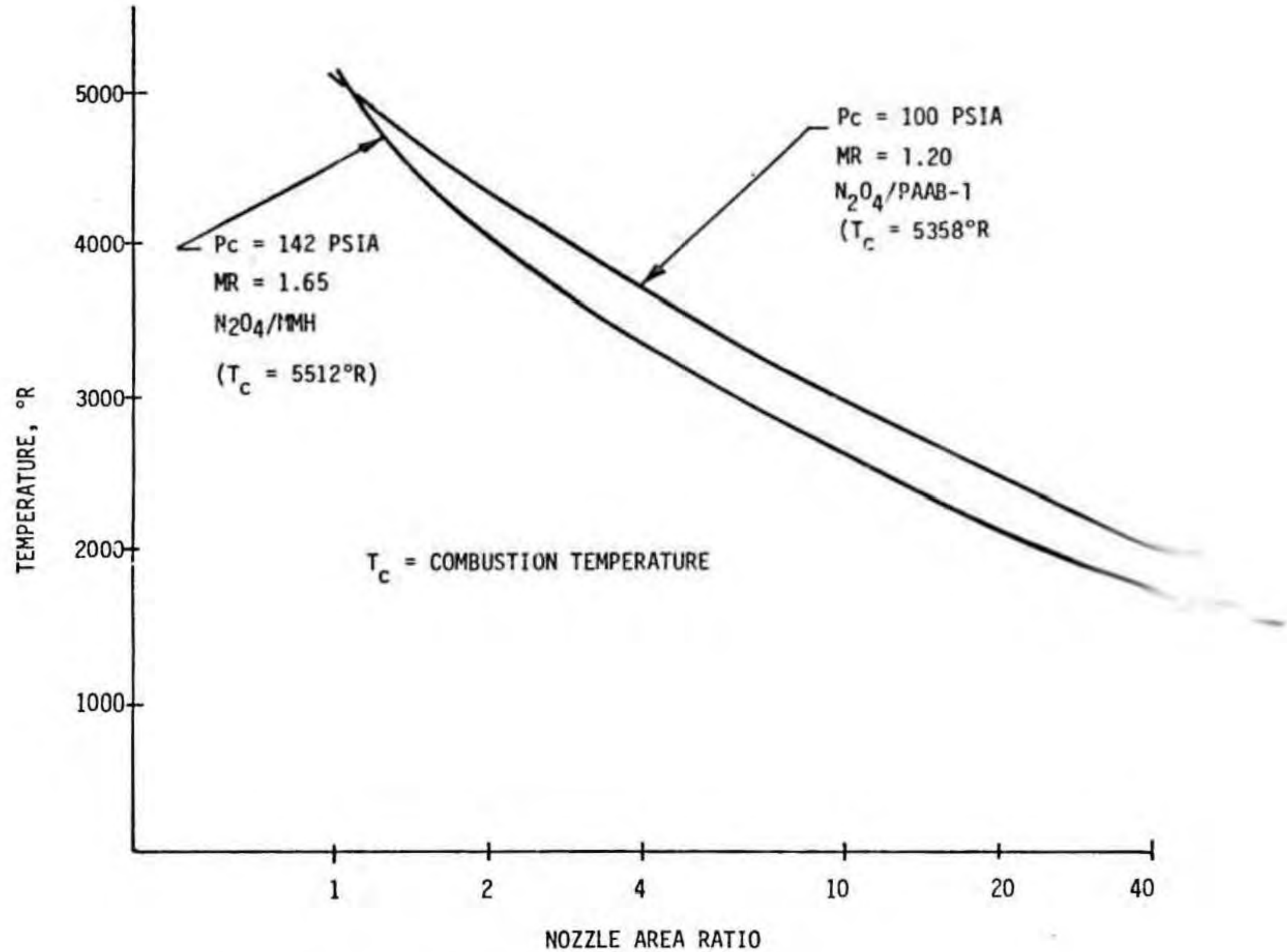

Figure A-1. Exhaust Gas Temperature vs Nozzle Area Ratio (N_2O_4/MMH and N_2O_4/PAAB-1)

- o The nozzles were assumed to be facing the sun immediately after PCE firing.

The most important results of the heat transfer analysis are noted below:

- o The nozzle material equilibrium temperature of approximately 251^{o}F is reached in approximately one to three minutes. Three minutes is considered very conservative for at least three reasons:

  - Radiative heat transfer from the outside nozzle wall facing dark space or the Earth is not taken into account.

  - The effective emissivity of a cavity, such as the interior of a PCE module nozzle, is greater, by a ratio of approximately 0.677:0.50, than a simple flat plate of equal area.

  - The insulating effect of the tile material on portions of the hot nozzle are not accounted for. This insulating effect would tend to increase the radiative heat transfer from the hot nozzle by restricting the conductive heat transfer to the Spaceplane vehicle structure that would otherwise occur.

- o The actual radiative heat transfer from the hot nozzles to the vehicle structure will be minimal, because of (1) the orientation of the nozzels and/or (2) the time material between the nozzles and vehicle structure.

- o The conduction heat transfer, from the PCE module mounting points to the Spaceplane vehicle structure will likely be higher.

A more extensive heat transfer analysis, to verify these preliminary results, is recommended.

6.0 INTRODUCTION

The objective of the Pratt & Whitney Aircraft effort, under the Spaceplane Examination, was to define an external propulsion system (module) for use with the Spaceplane on high Δv missions. The primary study ground rules were:

- o Propulsion system with Spaceplane, to be launched by a 65,000 lb. maximum capability space shuttle.
- o Propulsion system designed to maximize Spaceplane capability.
- o Propulsion system to be available in 1987/1988.

In the course of the study, any enabling or enhancing technology features were to be identified. The schedule followed by Pratt & Whitney, during this study, is shown in Figure 6-1 and the results of the study are detailed in subsequent sections of this report.

6.1 GENERAL CONSIDERATIONS

In order to achieve the objective of obtaining the highest credible ΔV capability for the Spaceplane external propulsion system (within the overall program's ground rules), several basic characteristics were identified and are discussed below.

- o Propellants: The significantly higher performance of hydrogen/oxygen over the storeable propellant combinations (e.g. N_2/O_4/UDMH, O_2/RP-1) has been well established. Higher performance propellants combinations (e.g., hydrogen/fluorine) were ruled out due to handling problems as not being within the "near-term State-of-the-Art" requirements. Therefore the external propulsion system was defined as one using cryogenic hydrogen and oxygen as the main propulsion propellants.
- o Performance: Propulsion system ΔV capability is characterized by the following equation:

$$\Delta V = (Isp)\ (g)\ \ln M_i/M_f$$

where:

ΔV	=	total velocity change capability; ft/sec
Isp	=	effective vacuum specific impulse, sec
g	=	gravitational acceleration, ft/sec^2
M_i	=	vehicle initial mass (at ignition), lb
M_f	=	vehicle final mass (at burnout), lb

PRATT & WHITNEY STUDY SCHEDULE

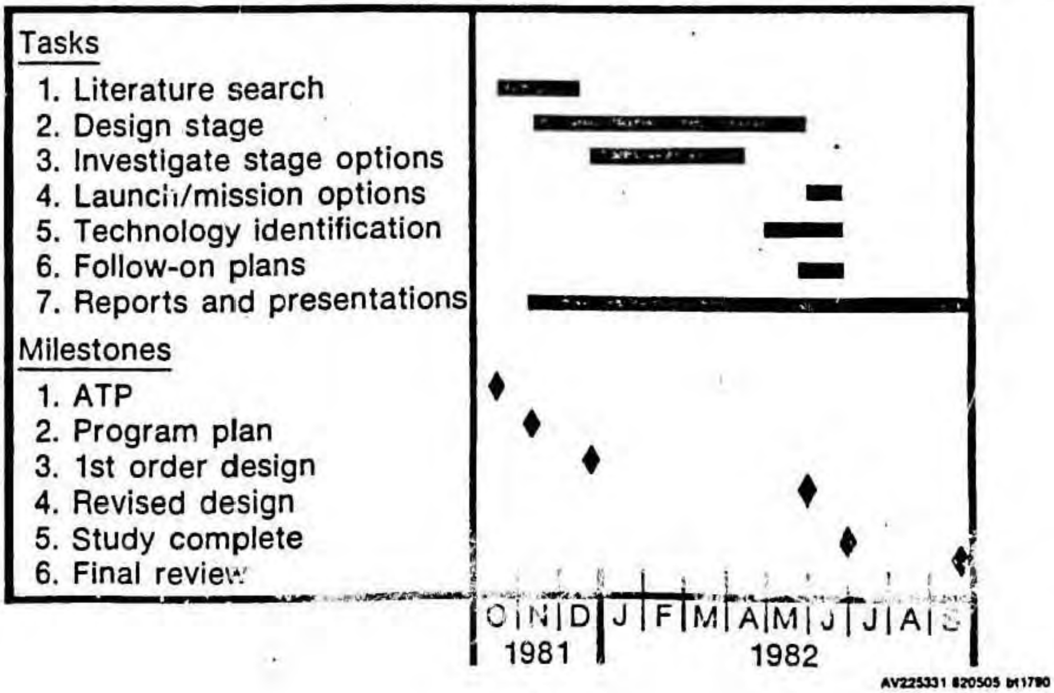

Figure 6-1

From this equation it is apparent that for a fixed initial mass (e.g., limited to 65,000 lb less Airborne Support Equipment weight by the Space Shuttle launch capability), maximum performance is achieved by the highest Isp and lowest final mass possible. This is illustrated by Figure 6-2 for typical specific impulse and burnout weight values.

- o Multiple Start: A typical spaceplane mission utilizing the external propulsion system (e.g., low-earth-orbit to geosynchronous orbit and return) would involve 4-6 main propulsion engine burns which might impart 4,000 to 8,000 ft/sec each to the vehicle. The propulsion system might also be required to provide many smaller impulsive burns for maneuvering/rendezvous with a multiple targets in earth orbit. The external propulsion systems should not therefore, be limited in starts other than by the mission consumables (e.g., main impulse propellants, pressurization gas, electrical power, etc.).

- o Reuseability: On missions where the spaceplane is returned to the vicinity of the shuttle by the external propulsion system, the vehicle could be reused on a later mission if provisions are made for re-insertion into the orbiter. An option to this would be provision for orbital re-fueling from tanks carried up by the shuttle along with a new refurbished spaceplane.

- o Operational Date: In keeping with the study groundrules of a vehicle which has a low development cost and a near-term operational capability (i.e., 1988 IOC), the vehicle definition must reflect relatively near-term State-of-the-Art technology. This, by implication, means that vehicle components must either be similar to current designs (or be readily demonstratable by about 1984) or have an acceptable fall-back design available.

- o Man-rating: Two mission safety issues are raised by the requirements of a manned system launched out of the shuttle cargo bay. The first and more straightforward is the requirement to not effect the shuttle system safety during launch and orbital checkout prior to release of the spaceplane and its external propulsion system. This should be rather a straight-forward task (even though complex) and has been analyzed for several systems (e.g., IUS, Centaur, Spacelab, etc.). The second safety issue is one of the overall spaceplane mission safety. This requires a definition of a manned mission reliability in terms of acceptable probability of mission success and of safe crewman return to the

EFFECT OF ΔV CAPABILITY AND ISP ON SYSTEM BURN OUT WEIGHT

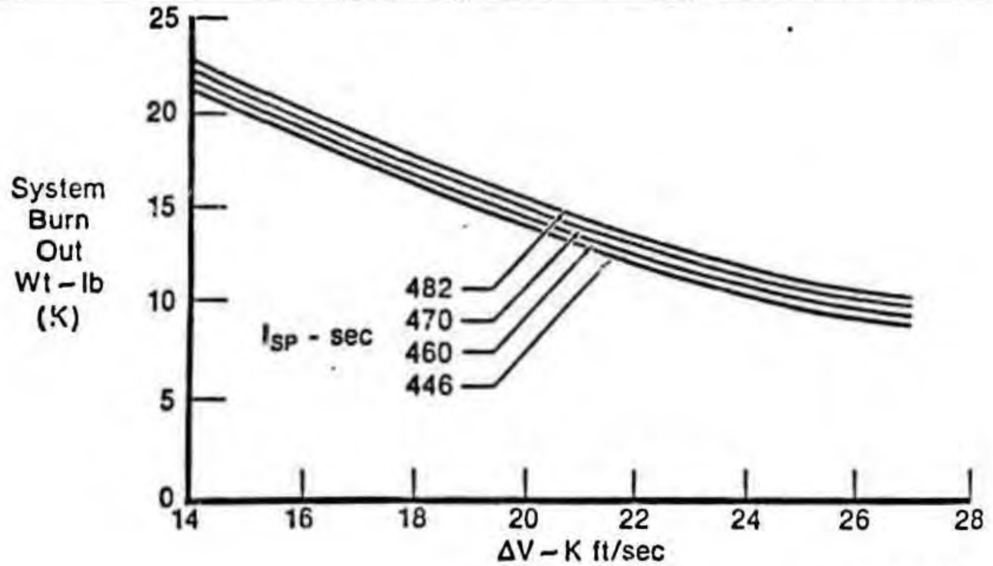

Figure 6-2

orbiter (or to a ground recovery site). All three OTV systems study contractors (References 7, 8, and 9) addressed the subject of manned mission safety but no consensus was reached other than all systems necessary for crew return should be at least single failure tolerant and that abort/rescue modes should be considered.

6.2 SPACE TUG STUDIES

During the early 1970's a number of studies were conducted to define a cryogenic upper stage which would operate out of the Space Shuttle in the 1980's (Reference 1-6). Typical of the later of these studies was that completed by the NASA-Marshall Space Flight Center as documented in report MSFC 68M00039, dated 15 July, 1974 (Reference 6).

The "MSFC Tug" was designed around requirements of retrieving a 3500 lb. payload from Geosynchronous Orbit (GEO) and retrieving it to the Space Shuttle. Figure 6-3 shows the inboard profile of this vehicle.

6.3 OTV STUDIES

During 1978-1980 several additional cryogenics stage "Phase A" studies were conducted for the NASA (References 7, 8, and 9). One of these, conducted by the Boeing Aerospace Company (Reference 9), was configured to meet the OTV program manifest shown in Figure 6-4. The Boeing OTV characteristics are summarized in the following section.

6.3.1 OTV Operational Description

The Orbital Transfer Vehicle (OTV) is a cryogenic (LO_2/LH_2) propulsive stage carried to low earth orbit within the payload bay of the Space Shuttle Orbiter. The design of the OTV is optimized for use with the Shuttle Orbiter and extends the reuse capability of the Space Transportation System to include upper stage. The OTV also is operated in the expendable mode when this mode optimizes the overall STS effectiveness.

To place in perspective the key design requirements presented later in this section, a brief discussion of OTV operations for a typical payload delivery mission with OTV recovery follows.

Following checkout, the OTV, its airborne support equipment, and its payload are mated and undergo integrated tests. The integrated assembly is then transferred to the launch pad and installed in the Shuttle Orbiter where propellant loading of the launch vehicle and the OTV are accomplished.

Following launch and circularization to a 160 nautical mile orbit with an inclination of 28.5^{o}, the Orbiter payload doors are opened and the OTV undergoes a predeployment checkout. The OTV/spacecraft is deployed and after the Orbiter and OTV/spacecraft separation distance is safe, the OTV attitude control system is enabled. The OTV/spacecraft phases in the 160 nmi orbit, accomplishes a phasing/plane change burn, coasts for one revolution in the phasing orbit and, at perigee, initiates the GEO transfer

1974 MSFC BASELINE FULL CAPABILITY SPACE TUG

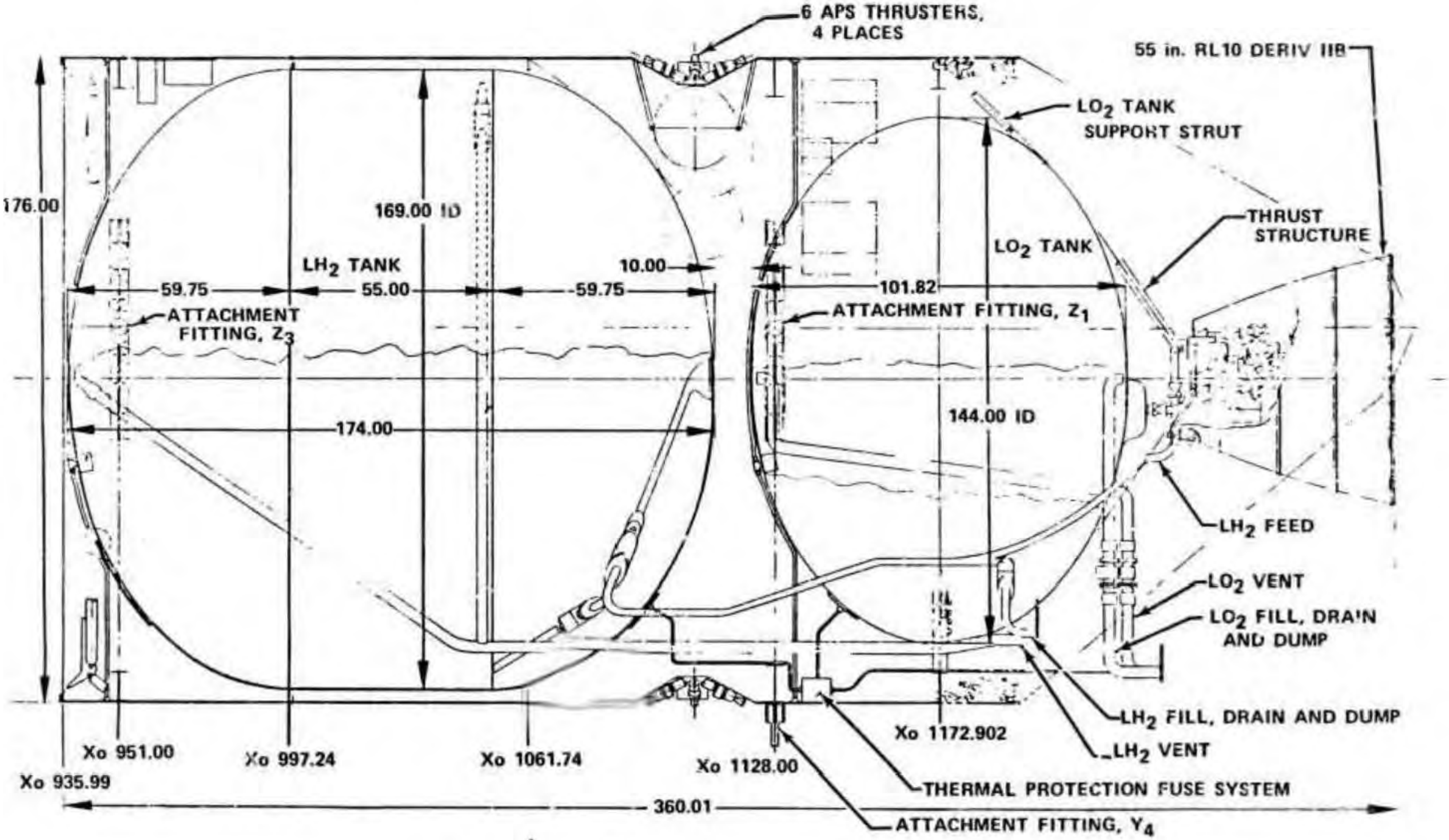

Figure 6-3

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743007

NO.	MISSIONS	TOTAL	MISSIONS/YR															
			1985	1986	1987	1988	1989	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000
1	GEO PLAT DEMO	1			1													
2	GEO PLATFORM	12								1▲	2▲	2▲	1▲	1▲	2▲	2▲	1▲	
3	PLANETARY	10			1		1		1	1	1▲	2▲		1		1		1
4	MULTIPLE PLD DEL	10			2	1		2		1		1		1		1		1
5	DoD CLASS 1A	26			1	2	2	1	2	2	2	2	2	2	2	2	2	2
6	DoD CLASS 1B	12			1	1	1	1	1	1	1	1			1	1	1	1
7	DoD CLASS 2	4						1▲	1▲	1▲	1▲							
8	DoD CLASS 3	8								*1▲		*1▲	*1▲	*1▲	*1▲	*1▲	*1▲	*1▲
9	SPACE BASED RADAR	8A/2B				1A		1A▲	2A▲	2A▲	*1B▲	*1B▲				1A▲		1A▲
10	PERSONAL COMM	4									*2▲		*2▲		*2▲		*2▲	
11	UNMANNED SERVICING	29							1▲		1▲	2▲	2	3 ▲1	2 ▲1	4 ▲1	4 ▲1	4 ▲2
12	MANNED SERVICING	13								1▲	1▲	1▲	1▲	1▲	2▲	1▲	3▲	2▲
13	DEBRIS REMOVAL	4								1▲		1▲		1▲		1▲		
14	GEO STATION	2															2 ▲1	
15	GEO STATION SUPPORT	8															2▲	4▲
28	GEO PLAT SERVICING	6									1▲		1'	1▲		1▲	1▲	1▲
	25K SHORT STAGE	13			1	2	1	1	1	1	1	1			1	1	1	1
	43.5K IOTV	102			6	3	3	5	7	8	8	8	8	8	8	12	11	10
	73K ADVANCED OTV	47								3	4	4	5	4	8	4	8	8
	ALL PROPULSIVE	39			4	3	3	3	3	3	2	3	2	3	2	3	2	3
	EXPENDABLE	20			2	2	1	1	1	2	1	1		1	1	2	3	2
▲	AERO BRAKED	103						2	4	7	10	10	8	8	10	12	15	15
	TOTALS	162	0	0	6	5	4	6	8	12	13	14	10	13	13	17	21	20

* SEPARATE LAUNCH FOR OTV & P/L

Figure 6-4

Selected Program Mission Manifest (Revision 2 Nominal Mission Model) orbit injection burn. Mid-course corrections are accomplished if required, and at 19,300 nmi a GEO circularization burn is performed followed by a coast and orbit trim period.

The payload is released at GEO and after phasing for the proper nodal crossing, the GEO to LEO transfer orbit injection burn is accomplished. A phasing orbit is used to accomplish proper rendezvous phasing with a Shuttle Orbiter. The rendezvous orbit circularization burn is performed and, after orbit trim, the OTV achieves a stable attitude and remains passive during recovery by the Orbiter.

The OTV is returned to the Orbiter payload bay using the remote manipulator system, latched into the airborne support equipment structual adapter, stowed into the payload bay, and returned to the launch site for subsequent refurbishment for a later flight.

During the period that the OTV is within the Orbiter payload bay, command and control is accomplished by GSE and Orbiter systems prior to launch and through Orbiter systems after launch. When deployed outside the Orbiter, command and control is accomplished by a STDN/TDRS compatible RF link. The OTV is capable of autonomous mission operation and is capable, by addition of a kit, of providing a secure communication link if required.

6.3.2 Configuration

The Orbital Transfer Vehicle is made up of the following subsystems:

- structural-includes an external, load-bearing body shell, a LH_2 tank, and a LO_2 tank
- thermal control-both active and passive to regulate heating loads
- avionics-redundant and includes all elements (i.e., guidance and navigation, data management, communications, and instrumentation)
- power supply and distribution-features redundant $O2/H2$ fuel cells and distribution and control units
- propulsion-consists of a RL-10 derivative II B main engine with an extendable nozzle, electromechanical actuator thrust vector control, and propellant delivery, pressurization, and vent subsystems
- attitude control-consists of four hydrazine thruster pods and associated tankage and control.

The OTV is 30.7 feet long including the 70-inch-long main engine (in the retracted position). The vehicle diameter is 176 inches. The volumes of the main propulsion system propellant tanks are 587 and 1,720 cubic feet for the LO_2 and LH_2 tanks, respectively. An inboard profile of the configuration is shown in Figure 6-5.

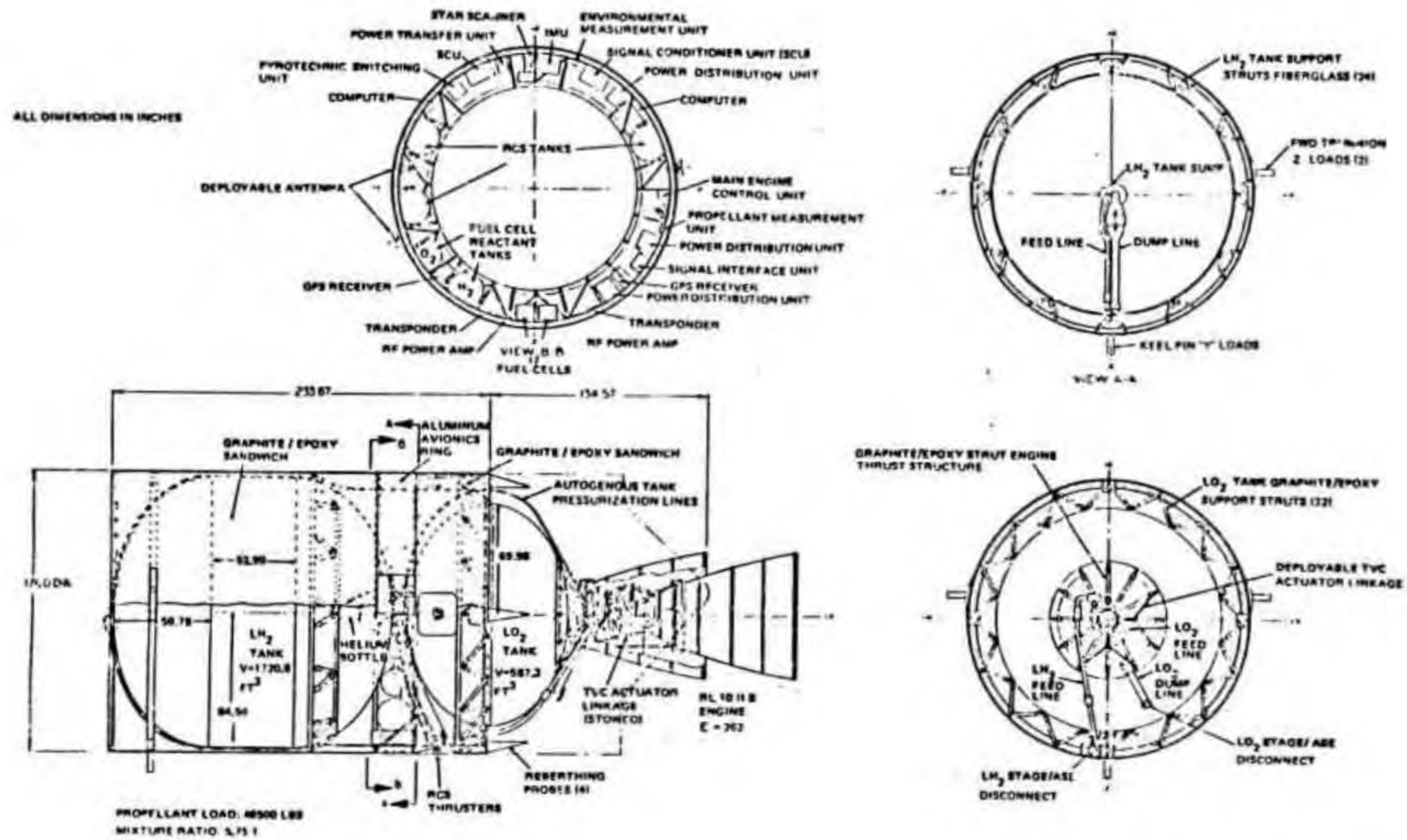
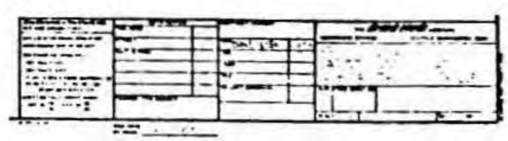


Figure 6-5

OTV IN-BOARD PROFILE



The airborne support equipment (ASE) is that portion of the OTV system flight hardware which remains in the Shuttle Orbiter payload bay when the OTV is deployed. The ASE provides for the adaptation of the OTV to the Orbiter payload bay and for the distribution of all loads in the X-direction to the Orbiter. The ASE provides for the interfaces for OTV fluids, electrical, and avionics subsystems to Orbiter provided interfaces. The ASE also provides a reserve electrical power source, a control and monitor panel mounted in the Orbiter aft flight deck, and a pressurization source used for the dumping of propellant in the event of an RTLS abort during ascent to low earth orbit.

Structures-An exploded view of the structures subsystem is shown in Figure 6-6. The LH_2 and LO_2 tanks are made from 2219-T87 aluminum and are supported by struts within the external body shell. The body shell consists of four sections, three of which are constructed of graphite/epoxy core. The remaining body shell section is the avionics ring assembly which is constructed of aluminum and provides for the installation of avionics, electrical power, and attitude control subsystem tankage (see Figure 6-7). The main engine is installed on the thrust structure which is supported by struts connected to a thrust ring incorporated into the lower bulkhead of the LO_2 tank.

Thermal Control-Both active and passive techniques are used to provide thermal control of the OTV. Thermal control of the fuel cells is provided by an active thermal conditioning system consisting of a Freon 11 fluid loop with a radiator, located on the body shell exterior, and the associated pumps, valves, and control elements. The passive thermal control techniques include insulation blankets, thermal control coatings, and selected radiative surfaces. The thickness of the aluminum used for the avionics ring assembly is controlled to provide for proper heat flow from internally mounted components and its exterior surface is covered with flexible optical solar reflector (FOSR) to provide the radiative surface. Electrical heaters are provided for RCS components and avionics equipment as required. The LH_2 and LO_2 tanks are insulated using MLI. The MLI consistes of layers of doubly aluminized kapton with a dacron net spacer. Twenty-three layers of MLI are used on each of the tanks. The MLI wrapped tanks are enclosed within purge barriers which are purged with dry gas (helium for the LH_2 tank and nitrogen for the LO_2 tank) prior to launch.

Avionics-The avionics subsystem design for the OTV is based on the maximum use of components used on the IUS. The guidance and navigation components include an inertial measurement unit and a star scanner, both of which are internally redundant and are powered from redundant power buses. Included within the communications area are redundant RF links which are NASA STDN/TDRS compatible. Deployable pairs of antenna pods are diametrically mounted in the avionics ring assembly. Each RF link contains a 20-watt S-band power amplifier and a STDN/TDRS transponder. In the normal operating mode both transponders are operating in the receive mode, one transponder is operating in the transmit mode, and its corresponding RF power amplifier is on.

The data management subsystem contains two computers, two signal conditioner units, a signal interface unit, and a thrust vector control unit. Each computer has 65k memory capability, an operational capability

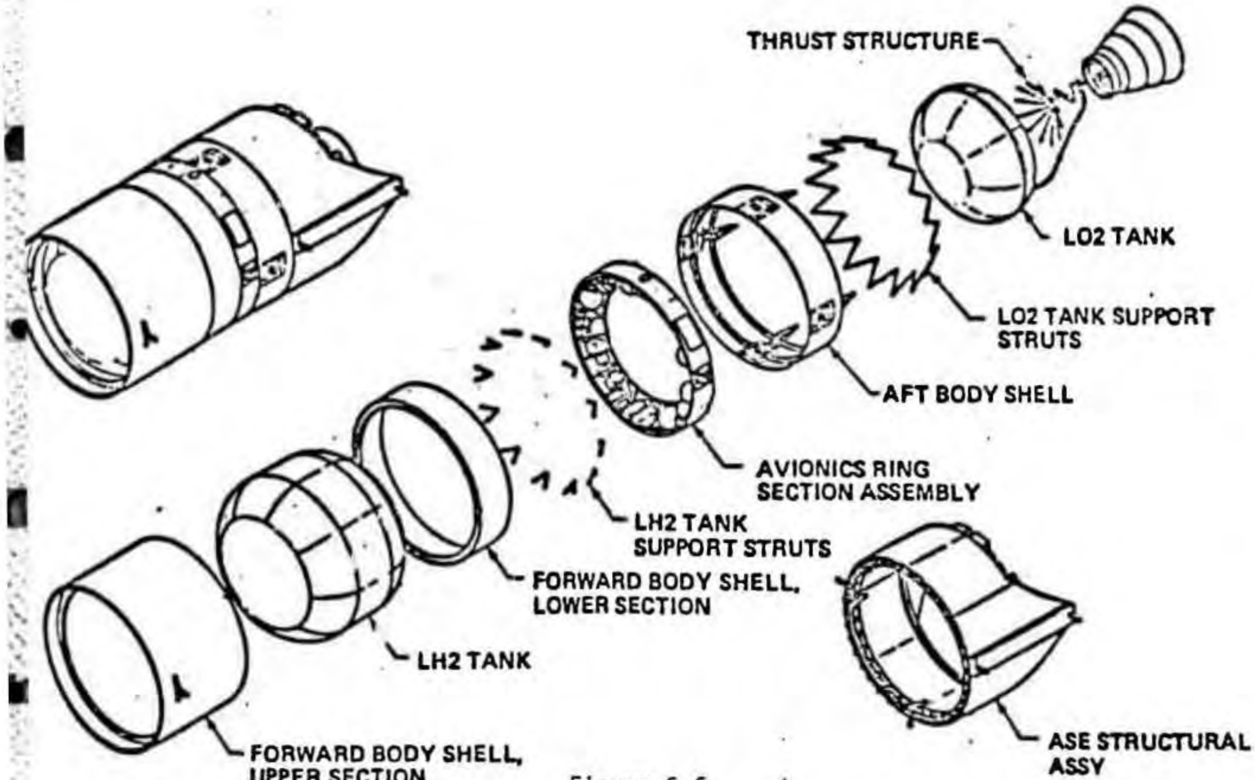


Figure 6-6
Exploded View of Structural Subsystem

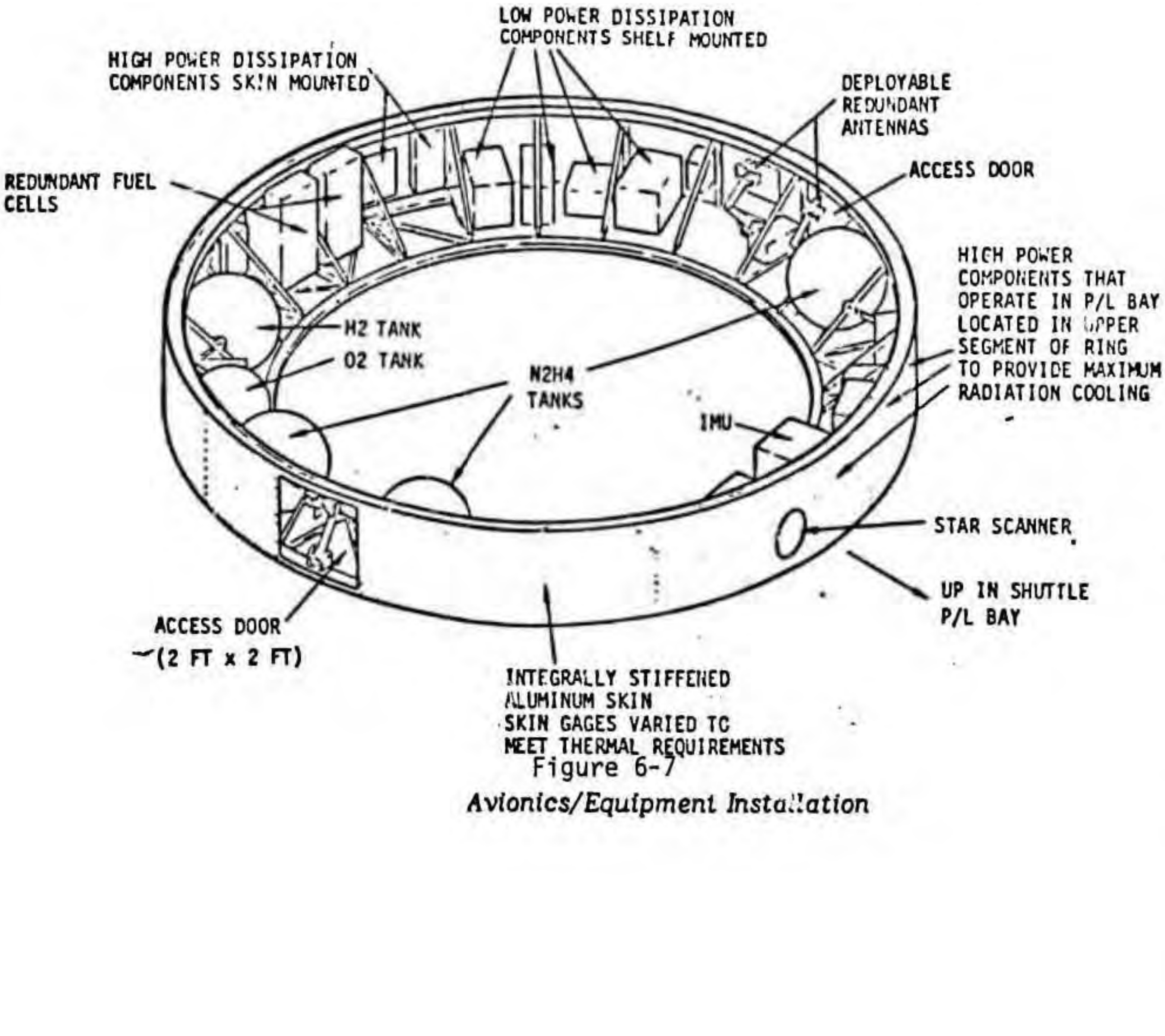


Figure 6-7

Avionics/Equipment Installation

of 550k operations per second, and performs all computation and data processing associated with guidance, navigation and control, command decoding, telemetry formatting, redundancy management, and communications. The signal conditioner units provide for measurement sampling and conditioning, performs redundancy management, and provides the interface between computer generated commands and other OTV subsystems. The signal interface unit performs buffering, switching, formatting, and filtering of communications signals.

The instrumentation subsystem provides for status monitoring of OTV subsystems. The preliminary measurement list contains approximately 160 analog measurements and over 400 discrete (bi-level) measurements. Included within the instrumentation are capacitance liquid level measurements used for propellant loading and consumption monitoring during main engine firing and also propellant depletion sensors.

Power Supply and Distribution-The primary power source for the OTV is a set of redundant 2kw-nominal, 3.5kw-peak fuel cells. The fuel cells are actively redundant (i.e., both operating in the normal mode). Each fuel cell is capable of providing normal mission power. A 25-amp-hour nickel-hydrogen battery is provided for backup power and for smoothing of line transients. Distribution and control is provided by redundant power distribution units and a power transfer switch. The power distribution units are used for switching of highly inductive or high current loads. The power transfer unit is for control of the spacecraft power from GSE, Orbiter, and OTV power sources.

Propusion-The main engine is a Pratt & Whitney RL10 derivative II B engine with a retractable nozzle. The engine has the capability of operating in three modes: tank head idle, pumped idle, and mainstage. The propellant delivery system LO_2 and LH_2 feedlines are of aluminum and include bellows expansion joints to compensate for thermal expansion and engine gimballing. Screened surface start baskets are located at the inlets to each line.

Pressurization of the tanks is accomplished using an autogeneous pressurization system. Two separate propellant vent/relief systems are installed for each tank; one for use when stowed in the Orbiter payload bay and one for use in space.

Thrust vector control is provided by two electroemchanical ball-screw linear actuators. Each actuator is equipped with redundant electric motor drive.

Attitude Control-The attitude control subsystem (ACS) uses hydrazine monopropellant with a blow-down pressurized system. The ACS design uses 12 thrusters positioned around the aft OTV body shell to provide rotational moments about the vehicle c.g. in pitch, roll, and yaw.

The propellant storage consists of three 21-inch diameter tanks each with a storage capacity of approximately 120 pounds of hydrazine. The tanks are wrapped with MLI for thermal insulation and are connected to the ACS feed manifold.

The thrusters use a catalytic decomposition gas generator and produce 30 pounds of thrust at 380 psia supply pressure and at the 100 psia supply pressure produce 8 pounds of thrust. Heaters are installed on ACS valves, lines, decomposition chambers, and the hydrazine tank exterior.

Airborne Support Equipment-The airborne support equipment (ASE) portion of the OTV system provides for the interfacing of the Orbital Transfer Vehicle to the Shuttle Orbiter. The ASE consists of the structual assembly, fluids subsystem, electronics, batteries, and cabling. The ASE structure is fabricated from graphite/epoxy. The structual assembly pivots about the ASE-to-Orbiter trunnions using a tilt mechanism. The structual assembly interfaces with the OTV through a circular ring frame with 36 latching mechanisms and fluids and electrical umbilibals. On deployment the tilt mechanisms rotate the combined OTV/ASE to the release position, the latching mechanisms are released, and the OTV is deployed by a series of springs around the mating circular ring frames. On retrieval, the OTV is maneuvered by the Orbiter RMS into the ASE using the OTV berthing probes, the latching mechanisms are relatched, and the mated assembly is rotated back down into the Orbiter payload bay.

The ASE fluid system consists of Orbiter to OTV fill, drain, dump, vent lines, umbilicals, and pressurant gas storage for use in the event that dumping of OTV propellant is required during an RTLS abort. Redundant valving is incorporated into all lines as required to meet Orbiter payload bay safety requirements. The pressurant gas used for abort dump and ASE valve control is helium. Approximately 100 pounds of helium gas storage is provided by bottles mounted on the ASE structural assembly.

The electrical/avionics system provides for backup electrical power and control, ASE status monitoring command and control, cabling and interfacing cabling between OTV, space craft systems, and orbiter-provided payload bay data interfaces. A power control unit is provided to select between ground power (supplied via the Orbiter) or Orbiter power. The ASE batteries float online to prevent power dropout during the periods when the ASE is powered up. Power distribution units are included to provide switching capability of large or highly inductive loads. A control unit mounted in the aft flight deck performs the functions of the monitoring of critical ASE and OTV functions and for the control of the ASE during the deployment and berthing processes.

6.3.3 Weight Summary

A weight summary for the initial OTV (and its ASE) is presented in Table 6-1 for the reusable delivery mission to GEO. A weight growth margin of 633 pounds is included in the definition of OTV dry weight. This margin reflects 5% for existing hardware and 15% for new design.

6.3.4 Mass Characteristics

The longitudinal center of gravity compatibility of the OTV/ASE/spacecraft payload with Orbiter requirements is presented in Figure 6-8. As shown, Orbiter requirements are met with the exception of the ASE return only case. The current estimate of total ASE weight (contractor plus government) is 5,500 pounds. This weight is located 50.7 feet aft of the

TABLE 6-1

Summary Weight Statement (GEO Delivery Mission)

Structure	2464	
Thermal Control	315	
Avionics	649	
Power Supply/Distribution	634	
Propulsion	937	
Attitude Control	138	
Weight Growth	633	
(OTV Dry Weight)	(5770)	
Residuals	476	
Reserve Fuel Cell Reactants	15	
(OTV Mission Sequence Final Weight)	(6261)	
Reserve Attitude Control Propellant	22	
Reserve Main Impulse Propellant	258	
Inflight Losses	181	
Fuel Cell Reactants	73	
Attitude Control Propellant	223	
Mail Impulse Propellant	46,247	
(OTV Gross Weight)	(53,265)	
Payload	6,170	
(Start Mission Weight)	(59,435)	
Contractor ASE	4,300	
Government ASE	1200	
ASE Ballast	120	
(Total Launch Weight)	(65,095)	
OTV Mass Fraction		0.873

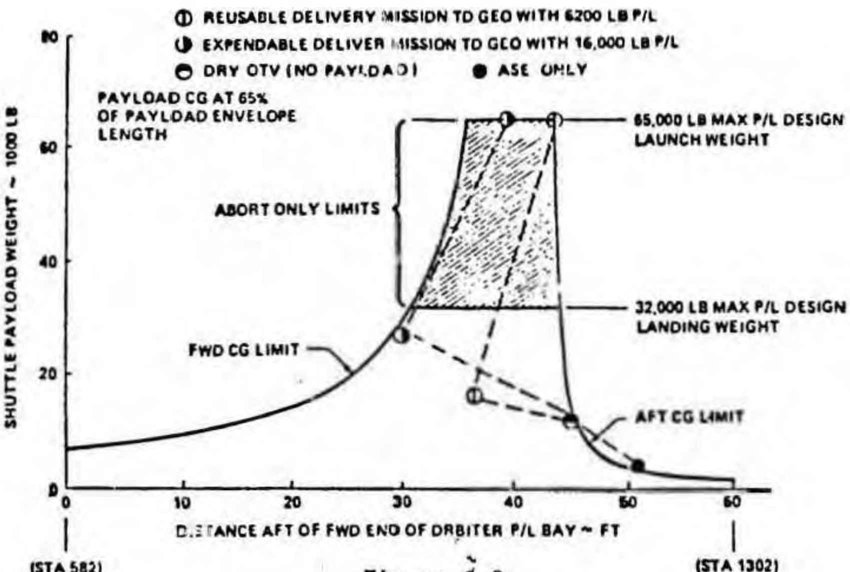


Figure 6-8
Longitudinal CG Compatibility

forward end of the payload bay, and is 1.6 feet aft of the c.g. envelope. The addition of approximately 160 pounds of ASE ballast (on the aft flight deck) is required to satisfy c.g. limits.

6.3.5 Performance Capabilities

The initial OTV primary design reference mission is payload delivery to GEO in the reusable mode. Secondary mission capabilities include expendable GEO delivery (in both normal and low G modes), insertion into GEO transfer, and planetary missions. Table 6-2 summarizes the GEO performance capabilities of the initial OTV.

Table 6-2

OTV-GEO Performance and Weight Summary

Mode	P/L Wt.[1]	Prop. Wt.[2]	Start Mission Wt.
Reuseable	6170	46500	59410
Expendable (offloaded for 65K STS)	16390	36300	59390
Expendable (fully loaded)	22890	46500	76160
Low G Expendable (off-loaded for 65K STS)	15970	36710	59380
Low G Expendable (fully loaded)	22030	46500	75300
GEO Transfer (off-loaded for 65K STS)	22870	30000	59400
GEO Transfer Injection (fully loaded)	45360	46500	98450

1 Nominal plus reserves

2 OTV plus payload plus propellant (does not include ASE)

6.4 Centaur Studies

Numerous studies have been conducted by General Dynamics over the last decade related to integrating the existing Centaur stage into the Space Shuttle system. These studies resulted in the design of the Centaur F vehicle (Reference 10), which was optimized for the launch of the Galileo mission in 1985 or 1986. A summary description of the Centaur F Vehicle is given in the following section.

6.4.1 Centaur F Vehicle

The Centaur vehicle (Figure 6-8) consists of a 10-foot-diameter LO_2 tank that transitions to a 14 foot, 2-inch-diameter LH_2 tank. The cryogenic tanks are insulated with combinations of helium-purged foam blankets and radiation shields. The forward end of the vehicle consists of bolted-on cylindrical stub adapter and a conical equipment module, which provides mounts for all vehicle electronic packages. The aft end of the vehicle consists of a cylindrical aft adapter and a pyrotechnic separation ring.

The vehicle avionics system performs the functions necessary for autonomous control of the Centaur vehicle from a pre-defined safe separation distance following Centaur deployment from the Orbiter through postseparation maneuvers.

The GN&C system for Centaur F is the Atlas/Centaur GN&C system with minor modifications. The system was completely redesigned to NASA Hi-Rel standards in 1972. A star scanner has been added for attitude update, and the computer memory is now addressable via the TT&C up/down link.

The TT&C system is compatible with the Orbiter and Tracking and Data Relay Satellite System (TDRSS) links, and permits data uplink via the Orbiter or TDRSS.

Electrical power to safety-related avionics control functions is inhibited until the Centaur is a safe distance from the Orbiter.

6.4.2 Centaur F Integrated Support System (CISS)

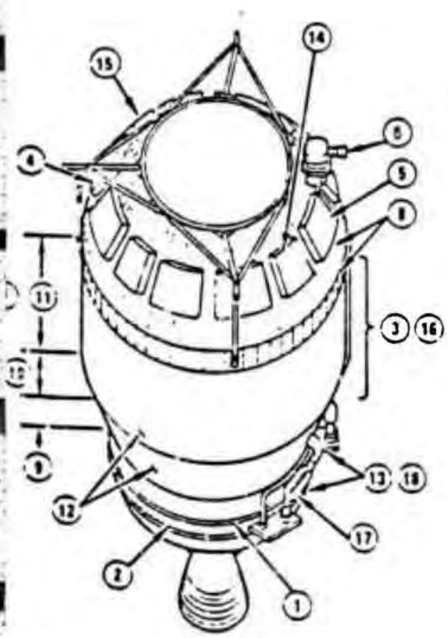
The CISS consists of a Centaur support structure (CSS), a deployment adapter, and the associated CISS electronics and fluid lines (Figure 6-10). the CSS adapts the Centaur vehicle and deployment adapter to the Orbiter through a five-point support system. The deployment adaptor attaches to the aft end of Centaur at the separation ring and to the CSS through two rotation trunnions.

During deployment, the vehicle is rotated to its separation attitude by a rotation mechanism attached to the deployment adaptor.

Fluid systems ducting and flexible sections are provided to interconnect the various propellant tank service lines to their associated Orbiter over-board service ports. Flexible sections permit the Centaur to be rotated to the deployment position while maintaining all safety-related systems in a connected and functional state.

Helium storage spheres and two-failure-tolerant pressurization and pressure regulation systems supply all helium for pressurizing Centaur tanks, actuating vent and dump system purges to manage Centaur propellants safely.

CISS avionics performs all control functions for vehicle safety while the Centaur is attached to the Orbiter and for deployment.



INTERFACE COMPATIBILITY & MISSION REQUIREMENTS

1. NEW AFT ADAPTER
2. NEW SEPARATION RING ENCOMPASSING LOCKHEED "SUPER* ZIP" SEVERANCE SYSTEM
3. NEW LH_2 TANK SIDEWALL INSULATION BLANKET/RADIATION SHIELD
4. NEW TDRSS-COMPATIBLE S-BAND TRANSPONDER AND RF SYSTEM
5. MODIFICATIONS TO DIGITAL COMPUTER UNIT (DCU) INPUT/OUTPUT FUNCTIONS
6. MODIFICATIONS TO INERTIAL MEASUREMENT GROUP (IMG) AND NEW STAR SCANNER FOR ATTITUDE ALIGNMENT AND UPDATE
7. NEW ZERO-G VENT SYSTEM
8. NEW EQUIPMENT MODULE & STUB ADAPTER
9. ADDED CYLINDRICAL SECTION TO LO_2 TANK
10. NEW CONICAL TRANSITION TO 170 IN. DIAMETER LH_2 TANK
11. NEW 170 IN. DIAMETER LH_2 TANK CYLINDRICAL SECTION AND FORWARD BULKHEAD
12. MODIFIED PROPELLANT SENSING SYSTEM

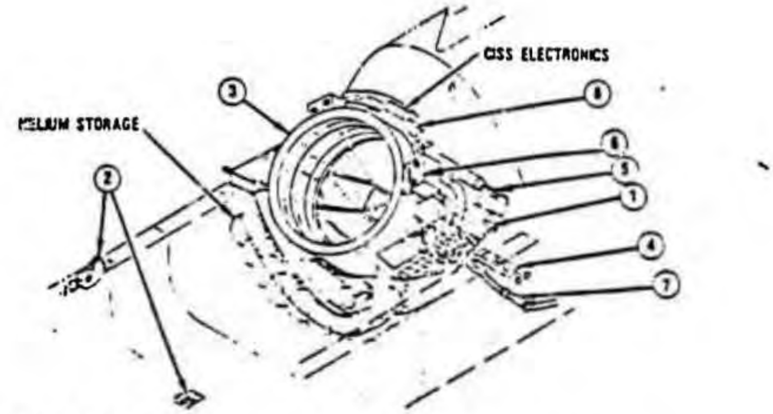
SAFETY CONSIDERATIONS

13. ADDITIONAL VALVES FOR TANK PRESSURIZATION AND VENTING, AND PROPELLANT SUPPLY
14. NEW RF COMMAND LINK AND REDUNDANT TIMERS
15. ADDITIONAL TANK PRESSURE TRANSDUCERS AND REVISED INSTRUMENTATION FOR REDUNDANT DATA PATHS
16. MATERIAL CHANGES
17. NEW PROPELLANT DUMP SYSTEM
18. MODIFICATIONS TO HELIUM PURGE SYSTEM

1588-35A

Figure 6-9

Centaur F vehicle modified from Atlas/Centaur vehicle.



INTERFACE COMPATIBILITY

1. FIVE-POINT AFT SUPPORT SYSTEM BETWEEN CISS AND ORBITER (STANDARD FITTINGS)
2. THREE-POINT FORWARD SUPPORT SYSTEM BETWEEN CENTAUR AND ORBITER (STANDARD LATCHES WITH STOPS ADDED FOR PASSIVE RESTRAINT)
3. DEPLOYMENT ADAPTER WITH SPRING THRUST AND SUPER-ZIP SYSTEM TO SEPARATE CENTAUR
4. FLUID SERVICING FOR LO_2 AND LH_2 FILL & DRAIN, PRELAUNCH/FLIGHT VENTING, ABORT DUMP AND BULKHEAD CAVITY RELIEF
5. DEPLOYMENT ADAPTER ROTATION SYSTEM (TO 45 DEGREES)
6. FUEL AND OXIDIZER UMBILICAL PANELS

SAFETY CONSIDERATIONS

7. PROPELLANT ABORT DUMP
8. FIVE AUTONOMOUS CONTROL UNITS PROVIDE TWO-FAILURE TOLERANT CONTROL

1588-17A

Centaur integrated support system.

Figure 6-10

Two-failure-tolerant control is achieved with five strings of microprocessor-control avionics, associated sensors and controllers.

6.5 MAIN ENGINE STUDIES

Throughout the 1970's numerous studies have been conducted by three propulsion contractors (Aerojet, Pratt & Whitney and Rocketdyne) of engines applicable to Shuttle cryogenic upper stage vehicles (References 11 - 21). The spectrum of engines covered by these studies range from the current RL10 and its near-term derivations to advanced high-performance expander and stage-combustion cycle engines. Figure 6-11 shows a representative summary of these engines at a common operating point (15,000 lb. thrust, 6.0:1 mixture ratio) and at a common installed length (for those engines having two-position nozzles) for comparison purposes.

The requirements of the main propulsion engine are similar to those discussed for the overall-vehicle in Section 6-1, namely:

- High Specific Impulse
- Light Weight
- Multiple Start
- Man Rateable
- Available ~1988
- Low Thrust Capability
- Reuseable

From those engines meeting all the requirements, the only engine available within the 1988 operational capability requirement is the RL10 Derivative IIB (the RL10A-3-3A and RL10 Derivative IIC engines were primarily eliminated due to a lack of a low thrsut capability). In addition, the higher performance of the advanced expander cycle engine would only produce approximately 2% higher ΔV capability which is probably not worth its considerably increased cost and risk.

6.6 EXTERNAL PROPULSION SYSTEM DESCRIPTION

From the studies available on cryogenic propulsion systems for shuttle upper stages (references 1-10), the major concepts available for use as a Spaceplane external propulsion system definition starting point are the Centaur F, the Orbit Transfer Vehicle and the Space Tug. A summary of the basic charactertistics of these stages is given in Figure 6-12.

The Centaur F has the most detailed design background (it is derived from the operational Centaur D-1A and has been evaluated at the PDR level, while the others are only Phase A studies), and is the earliest operational vehicle. Primarily for these reasons, the Centaur F was selected as the external propulsion system design starting point. The following subsections describe a Centaur F modified (hereafter called Centaur SP), to meet the requirements of a high ΔV Spaceplane mission.

CANDIDATE ENGINE CHARACTERISTICS

	RL10A-3-3A	DERIVATIVE IIB	DERIVATIVE IIC	CATEGORY IV	ADVANCED EXPANDER
FULL THRUST (VAC), LB	←		15,000		→
MIXTURE RATIO, NOMINAL	←		6.0 $\pm$ 0.5 P.U. Capability		→
CHAMBER PRESSURE, PSIA	425	400	400	915	1505
SPECIFIC IMPULSE, SEC	440.4	459.8	458.6	471.7	481.0
REQUIRED INLET CONDITIONS					
FUEL, NPSP, PSI	2	0.5	2	0	0.5
OXIDIZER, NPSP, PSI	4	4	4	0	1
INSTALLED LENGTH, IN.	70	←		55	→
WEIGHT, LB	305	392	374	371	410
NOZZLE AREA RATIO	61	205	205	388	570
ENGINE LIFE (TBO), FIRINGS/HR	10/1.25[1]	190/5[2]	10/1.25[1]	300/10[2]	300/10[2]
ENGINE CONDITIONING	OVERBOARD DUMP COOLDOWN	TANK-HEAD IDLE	OVERBORAD DUMP COOLDOWN	TANK-HEAD IDLE	TANK-HEAD IDLE
MANEUVERING THRUST CAPABILITY	NO	YES	NO	YES	YES
AVAILABILITY[3]	CURRENT	∿1988	∿1986	∿1991	∿1992

Notes:
1. Expendable engine, one mission life
2. Reuseable engine, time between overhaul~
3. Assumes development ATP 1/1/84

Figure 6-11

CYROGENIC VEHICLE COMPARISON

	Centaur F	OTV	Space Tug
Contractor	GDC	GDC/Boeing	NASA-MSFC
Date	Current	1980	1974
Propulsion concept	All propulsive	Aero braked	All propulsive
Main engine	RL10A-3-3A(2)	RL10 Deriv IIB	RL10 Deriv IIB
Max propellant wt, lb	46,000	55,000/46,250	50,780
Dry weight, lb	5776	9880/5770	5150
Status	PDR complete	ϕA study	ϕA study
Operational date	1985	ATP + 7 yrs	ATP + 5 yrs
Design mission	Galileo	Leo-Geo (deploy) Geo-Leo (return)	Leo-Geo (deploy) Geo-Leo (return)

Figure 6-12

6.7 STAGE CONFIGURATION

The basic Centaur SP vehicle with the Spaceplane attached in a typical flight configuration is shown in Figure 6-13. A detailed description of the stage configuration is given in the following subsections.

6.7.1 Centaur SP Tank Configuration

The basic propellant tank arrangement is an LO2 tank and an LH_2 tank, as illustrated in Figure 6-14. The weight effective, pressure stabilized tank configuration has been proven in 463 Atlas flights, 56 Atlas/Centaur flights, and 7 Titan/Centaur flights. This structurally efficient Centaur tank contains the main engine propellants, establishes vehicle primary structural integrity, and supports vehicle systems and components. NASA has determined that the cryogenic Centaur F can be safely integrated into the Space Transportation System. Extensive testing of the Atlas/Centaur tank has demonstrated that the tank has a much greater strength capability than the design values.

The basic tank material is 301 CRES purchased from the mill to an exacting Convair specification. The entire welded tank assembly, including all rings and brackets, is made of 300 series CRES, which minimizes galvanic corrosion. Stress corrosion cracking problems are avoided by the selection of the tank materials and by storage and maintenance activities that provide a periodic WD-40 protective coating.

Tank raw stock is ultrasonically inspected. Tensile, elongation and weld joint fatigue tests are run on all tank raw stock and this procedure will be continued. Major structural member integrity is verified by test coupons from parent material. Tank weld samples are tested before, during, and after the machine welding operation. This existing procedure will be continued. All tank weld joints are 100% radiographically inspected. Leak testing is performed on all tank weld joints. All existing and proven Centaur Reliability and Quality Assurance requirements will continue to be imposed on the tank design and manufacturing.

The LO_2 tank consists of a 120-inch-diameter cylindrical section closed at each end by an ellipsoidal bulkhead. The LH_2 tank will consist of a 170-inch-diamter cylindrical section closed by an ellipsoidal forward bulkhead and a 24-degree conical aft bulkhead that attaches to the LO_2 tank at its forward bulkhead/cylindrical section joint; refer to Figure 6-15.

The 170-inch-diameter for the LH_2 tank cylinder was selected to provide adequate dynamic clearance between the Centaur and the STS Orbiter payload envelope. Structural rings, insulation, and LH_2 vent duct space requirements were accounted for in selecting the tank diameter as shown in Figure 6-16.

The tank volumes will be designed to provide total propellants of 47,300 pounds, including residuals at a burn mixture ratio of 5.0 (LO_2) to 1.0 (LH_2). Tank skin gages will be based on internal pressures resulting from Centaur engine inlet pressure requirements; see Figures 6-17 through 6-21.

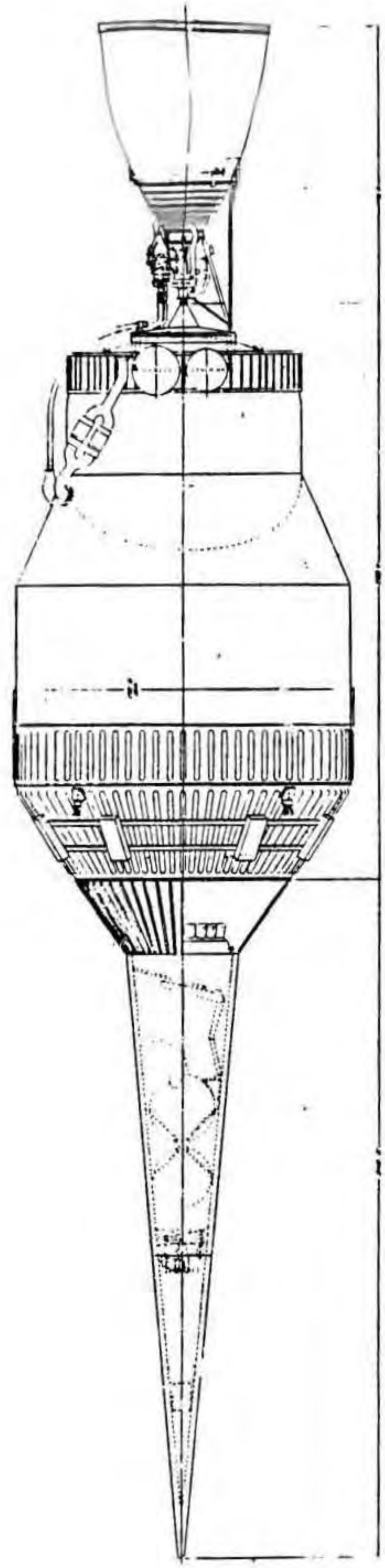
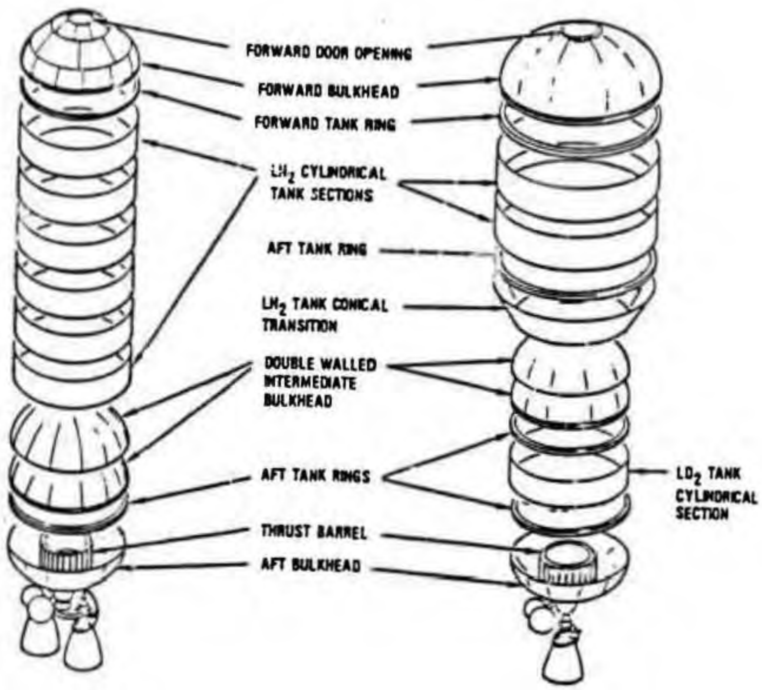


Figure 6-13. CENTAUR SP/SPACEPLANE IN FLIGHT CONFIGURATION



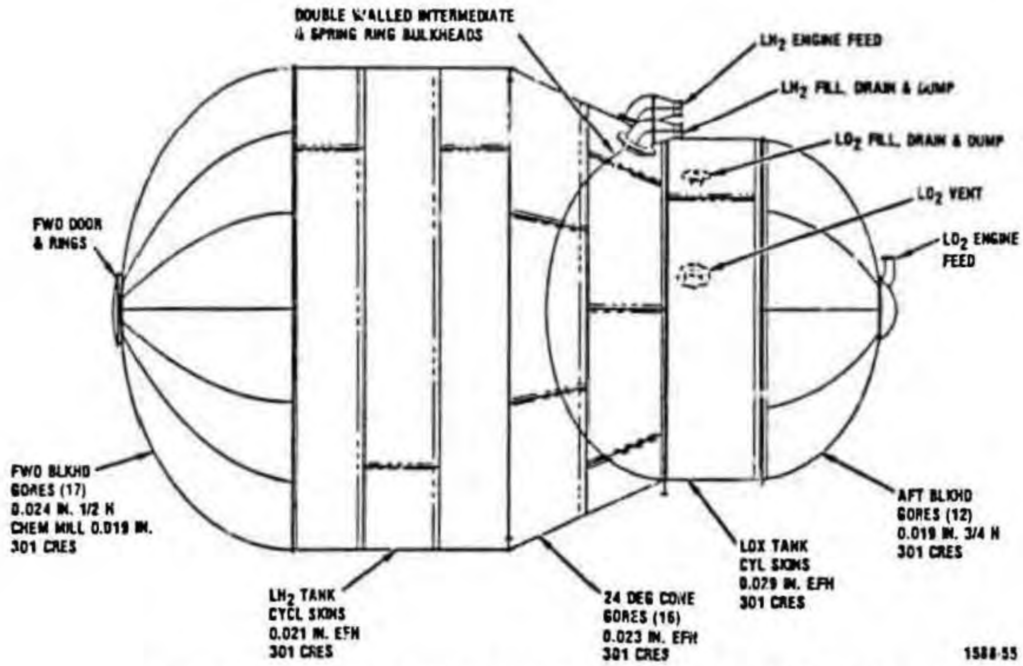
Atlas Centaur

Centaur SP

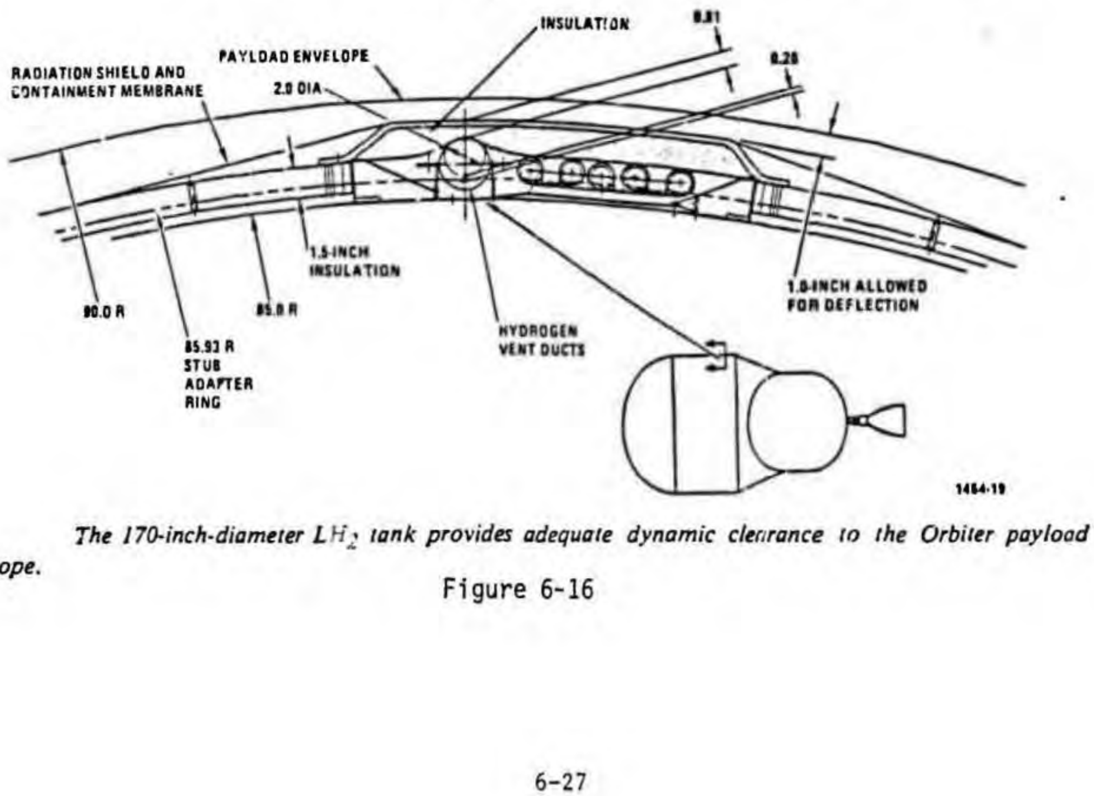
Centaur SP tank is developed from flight-proven Atlas Centaur

Figure 6-14

Figure 6-15



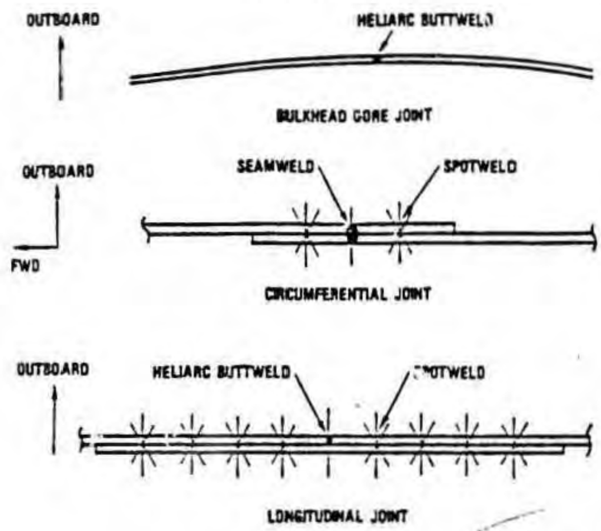
Propellant tank configuration is a direct development of the Atlas/Centaur tank.



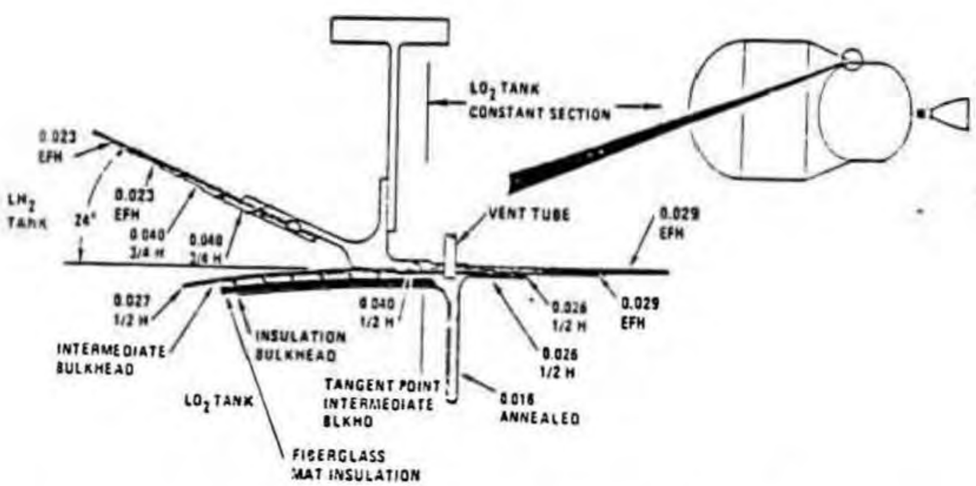
The 170-inch-diameter LH_2 tank provides adequate dynamic clearance to the Orbiter payload envelope.

Figure 6-16

Figure 6-17



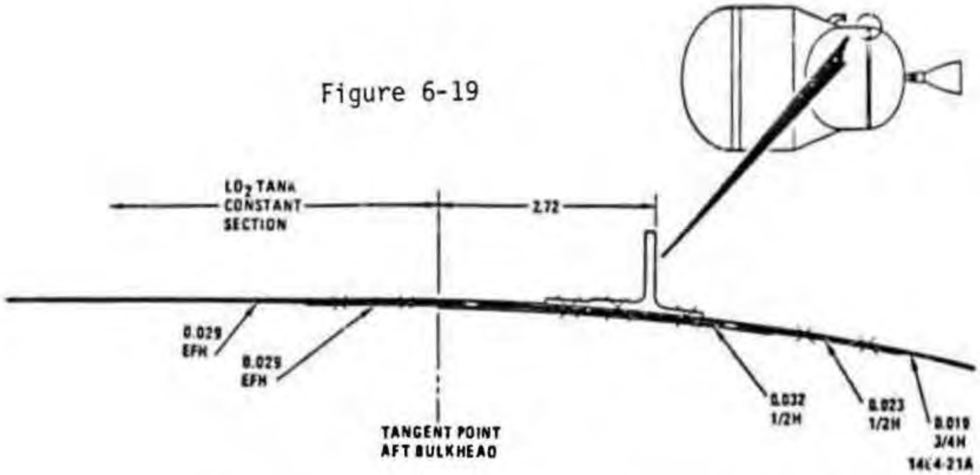
Typical Atlas/Centaur tank joints are also used for Centaur SP tanks



A 24-degree LH_2 cone-to-LO_2 tank transition joint is used for Centaur SP

Figure 6-18

Figure 6-19



LO_2 tank cylinder-bulkhead transition joint used on Centaur

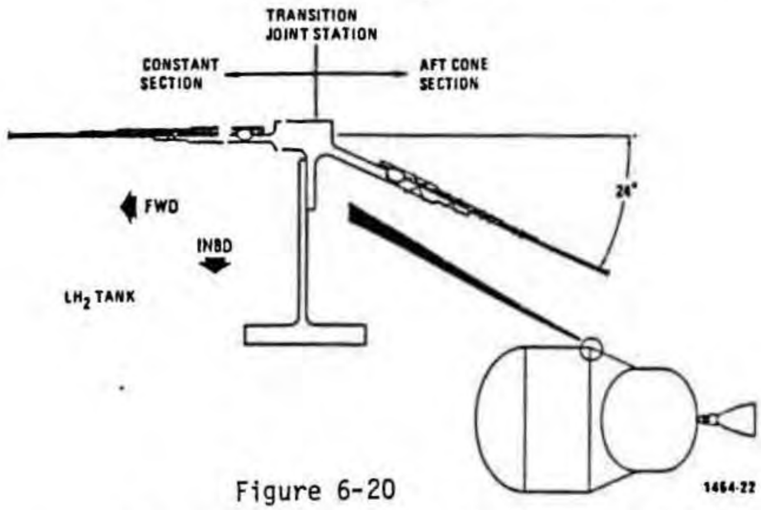
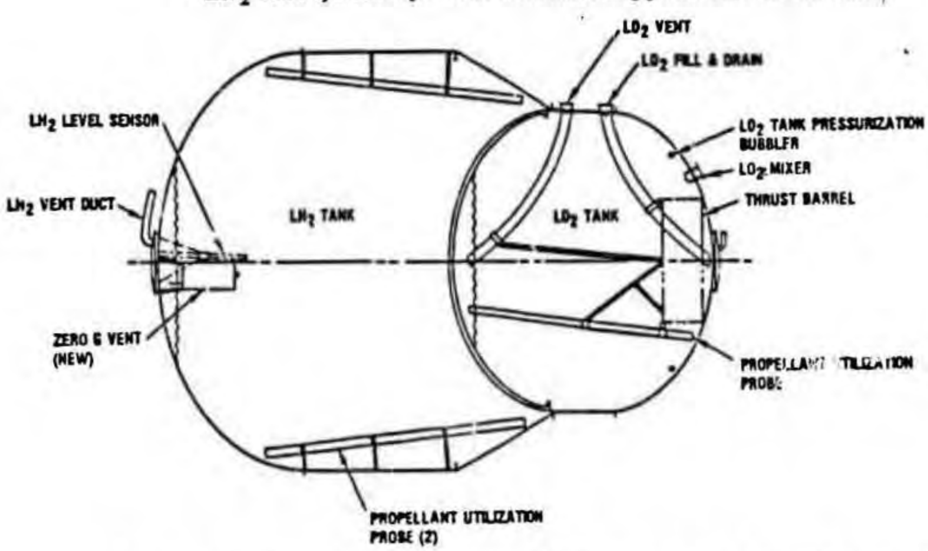


Figure 6-20

LH_2 tank cylinder-aft conic transition ring joint used on Centaur



Centaur SP tank internal installations are adapted from existing Atlas/Centaur designs

Figure 6-21

The LH_2 tank forward bulkhead consists of 17 gore sections buttwelded to form an ellipsoid as shown in Figure 6-17. Except for the buttweld land areas, the gores will be chem-milled to reduce weight. At the forward end of the bulkhead, and access door will bolt to a door ring, which will be welded to the bulkhead. Welded to the inside of the door will be support structure for the pressurizing gas diffuser/dissipator and the zero-g vent system as shown in Figure 6-21.

The LH_2 tank cylindrical section will consist of three sections about 25 inches long and 170 inches in diameter. The three skins will be spotwelded, seamwelded, and stove-piped together using existing production techniques as shown in Figure 6-17. The forward skin will mate with the aft end of the forward bulkhead and will contain the stub adapter support ring. The aft skin will mate to the LH_2 tank aft cone transition as shown in Figure 6-20. Both joints will use existing Atlas/Centaur-type spot/seamweld joints. The cylindrical section will contain propellant utilization probe support brackets welded to the interior surface. Support structure for items such as vent and pressure lines, wiring, and insulation, will be welded to the external surface.

The LH_2 tank aft cone section will consist of sixteen 24-degree conic skin gores buttwelded together and attached to transition rings at each end. The aft transition ring will also attach to the LO_2 tank at the forward end of its cylindrical section. The aft cone LH_2 section inner surface will contain LH_2 propellant feed and LH_2 fill and dump outlets as shown in Figure 6-18. Support structure for various items such as vent and pressure lines, wiring, and insulation, will be welded to the external surface. The LO_2 tank forward structural bulkhead will be identical to the existing Atlas/Centaur. It will consist of gore sections buttwelded together to form an ellipsoid and will attach to the forward end of the LO2 tank cylindrical section. The gore sections will be chem-milled to reduce weight except in areas of the buttwelds. Refer to Figures 6-14 and 6-18. The structural bulkhead will be spot and seamwelded to the aft transition rings, provising the same sealing and structural integrity as in the existing Atlas/Centaur.

The LO_2 tank forward spring ring/insulation bulkhead will be identifical to the existing Atlas/Centaur assembly, except the aft flange of the spring ring will attach to the LO_2 tank cylindrical section rather than the ellipsoidal aft bulkhead contour. The insulation bulkhead will consist of gore sections buttwelded together to form an ellipsoid and will be welded to the formed spring ring. The insulation bulkhead will be spot and seamwelded to the LO_2 tank cylinder skin, providing the same sealing and structural integrity as in the the existing Atlas/Centaur. The cavity between the forward structural bulkhead and insulation bulkhead will contain insulation identical to the existing Atlas/Centaur; see Figures 6-14 and 6-18.

The LO_2 tank cylindrical section will consist of one section about 30 inches long and 120 inches in diameter. The forward end will be welded to the Lh_2 aft cone/LO_2 forward bulkhead/spring ring transition joint as shown in Figure 6-18. The aft end will be welded to the LO_2 tank aft bulkhead. This joint will also contain the aft tank ring on the aft bulkhead near the cylindrical section of the LO_2 tank as shown in Figure 6-19. The ring will be used to attach the aft adapter. The LO_2 tank cylindrical section will contain the internal vent line, terminating in external parallel vent valves mounted to the skin, and the fill/dump siphon duct terminating in parallel fill and drain valves mounted to the skin. Support structure for various items, such as vent and pressure lines, wiring, and insulation, will be welded to the external surface; see Figure 6-21.

The LO2 tank aft bulkhead structure will be similar to that of existing Atlas/Centaur. It will consist of gore sections buttwelded together to form an ellipsoid as shown in Figure 6-15. Support structure will be welded to the bulkhead to support propellant lines, helium and hydrazine bottles, wiring, the engines, electrical boxes, radiation shields, etc. Minor differences in these items will be required from the existing vehicle. Internally, the bulkhead will contain the LO_2 tank pressurzing bubbler ring, LO_2 mixer, and the engine thrust barrel. The LO_2 propellant feed sump will be mounted on the aft end of the bulkhead as shown in Figure 6-21.

The thrust barrel structure inside the LO_2 tank will be identical to the existing Altas/Centaur structure, with the exception of some support structure details. Detail mounting structure will be revised for installing the LO_2 vent standpipe supports, and the new configuration fill and drain duct will be routed through the structure.

The thrust barrel is a 50-inch-diameter cylinder 15.5 inches high of skin-stringer construction. It reacts engine thrust loads and distributes them into the LO_2 tank aft bulkhead. The forward ring and thrust longerons are 2124 aluminum alloy; the skin and stringers are 2023 aluminum alloy. The cylinder attaches mechanically to the aft ring, which in turn is welded to the LO2 tank aft bulkhead.

6.7.2 Centaur SP Adapters

Three new vehicle adapters (equipment module, stub adapter, and aft adapter) will be designed for Centaur SP as shown in Figure 6-22. They are similar to existing Centaur adapters in form and function. Extensive use of finite-element analysis (which has shown a very close correlation with test results) will be made for sizing and designing these adapters.

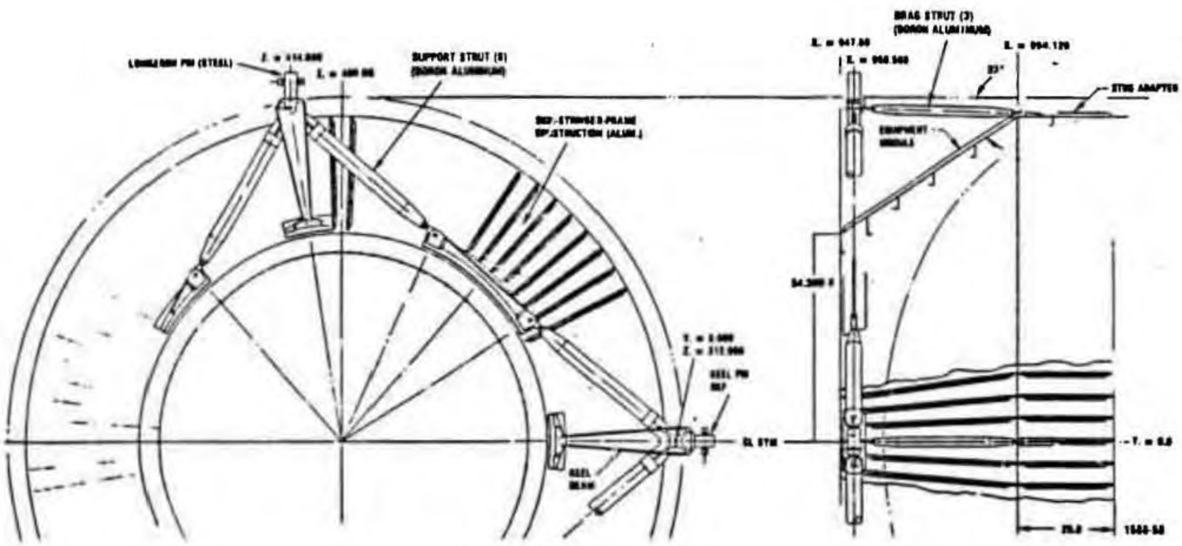
Equipment Module

The equipment module is a 33-degree conical skin-stringer, aluminum alloy structure with a 170-inch-diamter base. It is 47 inches long and 108 inches in diameter at the forward end. Centaur equipment, chiefly avionics, are mounted on the conical surface. The module will include:

* A digital computer unit (DCU) used to convert Spaceplane command signals (e.g., engine start, pitch, yaw and roll) into Centaur SP subsytem command signals.

* An air-conditioning duct for ground cooling of equipment.

* A vent door that will open for Orbiter ascent and close for abort return, as required.



Equipment module and stub adapter are conventional skin-frame construction.

Figure 6-22

In addition, the equipment module includes a beam and truss support system that interfaces with the Orbiter midfuselage payload support system. This support system will provide forward support for Centaur in the Orbiter.

The Spaceplane adapter is approximately 36 inches long, is also a conical skin-stringer structure and will interface with the forward end of the equipment module.

Stub Adapter

The stub adapter is a 170-inch-diameter cylindrical aluminum skin-stringer structure similar to that flown on Atlas/Centaur; it is 25 inches long. The aft end of the adapter attaches to the Centaur LH_2 tank forward ring. The forward end of the adapter attaches to the equipment module.

The forward end of the LH_2 tank sidewall insulation system attaches to the stub adapter. Also, the forward support truss fore and aft drag struts attach to the stub adapter.

Aft Adapter

The aft adapter is a 10-foot-diameter, 11.2 inch-long, aluminum alloy, skin-stringer cylinder structure with attachemtn rings at each end. This adapter distributes CISS support loads int othe Centaur tank and provides and interface for attaching the separation system. The forward ring bolts to the LO_2 tank aft ring and the aft ring attaches to the separation ring. The aft adapter is similar in design to the proven Atlas/ Centaur interstage adapter.

Common design features include attachment ring configuration and bolt pattern, cutout locations, and a stringer spacing of six degrees. The skin is of variable thickness. The adapter cutouts are desinged for routing the LH_2 tank cylindrical section to the LO_2 tank aft bulkhead area, and for routing small tubing. Light, efficient, support structure is mounted on the aft adapter for the vehicle separation springs, fluid disconnect panels, radiation shields, and wiring.

6.7.3 Separation System

The reliable Lockheed Super*Zip pyrotechnic separation system, which has become an industry standard, will separate Centaur from the Orbiter. Lockheed will provide a separation ring containing the Super*Zip system. It is a 10-foot-diameter, 5.50-inch-long, aluminum alloy cylinder section with attachment rings at each end. The separation ring simply bolts to the aft adapter and the CISS deployment adapter.

Super*Zip is a dependable, dual pyrotechnic system. When it fires, a spring system thrusts the Centaur from the CISS deployment adapter. Should the Super*Zip not separate, the Centaur and deployment adapter can safely be lowered back into the payload bay, thus providing a two-failure-tolerant system. A Super*Zip system was used for shroud separation on Titam/Centaur.

6.7.4 Insulation System

The Centaur Sp insulation system is functionally identical to the Atlas/Centaur forward bulkhead system. All materials have been selected to meet STS contamination and saf ty requirements.

Tank Insulation System

This system consists of two major portions (Figure 6-23): the forward bulkhead insulation and the tank sidewall insulation. The forward bulkhead two-layer foam insulation blankets are installed on the hydrogen tank forward bulkhead and enclosed by the cylindrical stub adapter and the conical equipment module. The tank sidewall two-layer foam insulation blankets are attached at the outboard flange of the forward ring of the stub adapter and extend aft along the full length of the hydrogen tank sidewall cylindrical and conical section and are attached to the purge collector plenum.

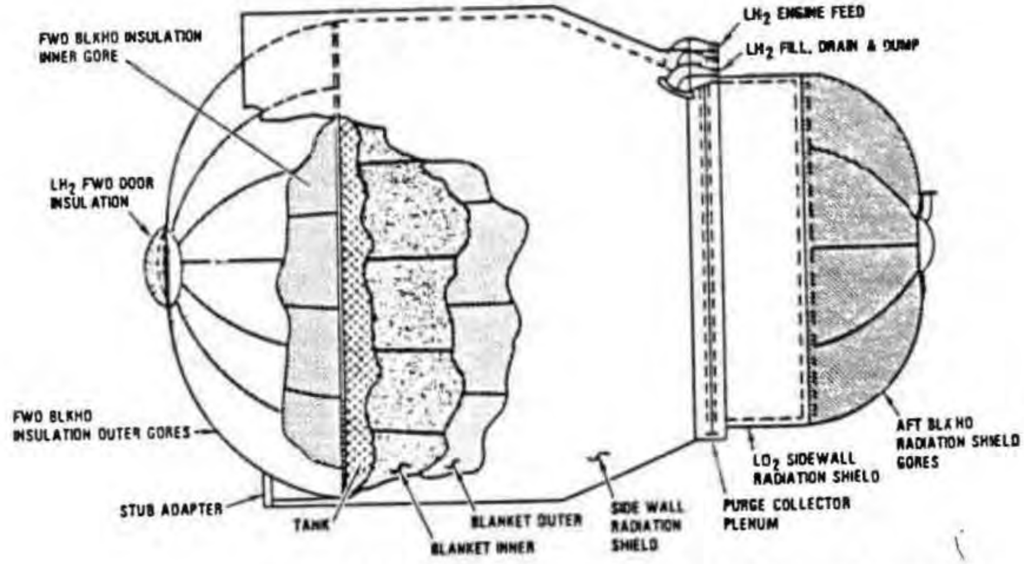
Sidewall insulation blankets are enclosed by a radiation shield which is sealed to the stub adapter retaining channel at the forward ring flange and the purge plenum at the aft end which provides containment of the blanket helium purge gas. Holes are provided in the stub adapter to connect the forward bulkhead blanket compartment to the sidewall blanket compartment. Vent doors are provided in the equipment module and the purge plenum to vent the insulation blanket compartment during ascent. The insulation blanket compartments are purged with helium before tanking to purge the GN_2 from the blanket. During tanking, this helium purge keeps out the payload bay GN_2 purge and maintains a positive ΔP across the radiation shield to provide insulation. In the event of an abort, the purge is activated to preclude moisture and air from entering during the abort descent and after shutdown. The purge gas enters the forward blanket compartment through a purge tube in the equipment module and flows through the forward bulkhead blanket, through the holes in the stub adapter, aft along the sidewall blanket, and exits into the payload bay through two check valves.

Forward Bulkhead Insulation

The LH_2 tank forward bulkhead is protected/shielded, first by a three-layer radiation shield and secondly by a two-layer foam blanket. The three-layer radiation shield has the same properties as the LH_2 sidewall radiation shield, except that the inboard layer shall also be perforated (see Figure 6-24). This shield is held in place by the same pins holding the foam gores in position.

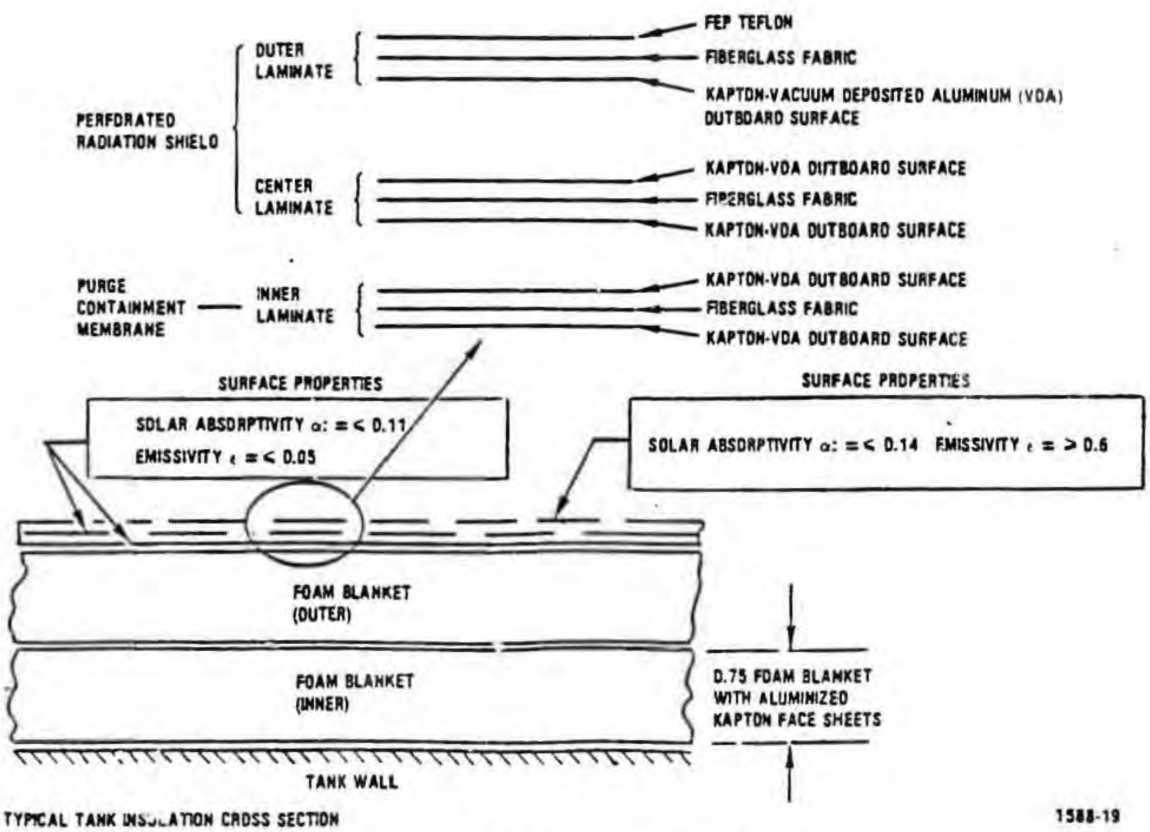
The foam insulation blanets encompassing the bulkhead is enveloped by a single sheet of 1-mil, double-aluminized kapton. The insulation blankets shown in Figure 3-12 are installed as two separate blankets, each 3/4-inch thick and installed one over the other, with the butt joints offset half the width of the gore, providing a total assembly thickness of 1-1/2 inches.

Each blanket is fabricated in 12 separate gores; they are fastened to each other with pin fasteners. A low-level helium purge is employed to maintain a prelaunch helium environment forward of the bulkhead, as shown in



Forward bulkhead insulation design for Centaur tank insulation.

Figure 6-23



Centaur insulation blankets.

Figure 6-24

Figure 6-25, and the effective thermal conductivity of the insulation approaches that of quiescent helium. However, early in-flight evacuation of the liberally vented insulation yields a system of three-layer radiation shield for on-orbit thermal control of the forward bulkhead.

LH_2 Tank Sidewall Insulation

The foam insulation design on the Centaur forward bulkhead is also used on the Centaur SP LH_2 tank sidewall for pre-launch thermal control. As shown in Figure 6-25, a Kapton/glasscloth/Kapton laminate containment membrane maintains internal helium in the blanket during prelaunch operation. Predicted prelaunch heat flux through the LH_2 tank sidewall is approximately 140 Btu/hr-ft^2. Two radiation shields are positioned outboard of the helium-purged blanket. These laminated shields are liberally ventilated to achieve rapid in-flight radiation shielding of the LH_2 tank sidewall. This system of radiation shields has been thoroughly tested and corroborated on Titan/Centaur missions.

Purge and Vent

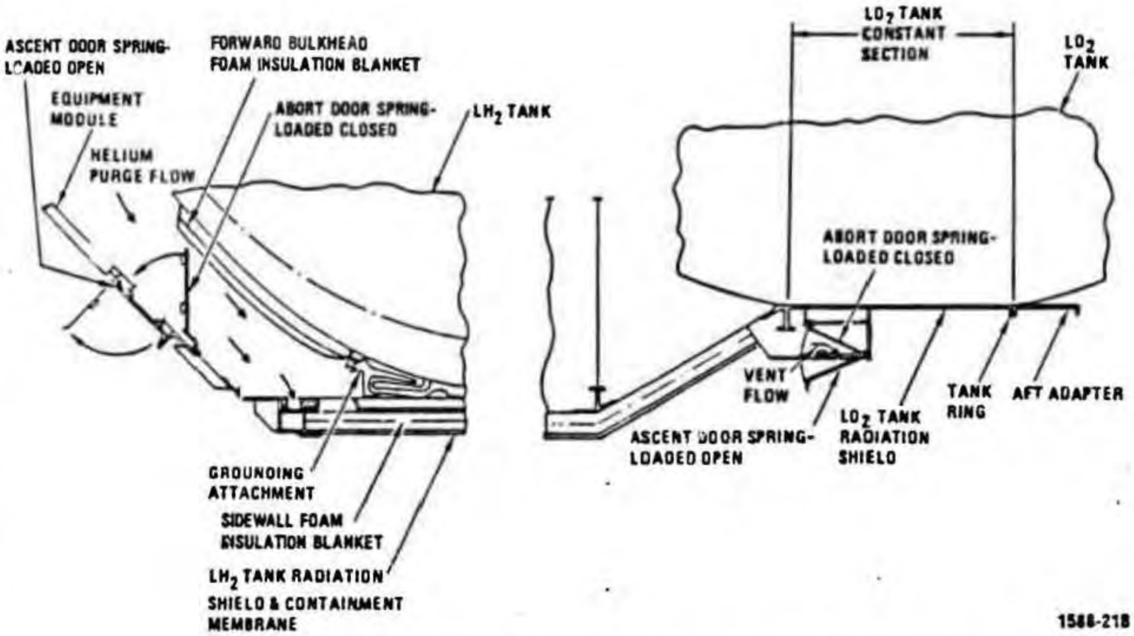
The insulation system is purged with helium gas introduced at ambient temperature and at 20 to 60 pounds per hour at the forward end of the equipment module. A dual-position vent door on the module opens to vent the compartment during ascent (Figure 6-25). Before riseoff, equipment module purge gas flows into the sidewall insulation at the forward end through holes in the stub adapter. The gas flows aft to an annular purge plenum and vents through relief devices and another dual-position vent door for ascent venting. The doors will close to permit blanket repressurization during an abort reentry sequence when the Centaur tank will contain postdump residual propellants. The restart of purge flow during reentry and the foam blanket rigidity prevents liquid air run off.

LH_2/LO_2 Tank Intermediate Bulkhead

The twin-skin vacuum bulkhead separating the two tanks has been employed successfully on all Centaur vehicles. As shown in Figure 6-18, the assembly contains a fiberglass mat insulation which is maintained in compression by the spring ring bulkhead and by LO_2 tank pressure. Cryo-pumping of the intervening volume following LH_2 tanking yields a low conductance of less than 0.045 Btu/hr-ft^2-R. A typical measured heating rate through the intermediate bulkhead is 710 Btu/hr. This corresponds to an effective conductance of only 0.0393 Btu/hr-ft^2-R.

LO_2 Tank Aft Bulkhead Radiation Shield

This shielding system includes an outer, semi-rigid fiberglass shield having an aluminized inner surface and a white (polyvinyl-flouride) outer surface, and twin double-aluminized Kapton inner shields and an inner membrane shield. The system has been employed successfully on numerous Centaur flight vehicles. LO_2 tank aft bulkhead heat flux through the shielding system was measured at 1.0 to 2.0 Btu/hr-ft^2 on TC-5.



Insulation purge and vent systems meet all STS safety requirements.

Figure 6-25

LO_2 Tank Sidewall Radiation Shield

This shielding system is simply an extension of the LH_2 radiation shield assembly and is identical thereto. However, a helium-purged foam blanket is not required for pre-launch insulation of the LO_2 tank sidewall and is thus omitted in this system.

6.7.5 Modifications for Spaceplane Mission Requirements

Several modifications have been identified which improve the usefulness of the Centaur SP over the basic Centaur F, relative to the Spaceplane mission requirements. These modifications are described in the following subsections.

Single Main Engine

The two RL10A-3-3A engines of the Centaur F have been replaced with a single RL10 Derivative II B engine. The higer area ratio nozzle of the Derivative II B produces an increase in specific impulse of approximately 26 seconds. The engine also has low thrust operating levels for small ΔV maneuvers, propellant settl ng, etc. The reduced thrust level (33,000 lb. versus 15,000 lb. vehicle thrust) has no significant effect on vehicle performance. A more detailed description of the Derivative II B engine is given in Section 6.8.

In addition to the removal of one engine, removal of propellant ducting, one of the vehicle tank outlet valves, one set of engine gimbal actuators, propellant settling thrusters (due to the low ΔV capability of the Derivative II B), and other miscellaneous hardware results in a significantly vehicle dry-weight reduction (see Section 6.7.6).

Avionics

The Centaur SP avionics components and weight are reduced from those of the Centaur F because the main computer functions (steering commands, telemetry, etc.) are performed by the Spaceplane itself. A much simplified computer is retained on the stage to interpret Spaceplane commands and to provide miminal control after the Spaceplane is separated. Because of the reduced avionic component requirements and the reduction of main engine power requirements, the vehicle electrical (battery) requirements are reduced, also providing a weight savings.

Basic Structure

Due to weight saved in other vehicle systems, the vehicle tanks can be lengthened slightly to add additional propellant capacity. In this case, a tank weight increase of approximately 28 lbs. allows a propellant addition of slightly over 1800 lbs., yielding a significant increase in vehicle ΔV capability.

Fluid Systems

For vehicle maneuvering purposes, it is likely that station-keeping and axial movement rearward might be better accomplished by using thrusters on the Centaur SP rather than those on the Spaceplane. Therefore, forward-facing hydrazine thrusters were added to the Centaur equipment module at four locations. Since a hydrazine supply is already on the Centaur SP for fine pitch and yaw pointing and for roll control, no significant weight impact was estimated for the addition of these new thrusters.

Airborne Support Equipment (ASE)

The basic Centaur F with its ASE in the shuttle cargo bay envelope is shown in Figure 6-26. Our initial approach was to use essentially the same ASE with only minor modifications for the Centaur SP, as shown in Figure 6-27. The weight of this ASE, however, is over 8600 pounds. Investigation of ASE weights projected in the OTV systems studies (references 8 and 9), as well as earlier studies, indicated that weights of 5000 to 5500 pounds could be achieved primarily through the use of composite materials rather that the more conventional aluminum construction of box beam bulkhead and skins of the Centaur F. A revised ASE for the Spaceplane was then conceptually designed which was estimated to weigh 5500 lb. and is shown in Figure 6-28.

6.7.6 Mass Properties

The estimated mass properties of the Centaur SP are shown in Table 6-III. Centaur F weights are provided for comparison purposes.

6.8 MAIN ENGINE DESCRIPTION

As stated in Section 6.5, the baseline main engine selected for the Centaur SP is RL10 Derivative IIB. This engine is derived from the basic RL10A-3-3 but has increased performance and operating flexibility for shuttle cryogenic upper stage (e.g., OTV). The Derivative IIB described here was specifically optimized for use with the Centaur SP/Spaceplane (e.g., norminal mixture ration is 5:1, engine installed length is 70 in.) and is summarized in Figure 6-29.

6.8.1 Definition and Requirements

The Derivative IIB engine is defined as an RL10A-3-3 with the following changes:

1 Two-position nozzle with recontoured primary section to give a large increase in specific impulse with engine-installed length no greater the RL10A-3-3 (70 in.)

2. Tank head idle (THI) capabilities, where the engine is run pressure fed without its turbopump rotating on propellants supplied from the vehicle tanks at saturation pressures.

Table 6-III

CENTAUR SP MASS PROPERTIES

	Centaur-SP	Centaur-F	Δ
Body/structure, lb	2,960	2,932	+28
Propulsion group	713	1,213	-500
Flight control	60	310	-250
Electrical	166	266	-100
Fluid systems	685	630	+55
Info. and safety	265	265	0
Aft sep. system	160	160	0
Dry weight, lb	5,009	5,776	-767
Residuals	609	609	0
Spaceplane adaptor	120	120	0
Burn-out weight	5,738	6,505	-767
Expendables			
Main propellants	47,262	45,450	+1812
He, N_2H_4	700	253	+447
Stage ignition weight, lb	53,700	52,208	+1492

SHUTTLE/CENTAUR F PROFILE

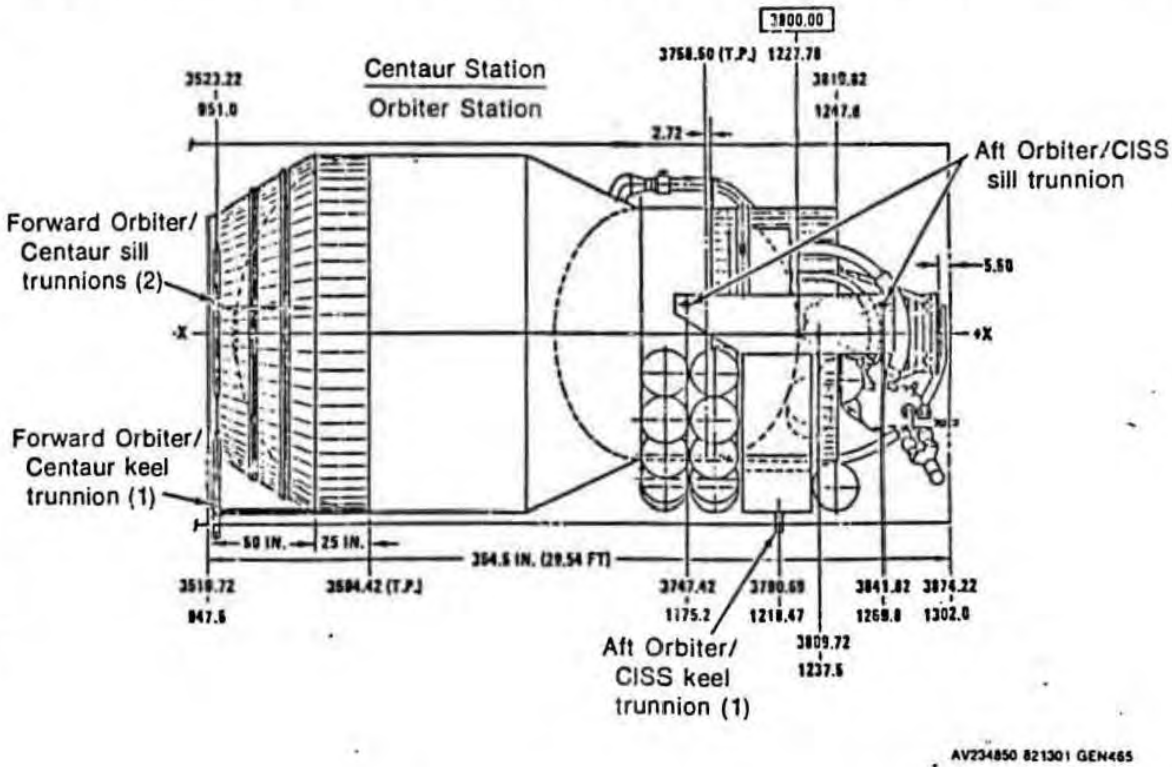


Figure 6-26

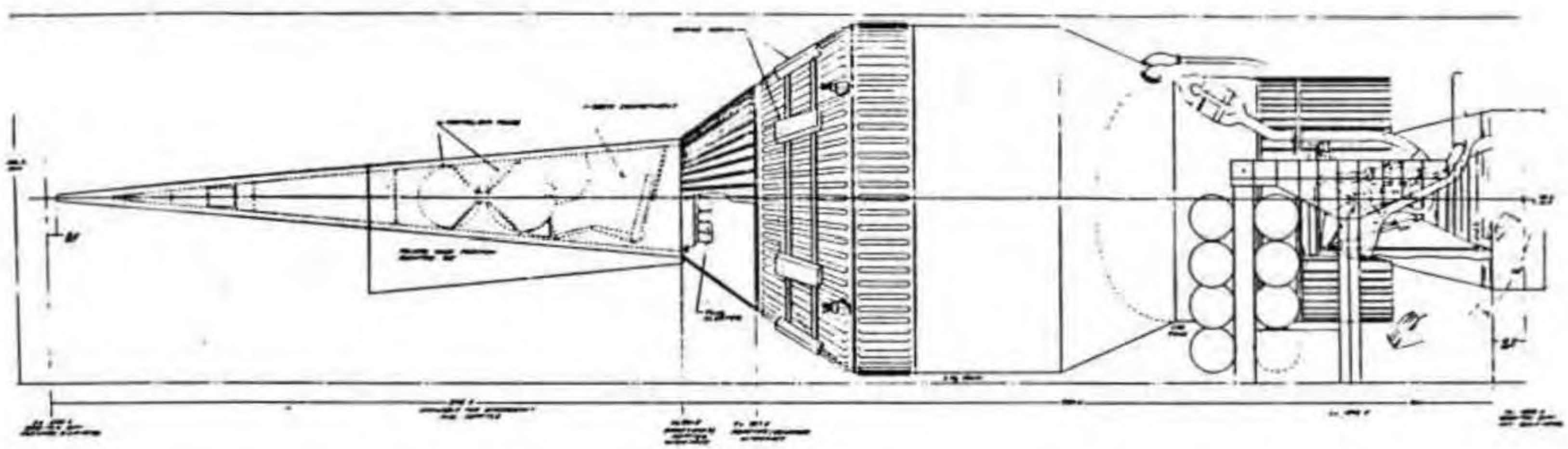


Figure 6-27 CENTAUR SP WITH CENTAUR F TYPE ASE

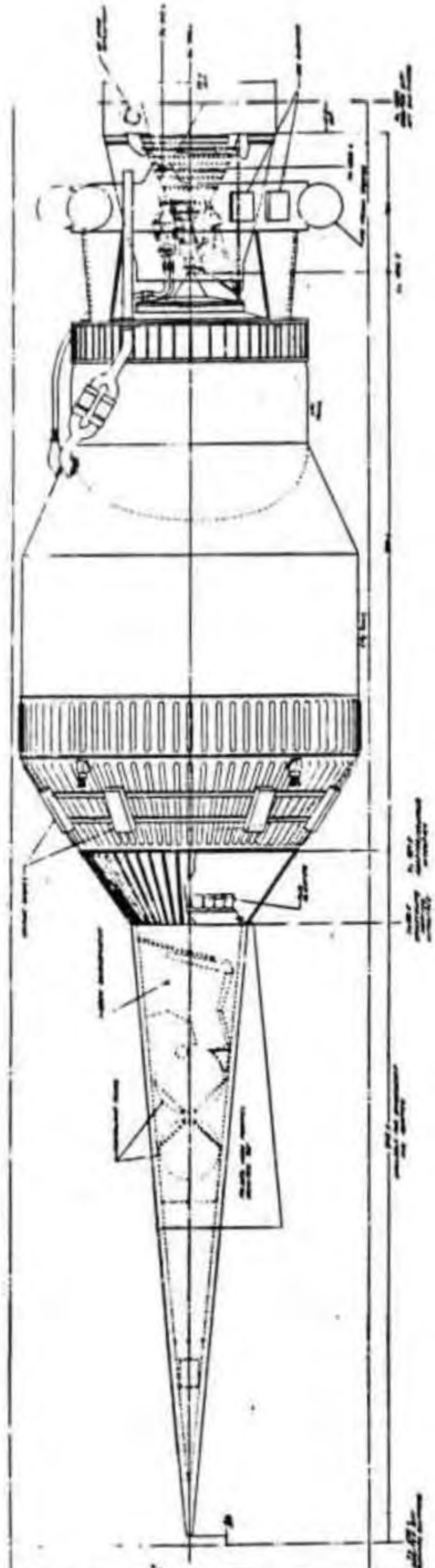


Figure 6-28 CENTAUR SP WITH OPTIMIZED ASE

Propellant conditions at the engine inlets can vary from superheated vapor, through mixed phase, to liquid. The objectives are to supply low thrust to settle vehicle propellants and also to obtain useful impulse from the propellants used to condition the engine and vehicle feed system.

3. Pumped idle mode, with saturated propellants in vehicle tanks, and bootstrap autogenous pressurization. This mode of operation allows the RL10A-3-3 Bill-of-Material turbopump to be run at a sufficiently low speed where prepressurization subcooling of the propellants at the pump inlets is not required. By using the engine's bootstrap autogenous pressurization capabilitiy, the tanks can then be prepressurized to satisfy the engine's full thrust pump inlet net positive suction head (NPSH) requirements before accelaration to full thrust.

4. Capability for both H_2 and O_2 autogenous pressurization, which may be required on very long burn missions in order to avoid excessively low propellant vapor pressure.

6.8.2 Description

The general arrangement of the RL10 Derivative IIB engine is shown by the installation drawings in Figures 3-30 and 3-31. This engine is interchangeable with the RL10A-3-3.

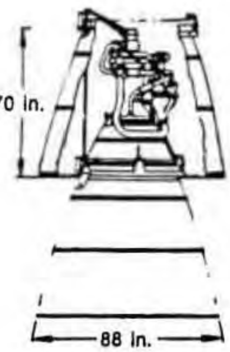
The principal components of this engine are shown in Figure 6-32. The fuel pump interstage chilldown valve is deleted, since the engine is conditioned by running in THI mode. A GO_2 heat exchanger, GO_2 control valve, and turbine bypass valve are added to enable the engine to run in THI. Fuel and oxidizer tank pressurization valves are added to give autogenous pressurization capability. Additional solenoid valves and modifications to the oxidizer flow control valve and thrust control valve give the engine its capability to operate in three modes. A dual exciter gives improved ignition reliability in THI. The primary nozzle is recontoured and a jackscrew-operated, two-position, radiation-cooled, extendible nozzle is added. The engine maintains the same design margins as the RL10A-3-3 engine since the chamber pressure level remains unchanged and the turbopumps are basically unchanged.

The dry weight of the engine and its subassemblies are summarized in Table 6-IV. Of the total engine weight of 415 lb, 42% is the weight of existing hardware, 40% is calculated from layout drawings, and 18% is estimated.

6.8.3 Operation

The engine is started in THI mode, with propellants supplied in vapor, mixed, or liquid phases.

With the inlet shutoff valves open, fuel flows through the pump, the thrust chamber cooling jacket, around the turbine, through the GO_2 heat exchanger, and in to the main injector. Similarly, the oxidizer flows the heat



Thrust	:	15,000 lb
Chamber pressure	:	400 psia
Area ratio	:	315
Specific impulse	:	472 sec at 5.0 MR
Operation	:	Full thrust (low NPSH)
	:	Pumped idle (~ 3700 lb thrust) (saturated propellants)
Conditioning	:	Tank head idle (~ 150 lb thrust)
Weight	:	415 lb
Life (TBO)	:	190 firing/5 hr

Meets system operational and schedule requirements

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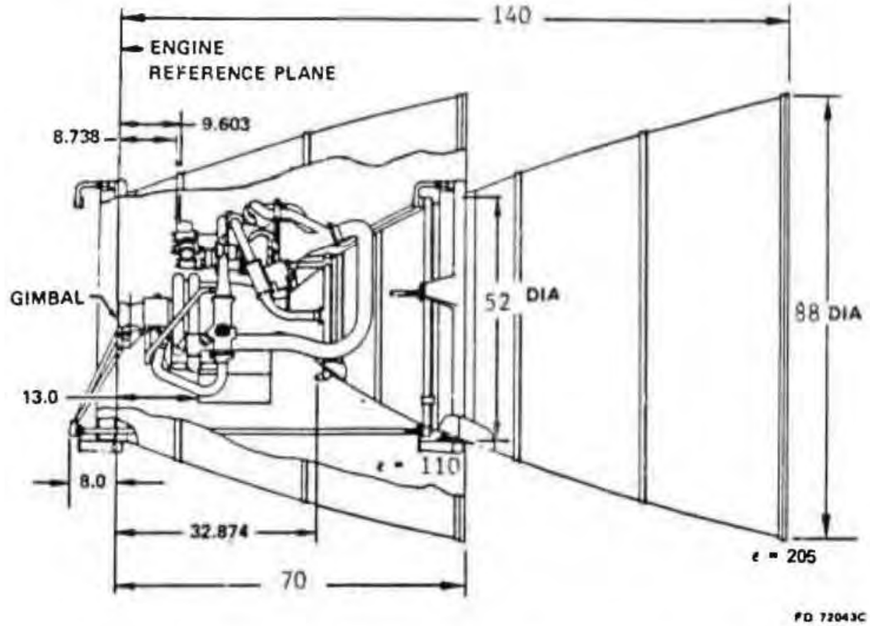
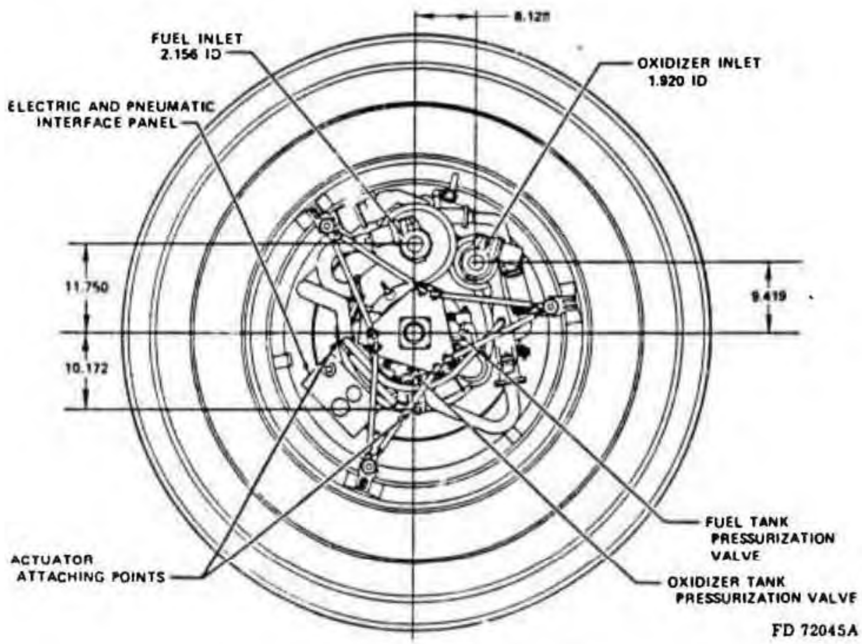
RL10 Derivative IIB Optimized for Centaur SP

Figure 6-29



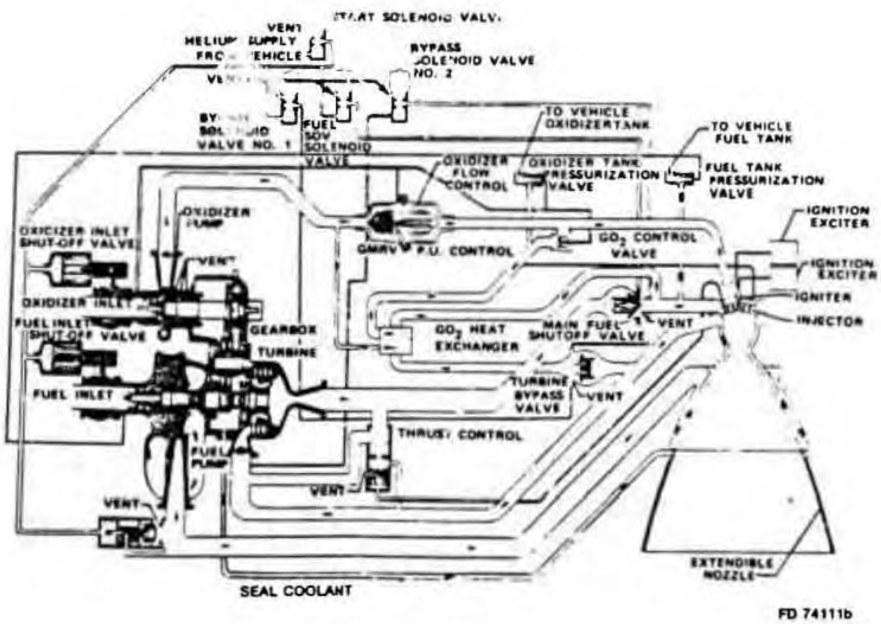
Derivative IIB Engine Installation Drawing (Sheet 1)

Figure 6-30



Derivative IIB Engine Installation Drawing (Sheet 2)

Figure 6-31



RL10 Derivative IIB Propellant Flow Schematic

Figure 6-32

exchanger to the injector. The operating conditions shown on Figure 6-33 are for a thermally conditioned engine with liquid propellants supplied at 16 psia.

After pump conditioning has been completed in THI mode, the engine is ready to be rought to its maneuver thrust level for low ΔV maneuvers or as a step on its acceleration to full thrust. To start the turbopumps, the main fuel shutoff valve is opened, and the turbine bypass valve is closed momentarily to give a high initial turbine torque and is then reopened to the maneuver thrust position. A wide range of prepressurization flowrates can be supplied in pumped idle mode with little change in engine thrust, although active control valves are not used.

Prior to acceleration to full thrust, propellants with positive NPSH have to be supplied to the engine, either by using the engine's autogenous pressurization system or with some vehicle-supplied system, i.e., boost pumps, helium pressurization, etc.

By closing the turbine bypass valve, the engine is accelerated to full thrust. At about 90% of full thrust, the thrust control valve opens to reduce thrust overshoot. Operation of the engine in full thrust and 6.0 mixture ratio is shown in Figure 6-34.

6.8.4 Performance

The steady-state performance characteristics of the Spaceplane optimized RL10 Derivative IIb engine are summarized in Table 6-V.

6.9 Performance

Performance capability of the external propulsion system is obviously a function of the Spaceplane weight. In this section, a Spaceplane wet-weight of 5800 lb. is assumed, with no losses prior to separation of the Centaur SP. (In most missions, however, some weight change would occur, due to consumption of reaction control gas, water loss, etc.)

6.9.1 <u>Maximum ΔV Mission</u>

On a maximum ΔV mission, the following table illustrates the Centaur SP capability.

Shuttle capability	65,000 lb.
ASE weight	5,500 lb.
Maximum ignition weight	59,500 lb.
Spaceplane weight	5,800 lb.
Maximum Centaur SP weight	53,700 lb.
(system weight @ Centaur SP burnout	11,538 lb.

Table 6-IV

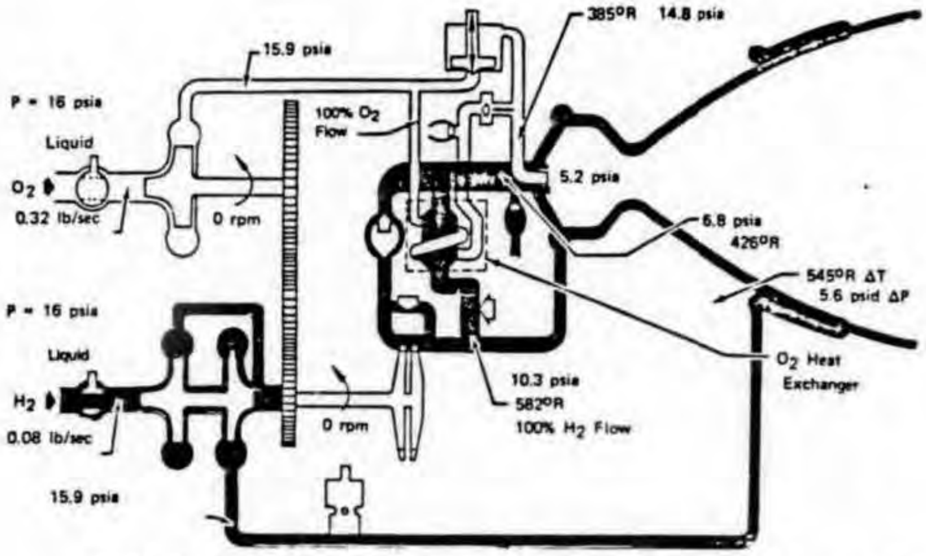
RL10 DERIVATIVE IIB ENGINE
WEIGHT (Centaur SP optimized)

Turbopump and Gearbox	79 lb
Thrust Chamber and Primary Nozzle	110 lb
Extendible Nozzle Actuator System	48 lb
Extendible Nozzle	40 lb
GO_2 Heat Exchanger	13 lb
Controls, Valves & Actuators	66 lb
Plumbing and Miscellaneous Hardware	44 lb
Ignition System	15 lb
TOTAL DRY WEIGHT	415 lb

Table 6-V

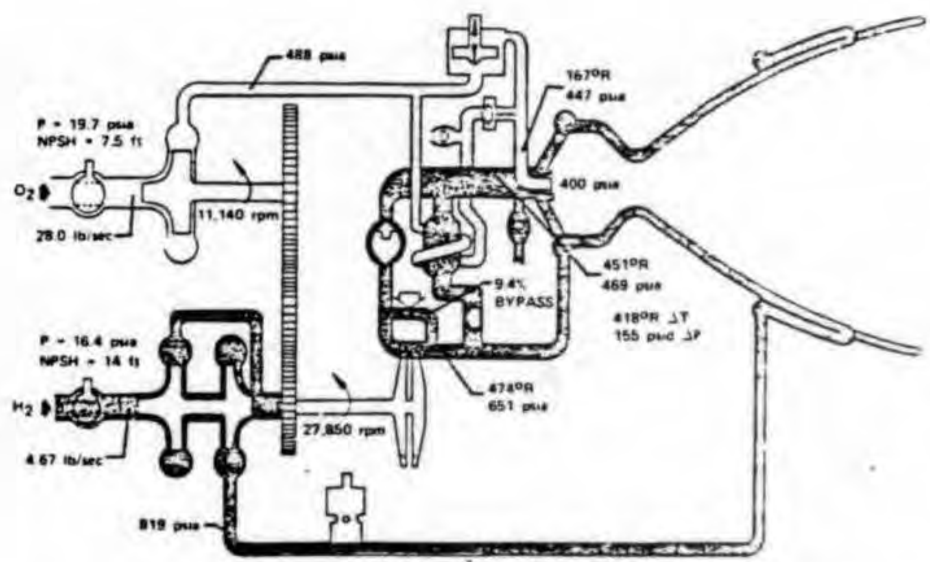
Performance Characteristics of RL10 Derivative IIB
Engine (Spaceplane Mission Optimized)

Operating Mode	Tank Head Idle		Maneuver Thrust	Full Thrust
Thrust, lb	170		3,750	15,000
Mixture Ratio	4.0		5.0	5.0
Chamber Pressure, psia	5.2		100	400
Specific Impulse, sec	438		461	472
Fuel Turbopump Speed, rpm	0		13,370	29,650
Pump Inlet Condition Limits				
Fuel	>16 psia	Superheated Mixed phase	10 psia 65% Vapor	>14 ft NPSH
Oxidizer	>16 psia	or Liquid	10 psia 45% Vapor	>7.5 ft NPSH
Fuel System Pressurization Supply	N/A		0 to 0.15 lb/sec 550° to 520°R	0 to 0.1 lb/sec 440°R
Oxidizer System Pressurization Supply	N/A		0 to 1 lb/sec 570° to 230°R	0 to.5 lb/sec 450° to 225°R



RL10 Derivative IIB Propellant Flow Schematic - Tank Head Idle Mode

Figure 6-33



RL10 Derivative IIB Propellant Flow Schematic - Full Thrust (MR = 6.0)

Figure 6-34

The Maximum system ΔV capability, using these weights, is approximately 24,930 ft./sec. A geosynchronous (circular) transfer from low earth-orbit (LEO) requires approximately 14,000 ft./sec. The return transfer orbit requires approximately 4,700 to 6,000 ft/sec., depending on whether a 28 1/2 degree plane change is accomplished, bringing the total to approximately 18,700 to 20,000 ft./sec. A circularization at LEO requires an additional ΔV of approximately 8,000 ft./sec., which is some 1,800 to 3,100 ft./sec. beyond the capability of the Centaur SP.

A geosynchronous mission should still be achievable, however, by using the Spaceplane's internal ΔV ($\sim$ 2500 ft./sec.) and synergistic plane change/reentry capability. Alternative approaches to obtaining additional Centaur SP ΔV capability are discussed in Section 6.10 (System Options).

If the Spaceplane mission weight were reduced below 5,800 lb., additional performance would also be achieved, as shown in Figure 6-35. This curve shows that if the Spaceplane mission weight were reduced by 800 lb., the current Centaur SP (5,009 lb. dry-weight) could provide an additional ~850 ft./sec. If the 800 lb. could be used for extra propellant (plus a small tankage increase), a total ΔV of 26,000 ft./sec. could be achieved. It is, therefore, significant to achieve a minimum Spaceplane weight.

6.9.2 Large Payload Capability

A mission has been proposed for the Spaceplane system which would start from a Shuttle launch to a 308 n. mi. orbit, at a 28 1/2° inclination, perform a plane change to an equatorial orbit (still at 308 n. mi.) release a series of payloads (estimated weight of 3,140 lb. plus a 150 lb. adapter), and return to the vicinity of Kennedy Space nter. A system weight breakdown is as follows:

Shuttle capability	65,000 lb.
ASE weight	5,500 lb.
Maximum ignition weight	59,500 lb.
Spaceplane weight	5,800 lb.
Payload/adaptor weight	3,290 lb.
Maximum Centaur SP weight	50,410 lb.
Main Propellants	43,972 (tanks off-loaded)

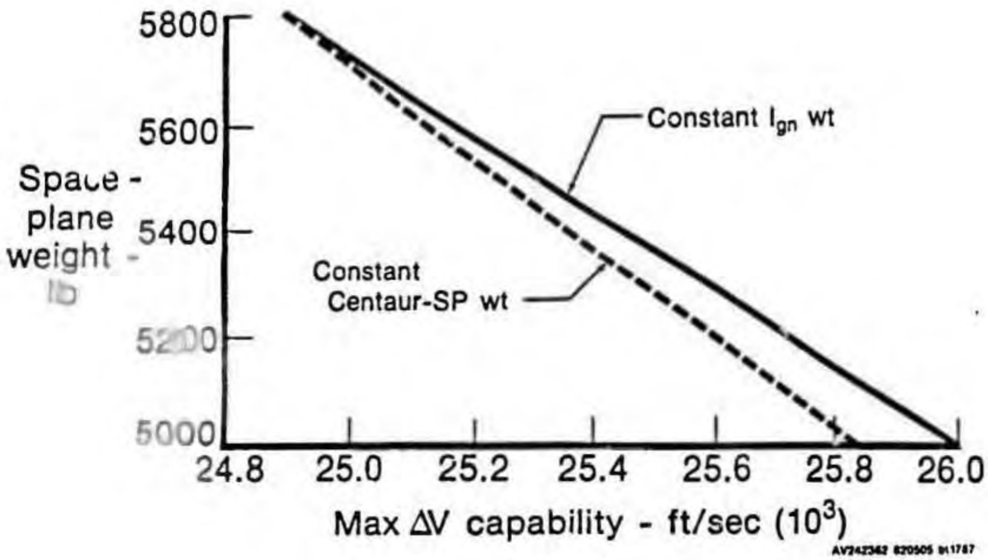
required ΔV: 11,800 ft./sec. (see Figure 6-25)

Weight at mission orbit:

$$11,800 = (472)\ (32.2)\ \mathrm{Ln}$$

$$M_f = 27,374 \text{ lb.}$$

If it is assumed that approximately 75% of the non-propellant expendables (i.e., N_2H_4, He) were used in achieving the mission orbit and in station-keeping, etc., 175 lb. remain, making a Centaur SP start-burn weight of 23,559 lb. for the return trip since the payload (and adaptor) have been released.



System Capability as a Function of Spaceplane Weight

Figure 6-35

System burn-out weight, in this case, is the same as for the high V mission and, therefore, the ΔV capability is:

$$\Delta V = (472)\ (32.2)\ \text{Ln}\ \left(\frac{23,559}{11,538}\right) - 10,850 \text{ ft./sec.}$$

This ΔV capability is equivalent to a plane change of ~25° at 308 n. mi. but can be used in combination with the Spaceplane's internal ΔV capability and synergistic plane change/reentry capability to easily accomplish the defined mission.

6.10 SYSTEM OPTIONS

During the course of the external propulsion system study, several options to the baseline Centaur SP system were found. Each of these options appear to promise improvements in either system ΔV capability or in system flexibility. In the following section, system options, which appear to be within an estimated two year (or less) technology capability, are discussed.

6.10.1 Aerodynamic Maneuvering

On Spaceplane missions to high orbit (e.g, GEO), the Centaur SP by itself, may not have a sufficient ΔV capability to achieve a circular orbit at LEO for Shuttle rendevous or return to the launch site. Studies over the past several years have shown that if a spacecraft is allowed to dip into the earth's atmosphere, the resulting drag can substantially reduce propulsion ΔV requirements to achieve a circular low earth orbit. The primary problems, of course, are surviving severe thermal environment and maintaining satisfactory vehicle control within acceptable system weight limits. The recent OTV systems studies by General Dynamics and Boeing (references 8 and 9) both proposed aerodynamic braking systems. The following discussion is based on the General Dynamic's approach, called the "Lifting Brake."

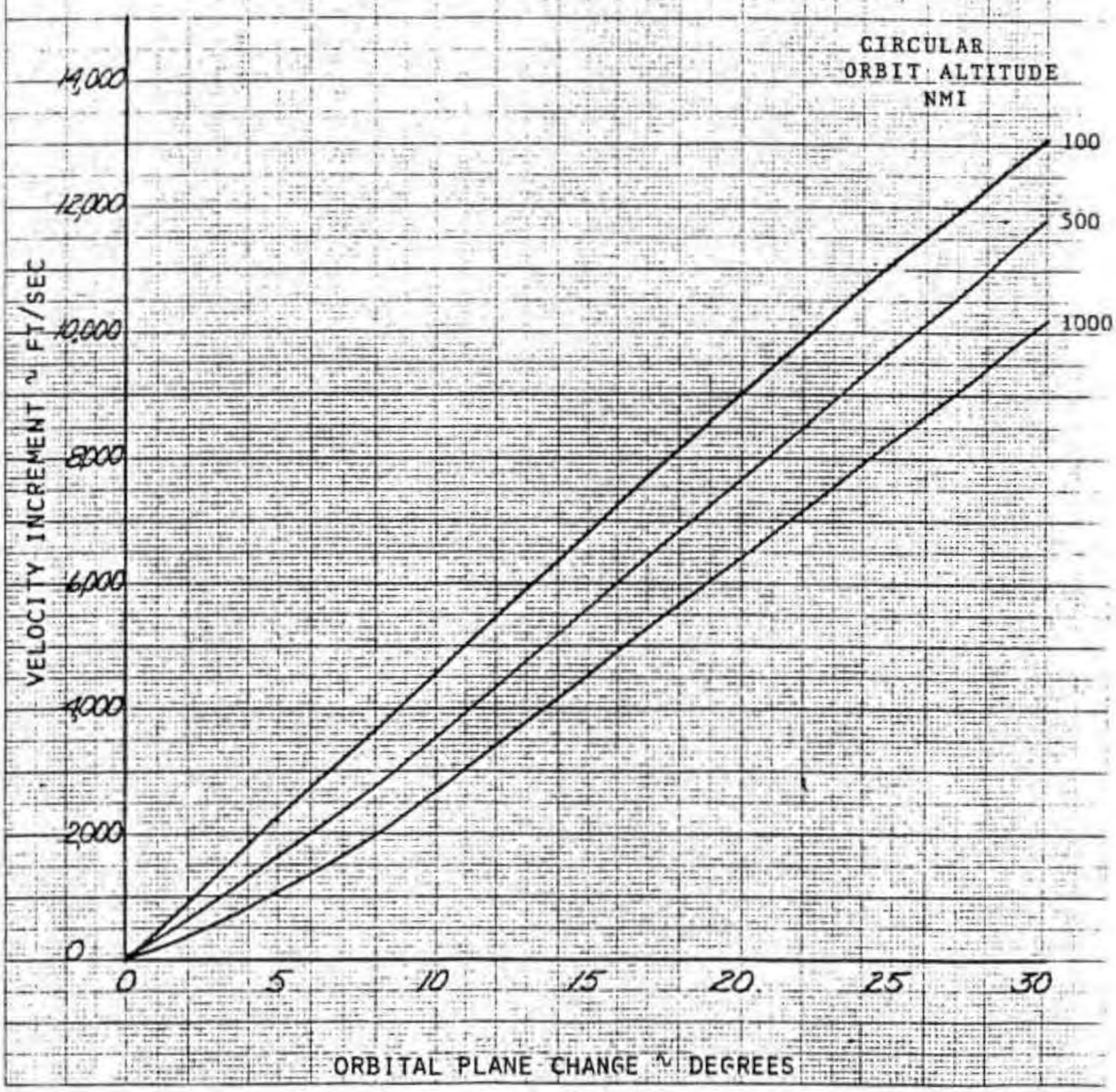
The general characteristics of a lifting brake return from GEO mission is illustrated in Figure 6-37. The characteristics of a lifting brake designed for an OTV are shown in Figure 6-38 and are based on the use of Nextel 312 cloth covering graphite/polyimide ribs. A detailed weight breakdown of the aerobrake, designed for a 30,000 lb. gross weight OTV, is given in Figure 6-39 and shows a total weight of 3,300 lb. including a 15% contingency.

Size of the brake is a function of peak dynamic pressure and vehicle gross weight. For an OTV with a gross weight of approximately 30,000 lb., flying a trajectory with a maximum dynamic pressure of 23 psf, a brake diameter of 50 ft. is required. A smaller vehicle flying to the same maximum dynamic pressure would utilize a smaller brake, as shown in Figure 6-40.

For a typical OTV trajectory analyzed by GDC, the minimum altitude was approximately 260,000 ft. during the braking maneuver and the relative velocity was reduced by approximately 7,500 ft./sec (Figure 6-41).

ORBITAL PLANE CHANGE ΔV REQUIREMENTS

Figure 6-36



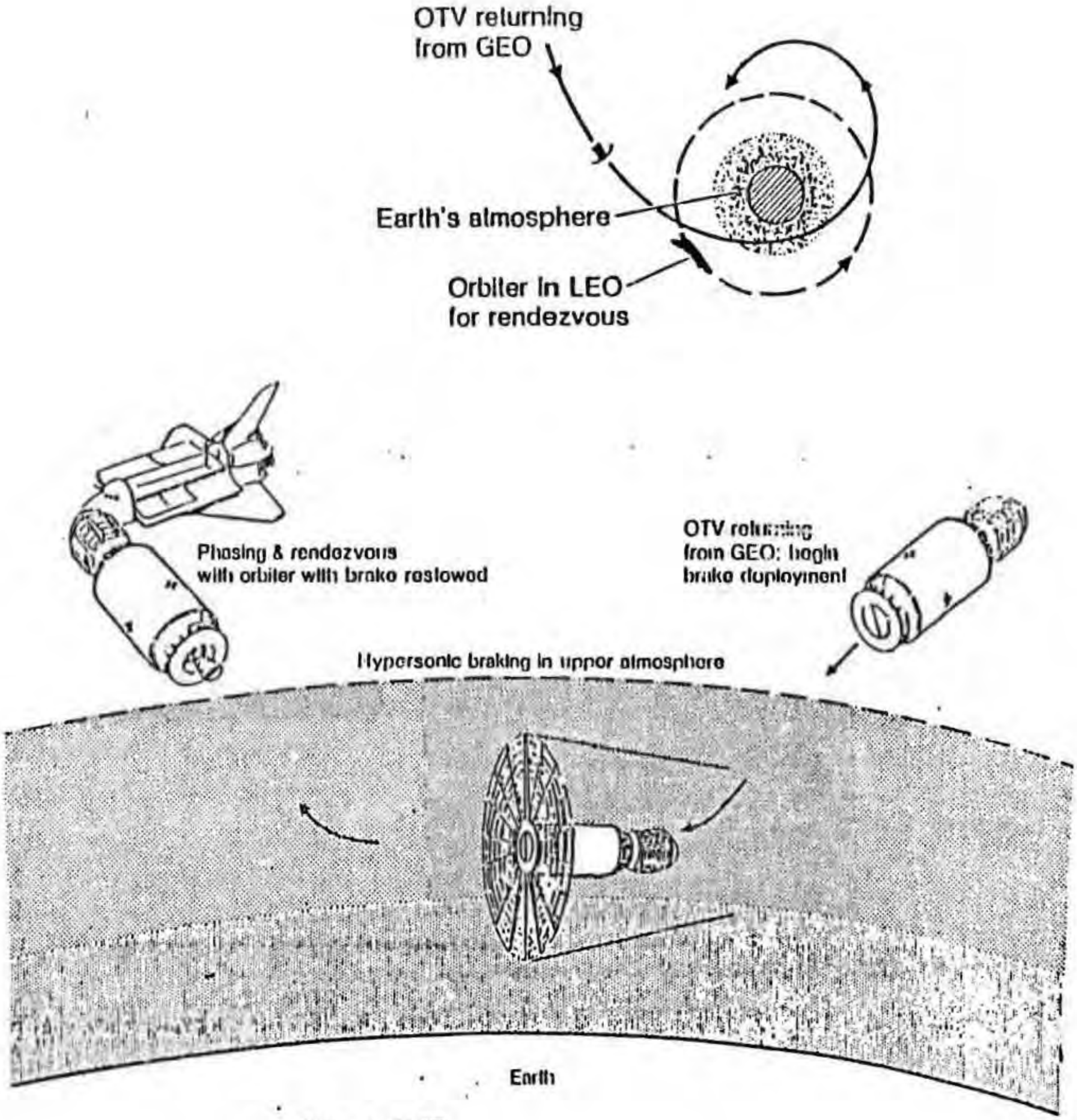
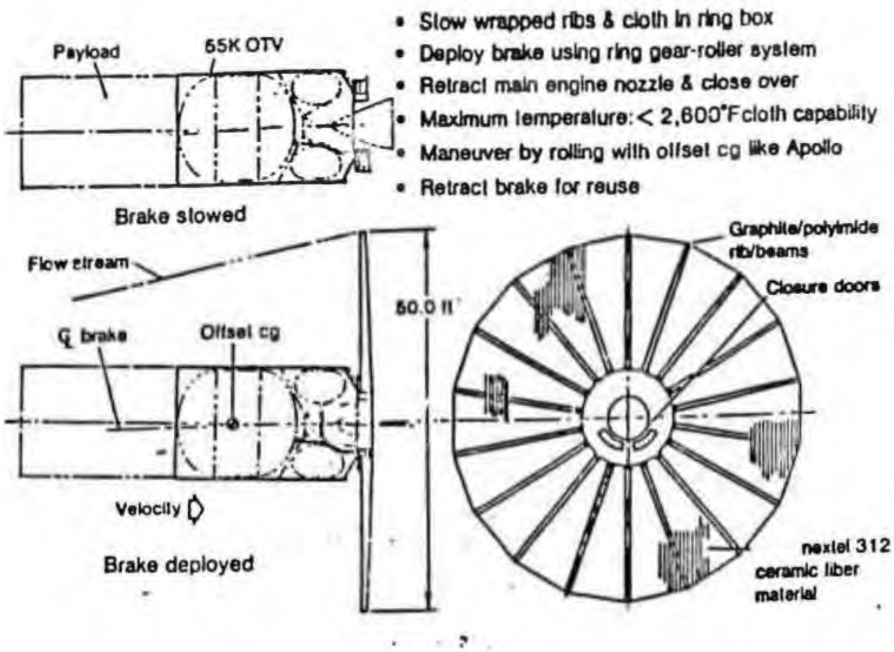


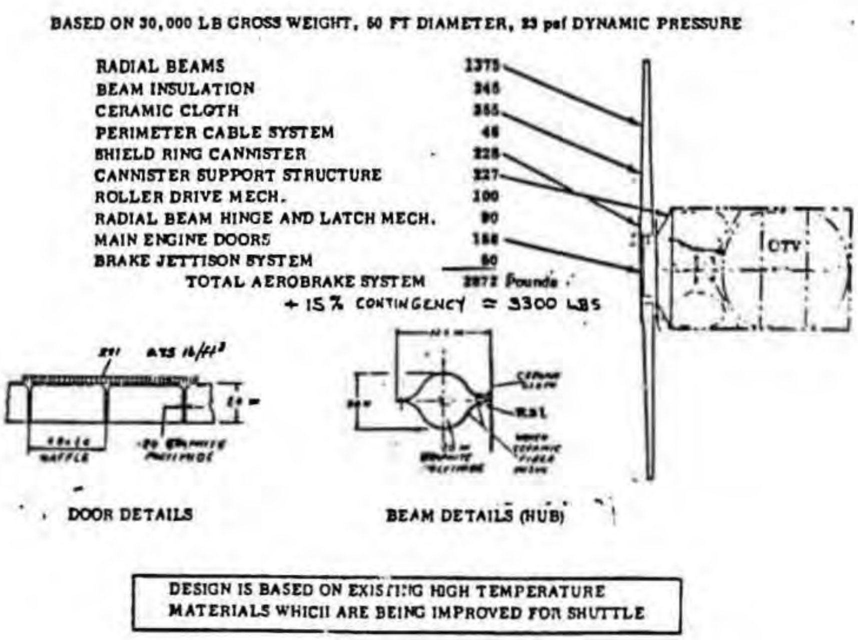
Figure 6-37

OTV Return From GEO



Lifting Brake Characteristics (OTV Design)

Figure 6-38



Lifting Brake System Weights (OTV Design)

Figure 6-39

The analysis also indicated that to rendezvous with the Shuttle, by using a minimum propulsive ΔV, two atmospheric passes would usually be preferred to a single pass, although several additional hours would be required for the mission.

The use of an aerodynamic braking trajectory for a Centaur SP Spaceplane LEO-GEO-LEO mission can provide a significant performance (payload) advantage over a non-augmented trajectory. An example of this performance is shown in Figure 6-42 and shows that over 1,100 lb. of additional payload capability would be available.

The basic problem of aerodynamic heating of the brake appears to be well within the capability of the Nextel 312 ceramic fiber cloth (typical maximum calculated temperature ~2100°F cloth capability). A vehicle such as the Centaur SP, with the Spaceplane attached, should not encounter significant heating since the base flow will not close for 1.5 to 2.5 times the brake diameter downstream (e.g, with a 40 ft. diameter brake, a vehicle would have to be longer than ~60 ft. to experience significant heating).

During the atmospheric braking maneuver, vehicle acceleration would be compared against expected values and changes in steering (L/D trim) would be made to control the characteristics of the final orbit. This requires the best possible local atmospheric density information prior to entry and a responsive guidance, navigation, and control system.

A technology effort should be undertaken prior to a commitment to the use of an aerodynamic braking system. An assessment of this technology by General Dynamics is shown in Figure 6-43.

6.10.2 SHUTTLE PROPELLANT SCAVENGING

As a part of a study for NASA-JSC on the Space Operations Center (Ref. 22), Rockwell International investigated the recovery of residual propellants from the External Tank for later use on orbit (e.g., in an OTV). From this investigation, it was determined that for a full-capacity shuttle launch (65,000 lb. payload), the low level 3σ residuals available would be 2,728 pounds and the nominal residual would be slightly over 9,300 lb. at a mixture ratio of ~2:1. The Rockwell analysis showed that the transfer of such propellants into a receiver tank could be accomplished within the basic shuttle mission requirements, including a safe impact of the E.T., even though jettisoned some 20 minutes later than normal.

For the Centaur SP vehicle, if approximately half of the available propellants could be recovered (1,375 to 4,650 lb.), an additional capability on the order of 330 to 1,100 ft./sec. could be achieved. Further analysis of such a concept should be pursued to determine if the additional V capability is justified. The next section of this report is also centered on the same basic concept.

6.10.3 DRY LAUNCH OF CENTAUR SP

In the baseline configuration, the Centaur SP/Spaceplane weight is limited by the 65,000 lb. Shuttle lift capacity. Of this total, over 47,000 lb. (73%) are the cryogenic main propellants. The remaining 18,000 lb. consist

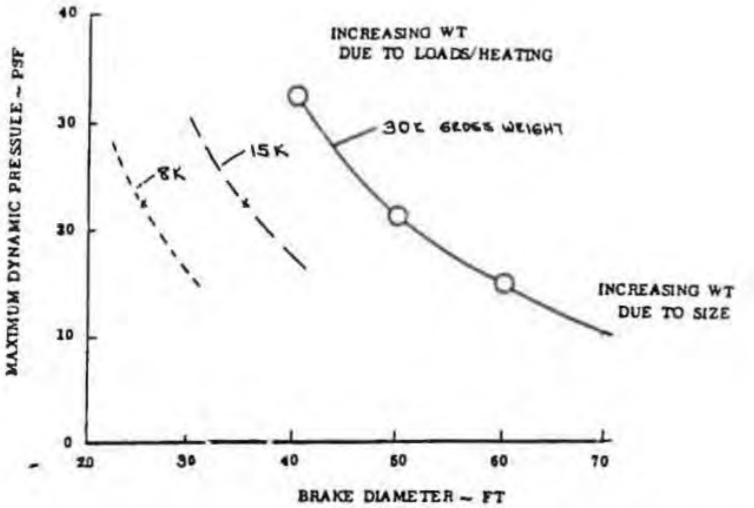


Figure 6-40
Effect of Maximum Dynamic Pressure and Vehicle Weight on Brake Size

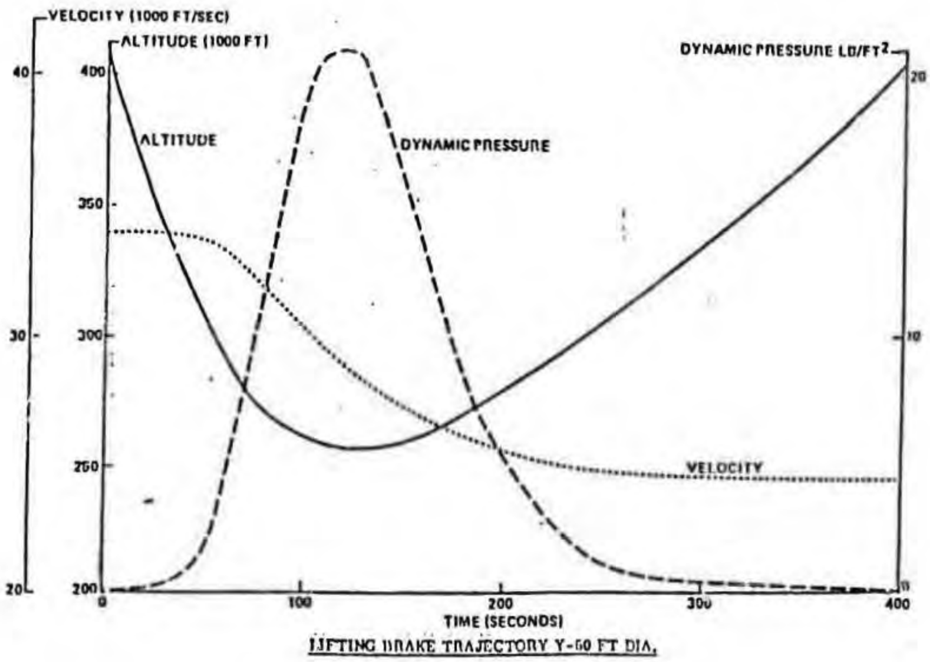


Figure 6-41
Typical OTV Aerobraking Maneuver Characteristics

• Baseline stage burn-out	5,738	
• Lifting aerobrake system	3,300	
• Spaceplane	5,800	
	14,838	Total B/O (min)
Expendables		
Main propellants	43,962	
He, N_2H_4	700	
	59,500	Total ignition

LEO-GEO-LEO transfer - 20,000 ft/sec

$$\therefore 20,000 = (Isp)(g_c) \ln \frac{59,500}{M_f}$$

Additional payload = M_f - 14,838 = 1121 lb

AV242366 820505 GN1069

Effect of Aerobraked Maneuver on Spaceplane GEO Mission

Figure 6-42

TECHNOLOGY AREA	CRITICALITY	UNCERTAINTY
AEROTHERMO-DYNAMICS	1 ENABLING	• UPPER ATMOSPHERE DENSITY • ANALYSIS OF FREE MOLECULAR SHIP FLOW • POSSIBLE HEATING PROBLEMS IN WAKE • STABILITY, CONTROL RESPONSE, DAMPING
HIGH TEMP MATERIALS & TPS	2 ENABLING	• BRAKE FACE MATERIAL CAPABILITY UNDER DYNAMIC LOADS (CERAMIC CLOTH VS METAL MESH) • INSULATION OF LIGHTWEIGHT STRUCTURAL RIBS • RSI ON CANNISTER AND DOORS (BEING IMPROVED FOR SHUTTLE)
NAVIGATION AND CONTROL	3 ENABLING	• PRECISE ENTRY INTO UPPER ATMOSPHERE • REAL TIME MEASUREMENT VERY SMALL QUANTITIES AS INPUT TO TRAJECTORY MODULATION • MULTIPLE PASSES TO ACCOMMODATE ERRORS & PHASING
STRUCT/MECH/ DESIGN	4 ENHANCING	• OPTIMIZE RELIABILITY AND WEIGHT (WRAPPED RIB VS UMBRELLA) • ENGINE CLOSURE DOORS MUST REOPEN AFTER BRAKING (SOLUTION AMENABLE TO GROUND TEST)
PROPULSION & ACS	5 ENHANCING	• VERY RAPID ROLL CAPABILITY AND DAMPING • POSSIBLE HOT PROPULSION COMPARTMENT • EXTENDABLE/RETRACTABLE MAIN ENGINE NOZZLE

NOTE: MAIN ENGINE DOES NOT FIRE DURING AEROBRAKING.

Aerobraking OTV Technology Assessment

Figure 6-43

of the Spaceplane, Centaur stage, with its other fluids (He, N_2 H_4) and the Airborne Support Equipment (ASE).

The Rockwell SOC Study (reference 22) has determined that the weight of the cargo bay payload plus the External Tank (ET) nominal propellant residuals varies between approximately 71,000 lb. (no cargo) and 74,400 lb. (65,000 lb. cargo). Therefore, in the case of a launch of the baseline configuration without propellant (~ 18,000 lb. cargo weight), some 53,500 lb. of cryogenic propellants remain in the ET at SSME shutdown (MECO) compared to the nominal Centaur SP value of ~47,300 lb.

An additional consideration for the dry-launch case is that, with a substantially reduced stage launch weight, it should be possible to reduce the dry-weight of the vehicle and of the ASE. Also, since the RTLS and AOA abort requirements are simplified for the Centaur (in particular the requirement for a rapid dump of the propellants), ASE weight could probably be further reduced.

The effect of mission performance is dependent on the actual system weight reduction, propellants transferred, etc. However, for illustration purposes, the following estimates were made:

Centaur SP weight reduction:	-500 lb.
ASE weight reduction:	-1500 lb.
Orbiter weight for transfer system:	+1000 lb.
Cargo bay (payload) weight:	-1000 lb.

Therefore, a "65K" Shuttle launch would carry a payload of approximately 17,000 lb. This, in turn, would provide residual propellants in the ET of some 54,500 lb. Rockwell assumed a transfer efficiency of 90%, thereby producing a propellant weight of 53,400, transferred to the Centaur Sp on orbit. (To minimize boil-off losses, Rockwell recommended a ground chill of the LH_2 tank to 170°R). This propellant weight is approximately 5,600 lb. greater than the baseline configuration.

The mission start weight is approximately 65,600 lb. (+ 5,600 lb. propellant - 500 lb. vehicle weight) and the stage burn-out weight would be approximately 11,040 lb., yielding a maximum ΔV capability of 27,080 ft./sec. (an increase of 2150 ft./sec. over the baseline.).

Further analysis of this concept should be undertaken, regarding the general subject of propellant transfer. Specific analyses, relative to the Centaur, should be in such areas as evaluation of weight reduction potential, the allowable Centaur tank pressures during transfer (considering the common bulkhead ΔP limits) and the safety aspects of the dry launch and of an abort from orbit following a propellant transfer.

6.10.4 DOCKING ADAPTER

The current baseline Spaceplane/Centaur interface is the forward end of the payload adapter which is a hard mount system with a one-time separation capability. A Spaceplane mission might call for many small ΔV maneuvers at a specific target site (e.g., rendezvous and repair of a satellite). For this type of maneuver, it would be quite advantageous to separate the Spaceplane from the Centaur for efficiency of control and conservation of propellants. The Spaceplane would then dock with Centaur for the next major maneuver.

Of the U.S. docking systems used in the past, the Apollo/Skylab, Apollo/LM, and ASTP modules were all much more complex than is required by the Spaceplane system. This is primarily due to the pressurized tunnel requirements for manned egress, which is not required here. (The Gemini/Agena system may have been similar but no data on that system was found during the current study).

The unique feature of a docking maneuver would be that the pilot would be backing up, probably requiring some sort of a mirror alignment system. Other requirements, such as capability by the Centaur for stationkeeping and for a single electrical cable connection, appear to present no significant problems.

6.11 PROGRAMMATICS

The total vehicle development program can be accomplished to achieve a 1987 first launch capability. The development program is based on several major assumptions.

- The Centaur F is integrated into the Space Transportation System for a 1986 launch.
- There is an on-going Atlas/Centaur production and launch support program.
- Centaur SP development program go-ahead is January 1984.
- There is an on-going RL10 product improvement program.

The estimated total vehicle development program cost in FY82 dollars is $145 million. This estimate does not include contractor fee, propellant cost, or system integration testing. This latter item cannot be realistically estimated until a site for such testing (for example, the AEDC J-4 test call) is established.

6.12.0 CONCLUSIONS AND RECOMMENDATIONS

The results of the analysis of an external propulsion system for the Spaceplane has led to the following conclusions:

1. An external propulsion can be designed, using State-of-the-Art technology, which will provide a useful ΔV on the order of 25,000 ft./sec. to a 5,800 lb. Spaceplane.

2. The external propulsion system must be based on LH_2/LO_2 main propellants.

3. In order to achieve a system which can be operational in the 1987-1988 time period at a reasonable cost and risk, the vehicle must be powered by an RL10 Derivative main engine. It also appears that the stage design should be based on the Centaur vehicle.

4. One of the major limitations on upper-stage propulsion capability is the 65,000 lb. weight limit of the Shuttle orbiter (a ground rule for this study.)

5. Several potentially significant technology options were identified, which can enhance the usefulness and flexibility of the Spaceplane/external propulsion system.

Based on these conclusions, the following recommendations are made for the follow-on Spaceplane system study:

1. The baseline for the external propulsion system of a modified Centaur and RL10 Derivative II main engine should be continued and refined.

2. The follow-on study should refine the airborne support equipment (ASE) characteristics considering Spaceplane mission requirements.

3. The possibility of increased Shuttle lift capability should be investigated (e.g., a lower altitude orbit).

4. The potential advantages of an aerobraking return from a high altitude orbit (e.g., by use of a lifting aerobrake) should be further investigated.

5. The potential use of residual External Tank propellants should be further refined.

6. The possible reduced system dry-weight (and thereby, improved mass fraction) by the use of a "dry launch" technique, should be further refined.

7. For enhanced maneuvering capability, the use of a relatively simple Spaceplane-to-Centaur SP docking system should be evaluated.

REFERENCES

Ref. No.	Title		
1	Orbit-To-Orbit Shuttle (Chemical) Feasibility Study	Rockwell	1971
2	Orbit-To-Orbit Shuttle (Chemical) Feasibility Study	McDonnell Douglas	1971
3	Space Tug Point Design Study	McDonnell Douglas	1972
4	Space Tug Systems Study	General Dynamics	1973
5	Space Tug Systems Study	McDonnell Douglas	1973
6	Baseline Space Tug Configuration Definition	NASA - MSFC	1974
7	Manned Geosynchronous Mission Requirements and System Analysis Study	Grumman	1979
8	Orbit Transfer Vehicle Concept Definition Study	General Dynamics	1980
9	Orbit Transfer Vehicle Concept Definition Study	Boeing	1980
10	Centaur F Technical Description	General Dynamics	1981
11	Orbit-To-Orbit Shuttle Engine Design Study	Aerojet	1972
12	Orbit-To-Orbit Shuttle Engine Design Study	Pratt & Whitney	1972
13	O_2/H_2 Advanced Maneuvering Propulsion Technology Program	Rocketdyne	1972
14	Advanced Space Engine Preliminary Design	Rocketdyne	1973
15	Advanced Space Engine Preliminary Design	Pratt & Whitney	1973
16	Application of RL10 Engine for Space Tug Propulsion	Pratt & Whitney	1973
17	Design Study of RL10 Derivatives	Pratt & Whitney	1973
18	Orbit Transfer Vehicle Phase A Engine Study	Aerojet	1979
19	Orbit Transfer Vehicle Engine Study, Phase A	Rocketdyne	1979
20	Orbit Transfer Vehicle Engine Study	Pratt & Whitney	1979
21	OTV Advanced Expander Cycle Engine Point Design Study	Pratt & Whitney	1981

7.0 VEHICLE SYSTEM EVALUATION

The principle performance features of the Spaceplane vehicle concept are:

- o Payload-maneuverability
- o Synergistic plane change
- o "Zero-speed" landing

From the vehicle performance evaluation point of view payload-maneuverability and synergistic plane change performance dominate.

7.1 PAYLOAD-MANEUVERABILITY

The basic component of payload-maneuverability is payload-velocity, the change in velocity or delta-V that the Spaceplane can give to a payload as a function of the payload weight and the Spaceplane's configuration. Payload-velocity is the normalized measure of payload-maneuverability in the sense that the total velocity available with a given payload can be used to perform a wide spectrum of maneuvers. We can evaluate performance in terms of payload-velocity without loss of generality. Several example maneuvers should then serve to present the transformation of payload-velocity to payload-maneuverability.

Payload-velocity function space can be used to evaluate Spaceplane performance. Figure 7-1 presents payload-velocity as a function of payload weight for several Spaceplane system configurations. The logarithmic curves result from the rocket performance equation.

The lowest curve represents the Spaceplane configuration in which the payload bays are empty and the vehicle is ballasted with the 492 lbm of ballast appropriate for atmospheric flight stability with empty payload bays. The maximum velocity of 2647 FPS occurs with zero payload, a vehicle gross weight of 5601 lbm, and 1281 lbm of usable propellant. A velocity of 2578 FPS is obtained with a payload weight of 250 lbm. This configuration meets the contractual performance specification of 2500 FPS with a payload of 250 lbm.

The curve marked bay-propellant represents the performance obtained by adding conformal auxiliary propellant tanks in the two payload bays and pumping the propellant to the PCE. A zero-payload velocity of 3714 FPS obtains and for 2500 FPS a payload weight of 2513 lbm results. The auxiliary propellant thus results in an order of magnitude increase in payload weight at a velocity of 2500 FPS. The wet weight of the vehicle with auxiliary propellant is 6291 lbm, with a usable propellant weight of 1921 lbm.

For reference, the maximum capability of the Shuttle Orbiter is overlaid. Operating at a very low mass ratio, its payload velocity curve is nearly a straight line. At zero payload the Orbiter has a maximum velocity increment of approximately 1000 FPS.

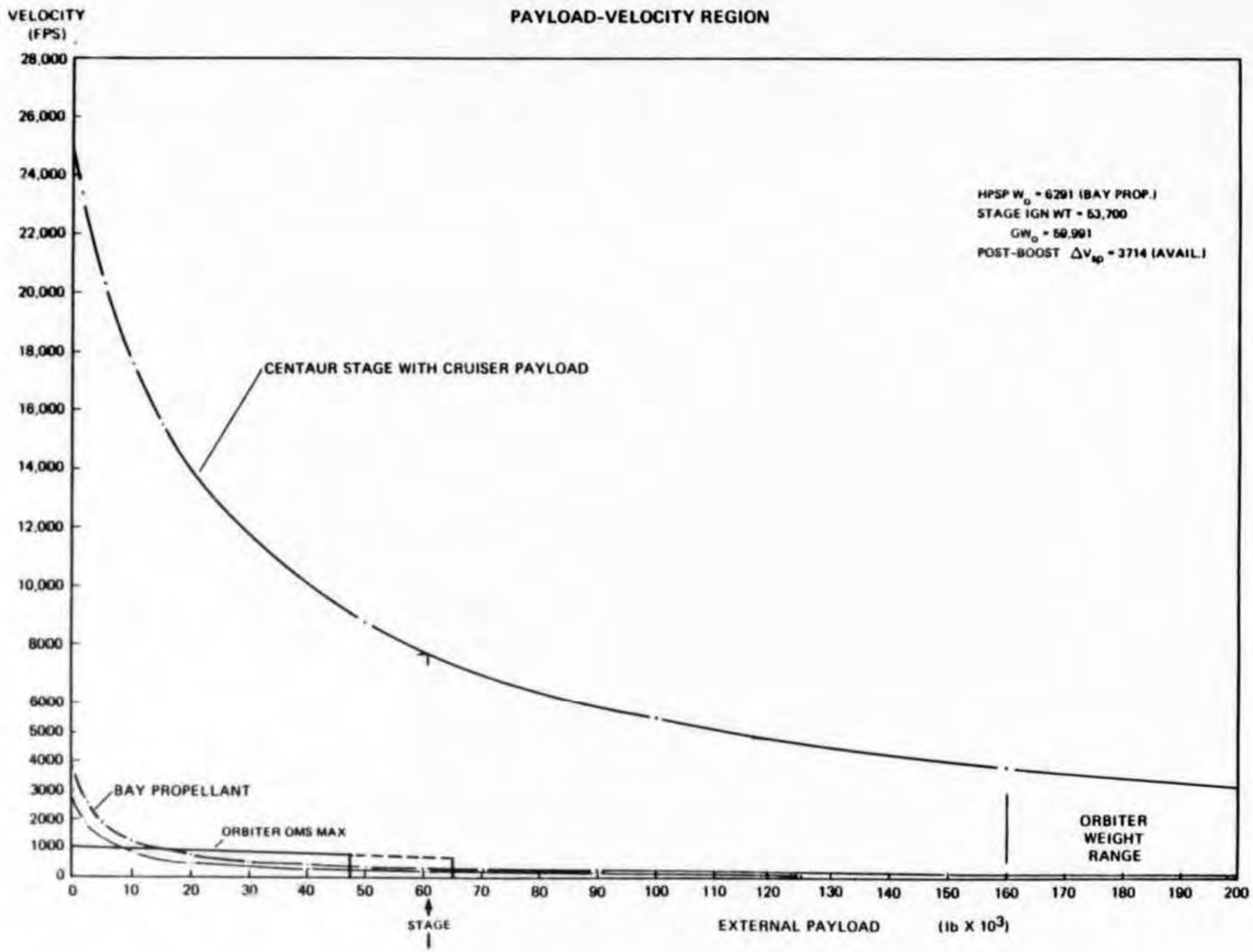


Figure 7-1 Payload-Velocity Map

The shuttle orbiter can carry up to eight Spaceplanes on a single flight, each of which has a capability of approximately 2600 ft/sec with a pilot and 250-lb payload. This greatly increases the scope of manned maneuvers possible from a single shuttle launch.

The upper curve represents the performance of the Centaur-Spaceplane when the Centaur is used as a stage or propulsion module. Its zero-payload velocity is 24,400 FPS and it can provide 20,000 FPS to a payload of 5,000 lbm. This corresponds to a 45-degree plane change at an orbital altitude of 100 nmi.

If another Centaur propulsion stage is added in tandem, it would see a payload of the Centaur-Spaceplane (60,000 lbm) and would provide 7600 FPS as a first stage. The Centaur-Spaceplane adds 24,400 FPS and finally the Spaceplane adds 3700 FPS. This adds up to approximately 35,714 FPS. The Apollo 15 used 28,832 FPS to land on the Moon and return.

The transformation of velocity into typical maneuvers in space is a straightforward matter. For example, a coplanar transfer from a 100n.m. circular orbit to a 900 n.m. circular orbit and return to the original orbit requires approximately 5000 ft/sec, A Spaceplane with a 2500-lb drop tank assembly could perform this maneuver with an 800-lb payload.

7.2 SYNERGETIC PLANE CHANGE

The detailed evaluation of the Spaceplane's performance in terms of synergetic plane change is the responsibility of the Sandia National Laboratories and will be included in its Final Report. Preliminary data supplied by Sandia indicate that a 10° plane change will require about 2300 ft/sec.

The use of auxiliary propellant in the payload bays should increase the synergetic plane change angle significantly by permitting an increase in velocity loss due to drag while retaining a sufficient delta-V to return to orbit and deorbit to recover. Plane changes up to 15-16° might be accomplished in this manner.

8.0 AVIONICS

8.1 INTRODUCTION

Honeywell is pleased to submit this final report on the Spaceplane Avionics Feasibility Study as an unfunded member of the study team. Honeywell acknowledges the conceptual leadership provided by SRI in providing timely inputs and constructive suggestions which have played a most significant role in bringing this phase of the program to a successful completion.

In this report we will review the Spaceplane requirements from an Avionics point of view, go over the baseline, and touch on some of the significant trade-offs involved in their selection. A lack of funding prevented complete analyses, but those performed are summarized. A major output for use by other study members, the systems physical characteristics of the Avionics System Baseline is given. Finally, critical tasks are recommended for the next phase.

8.2 REQUIREMENTS

8.2.1 Overall Requirements

In establishing the Avionics system requirements the starting point was the design reference mission and the alternate mission for a GEO Rendezvous. These were both shuttle deployed. To insure flexibility to handle the full envelope of future missions, both an air launch and booster launch capability (as far as the Avionics is concerned) was also considered a requirement.

8.2.2 Given and Derived or Assured Requirements

Avionics requirements are summarized in Table 8-1.

8.2.3 Functions and Operational Modes

8.2.3.1 Prelaunch

The inertial systems and processor must be initialized before deployment or launch.

8.2.3.1.1 Shuttle Deployment

For a shuttle deployment, the shuttle position, velocity, time and orbital parameters must be transferred to the spaceplane navigation system as well as rough attitude. Right after deployment the spaceplane will update its attitude precisely by celestial observations.

8.2.3.1.2 Ground Launch

Ground launch can be accomplished by the pilot inserting all needed initial conditions manually by means of his controls. Inertial alignment in the vertical channels is achieved by means of the accelerometer outputs. Depending on the mission accuracy requirement and ready time, azimuth alignment can be achieved by gyro compassing, theodelite transfer, or pilot input through the controls.

8.2.3.1.3 Air Launched

The inertial system would be aligned before carrier aircraft takeoff and alignment trimmed by matching to the mothership inertial system. Position and velocity before separation would be automatically transferred from the mothership inertial system.

8.2.3.2 Rocket Powered

When under rocket power, whether self-contained or with lower stages, all steering signals are generated in the Spaceplane computing system.

For vacuum burns a single attitude is calculated and held during the burn with the rocket cut off when the desired delta V has been

Table 8-I

SUMMARY OF OVERALL REQUIREMENTS

Mission length of 24 hours.

Dormant with near instantaneous ready time.

Provide continuous attitude in all modes.

Provide continuous navigation in all modes.

Provide continuous attitude control in all modes.

Provide velocity and translation control in all three axes.

Allow input flexibility to permit major mission changes in real time.

Require minimum temperature conditioning.

Tolerate high vibration levels.

Rendezvous with friendly target and unfriendly targets in precisely known orbits

Use minimum size, weight, cooling and power consistent with technology expected to be developed in the Spaceplane era.

For time critical mission phases such as boosting or re-entry provide fail operational capability.

For non-time critical functions provide a back-up capability if pilot recovery is involved.

Specific Numerical Parameters:

Longitudinal Acceleration	+ 6G - 3G	Roll Rates	$\pm$ 30^{o}/Sec
Yaw and Pitch Acceleration	$\pm$ 5G	Yaw and Pitch Rates	$\pm$ 10^{o}/Sec

accumulated. For ground and air launch rocket boosting through the atmosphere, a nominal zero g trajectory will be followed to minimize aerodynamics loads, with modification as necessary to accommodate winds aloft. While rocket firing, steering, staging, and rocket cut off nominally be automatically performed; pilot override will be provided.

8.2.3.2 Free Fall Orbital Transfer

For normal free-fall orbital transfer, the pilot at his option may allow the vehicle to slowly tumble or put the vehicle in a reaction control jet control mode. Submodes should allow inertial hold at any desired attitude, Earth local vertical with any desired bias, a slow rotiserrie roll for thermal control, or a sequence of attitude maneuvers to perform celestial attitude and navigation updates.

During free fall in orbit when attached to a lower stage such as a large cryogenic stage required to achieve GEO capability, the spaceplane reaction control jets would not have had sufficient control authority to handle the combined vehicles, and hence control commands to the lower stage would control the lower stage reaction control system.

8.2.3.4 Free Fall Precision Pointing

During free fall precision pointing (or scanning) can be commanded by means of the mementum wheel control system. Where a precision payload may be in use, vigorous motion by the pilot must be avoided in the mode due to the limited command authority of a modest sized momentum system.

8.2.3.5 Rendezvous and Docking

Rendezvous and docking involves a series of modes that starts with a transfer orbit to intercept the target at a predicted location. After radar acquisition, a series of thrusting burns are required to insure an efficient use of the fuel in making the rendezvous. Final approach and dock can be normally expected to be accomplished manually with the pilot having full three axis rotation and translation control. The normal rendezvous will be programmed to maintain the target in the forward area so that target tracking by the rendezvous radar is continuously maintained. Co-mounted with the tracking radar is a viewing camera that allows the pilot to inspect the target at greater magnification than normal vision. For night rendezvous and docking approaches, also co-mounted with the tracking radar is a laser illuminator.

It is not anticipated that docking would be accomplished with a large lower stage attached, but rather than the lower stage would be "parked" and docking would be made with Spaceplane alone.

8.2.3.6 Station Keeping

For pilot convenience during inspection or other similar missions, a station keeping mode allows holding the spaceplane off a given distance from a target as well as at any desired attitude.

8.2.3.7 Normal Re-Entry From Low Earth Orbit

Re-entry, like rendezvous, is a series of modes starting with the re-entry burn which puts the Spaceplane in a transfer orbit to intersect with a desired re-entry window. In the normal re-entry mode the initial desired re-entry attitude is held by the reaction control jet system until aerodynamic control is established. A minimum heating or other criteral corrider is flown. If landing at a particular place is required, lateral maneuvering and energy management maneuvers will be utilized. While re-entry is largely flown on the inertial system, the radar altimeter can be used as a monitor or backup. When a critical velocity is reached, the Spaceplane recovery sequence is initiated. Once under low velocity flight, steering signals are provided to the recovery system to insure touch down at any location. Naturally, pilot override at any point in the re-entry sequence is provided for. At this point the Spaceplane re-entry profile is not well defined. for reference purposes, a nominal shuttle re-entry from low Earth orbit is given in Figure 8-1.

8.2.3.8 Re-Entry from Energetic Orbits

Re-entry from higher orbits which involve the need to dissipate significantly more energy are the same as a normal orbit except either a braking burn or the use of an aerodynamic brake preceeds the normal re-entry sequence.

8.2.3.9 Synergetic Plane Charge

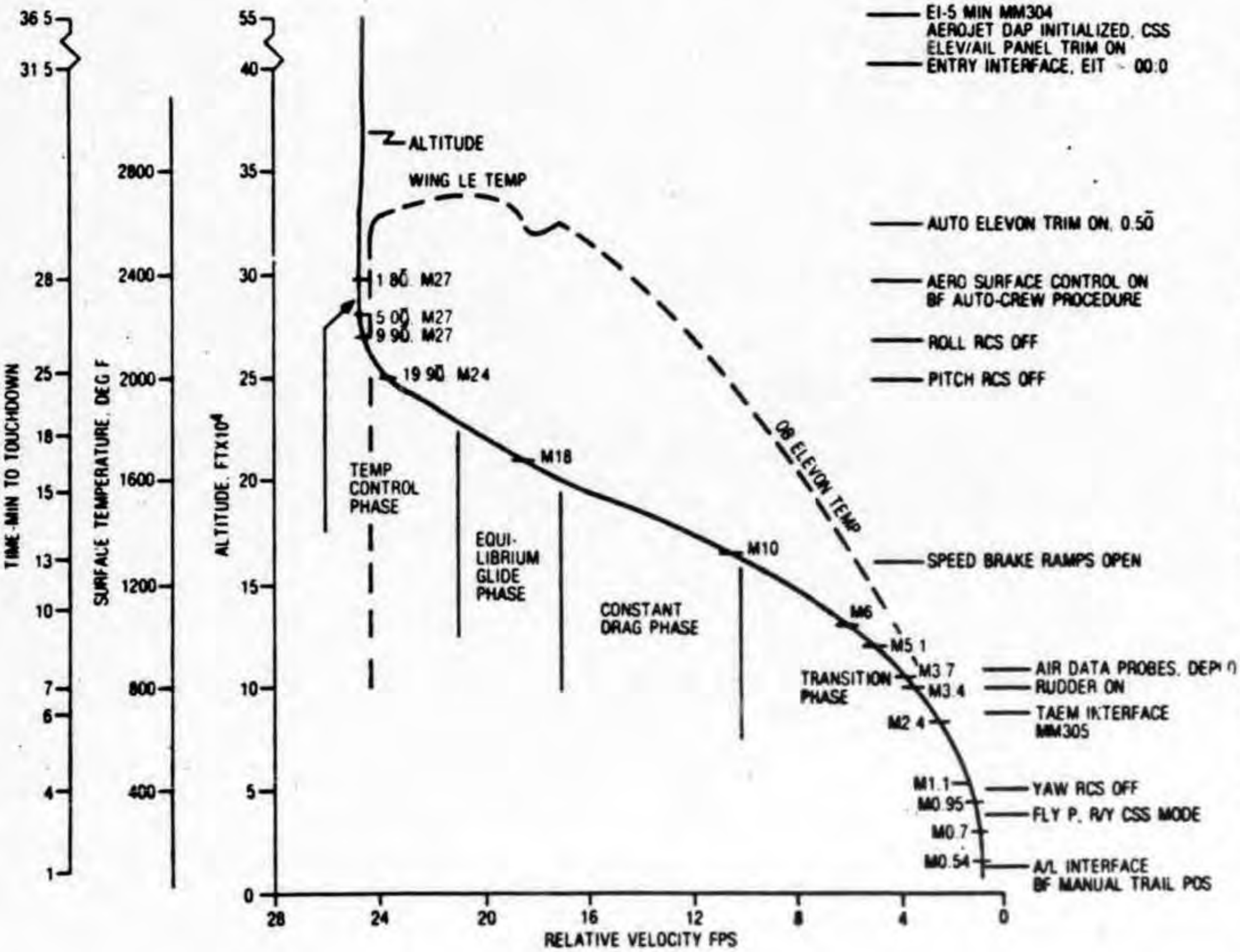
This sequence of modes starts out the same as a normal re-entry except that a higher window is targeted and a lateral as well as pullup aerodynamic maneuver is performed such that the spaceplane re-exits the atmosphere, but in a different and now sub-orbital plane. A subsequent burn can be performed if the velocity hasn't dropped to low in order to regain orbital status. The entire synergetic plane change maneuver would be done under the inertial system's control.

8.2.3.10 Ground Command

In the event of pilot disablement, it is anticipated that ground commands could initiate any of the above sequences that could be accomplished without the need for pilot action.

FIGURE 8-1

STS-1 ENTRY, NOMINAL MISSION



8.3 AVIONICS TRADEOFFS AND BASELINE

Because of the limited resources available, emphasis in the avionics system trade-offs and baseline selection was placed in those areas most critical in their effects on other aspects of the spaceplane. Areas judged non-critical received only cursory attention in that it was felt they could be dealt with in the next phase.

A top level block diagram of the Avionics System not identifying signal types or mode switching, is shown in Figure 8-2. This configuration is based on a dedicated custom interface design. At this stage an alternative input/output design using an I/O bus such as 1553B is also a viable candidate and would significantly alter the interfacing design.

To avoid obsolescent hardware with attendant weight, power, volume, and reliability penalties, the baseline hardware chosen was that which is near the edge of proven technology, but not so close as to involve development risk.

Most of the avionics hardware would therefore be modification of existing equipment, or could naturally be expected to be developed to a more mature status on other programs as the Spaceplane program comes into being.

At the heart of the avionics system are two three-gyro laser IMU's, appropriately oriented so that any failure is automatically isolated. The laser gyro IMU provides immediate use, does not require temperature control, and also has a capability for precise attitude reference while at the same time allowing body rates compatible with aerodynamic vehicles.

Automatic attitude updates are provided by the celestial sensor and horizon sensors, with a less accurate attitude update capability provided by the pilot using the TV camera sight. At low altitude, position and velocity would be updated by the use of GPS, while at high altitude, orbital parameters would be determined by combining knowledge of the Earth's local vertical with celestial observations and absolute time.

Using pilot or ground-input commands, the redundant computers would generate the appropriate control commands, whether rocket steering, reaction jets, aerosurfaces, or reaction wheels.

The pilot is given appropriate displays of his current situation and two very important displays for general orientation, a ball display showing the Earth under him as he would see it if he looked straight down, and a second ball display giving him his attitude with respect to the local vertical. The yaw reference could be selectable either to geographic North or to his velocity vector.

For both the Shuttle-deployed and air-dropped mode of launch, the system needs to be initialized from the mother vehicle. In the case of a 747 mother aircraft, position and velocity data needs to be continuously provided so that azimuth can be ascertained by velocity matching. Since the Shuttle is in free-fall, this is not possible, so a precise azimuth is ascertained by use of the celestial sensor.

FIGURE 8-2

Honeywell

FUNCTIONAL DIAGRAM
BASELINE AVIONICS CONFIGURATION

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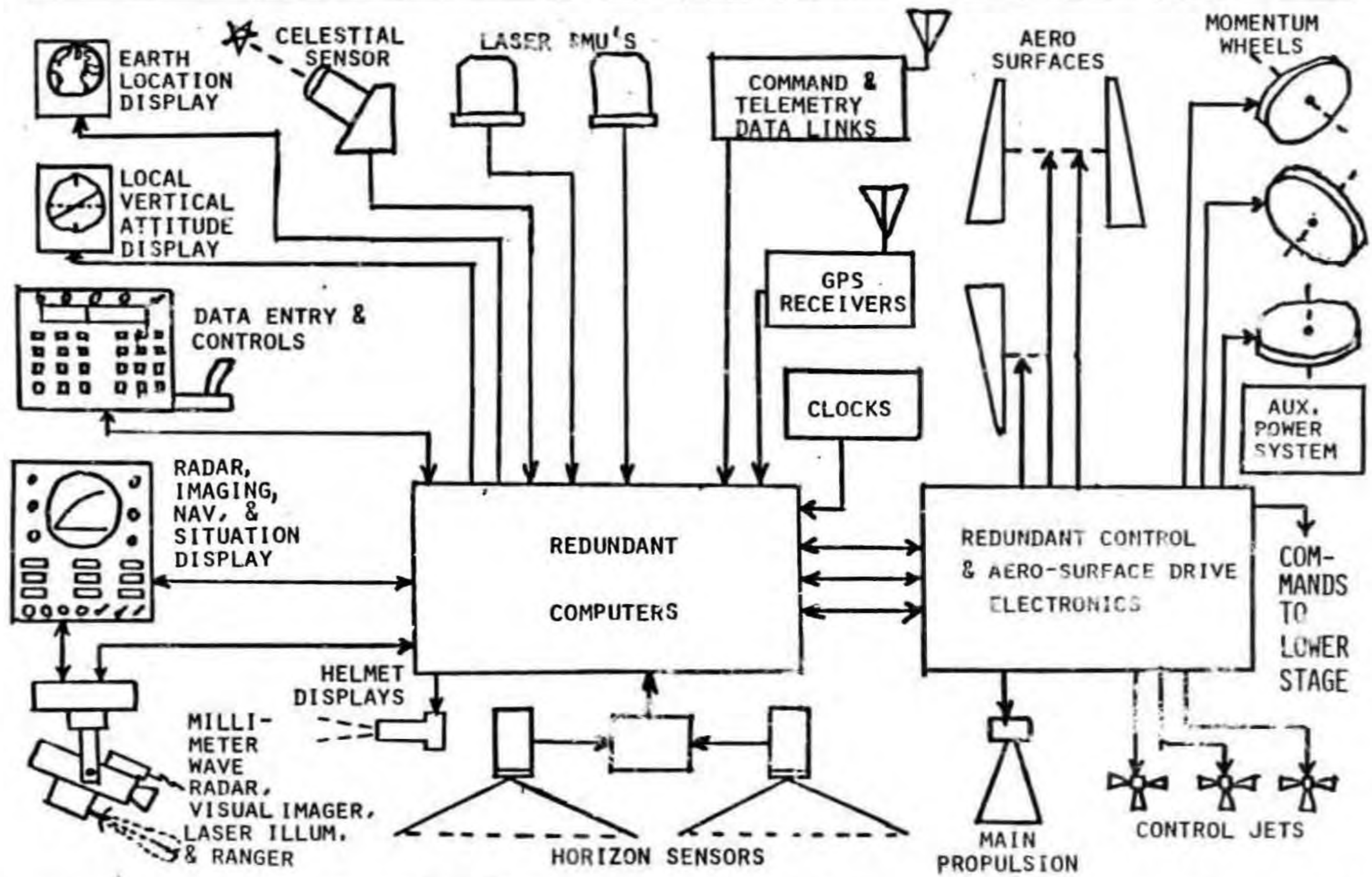
AVIONICS DIVISION



For minimum reaction time, the system should be capable of direct ascent to GEO. However, in the majority of cases a parking orbit provides more flexibility for phasing adjustments.

For ground and 747-launched missions, a parking orbit also allows a period to make precise updates before making the last burn to achieve a GEO transfer orbit.

For some operations, such as reentry or rendezvous the performance required may be so precise that a number of iterative observations and corrective maneuvers will be required.

8.3.1 Inertial Reference

An inertial attitude reference is needed for almost all operational modes, both powered, and free fall, and is a big element in the control of the spaceplane. In previous mechanizations a precision gyro package would be used for attitude reference and a rate gyro package would be used for high rates for rate damping. With the advent of laser gyros, a single accurate device is available with such a wide dynamic range that both functions of the previous separate attitude and rate gyros are combined in one device. This reduction in the required devices, plus the lack of need for temperature control, and immediate operation make the laser gyro the clear choice.

For purposes of providing fail operational capability during critical operating modes, either 3 IMU's or two skewed to each other can be used. If two IMU's each with orthogonal components, are arranged as shown in Figure 8-3, then each component in each triad is monitored by the three in the other, and visa versa. Parity equations mechanized in the navigation processor then can identify a failed gyro or accelerometer and operate on the remaining 5 components.

Honeywell has several sized RLG gyros in production, three being shown here in Figure 8-4. Performance currently being experienced is twice as good as the chart indicates. With attitude updates before major burns, the small gyro will provide adequate performance.

Honeywell has designed and produced about 10 laser gyro IMU's in recent years, the latest being the IMU for the Sentry Program. Because of the size and weight limitations for spaceplane, it is anticipated that a laser package would be optimized for spaceplane. A mockup of such a package which would utilize largely LSIC electronics is shown in Figure 8-5.

8.3.2 Celestial Sensor

The celestial sensor is required for precision attitude update after deployment from shuttle and also before any significant planned maneuvers and thrusting. Initially, a slit detector had been baselined as the device which could be implemented most simply. During the trade-off studies it was realized that if a solid state tracker type device could be used which could track stars close to the horizon, an autonomous navigation

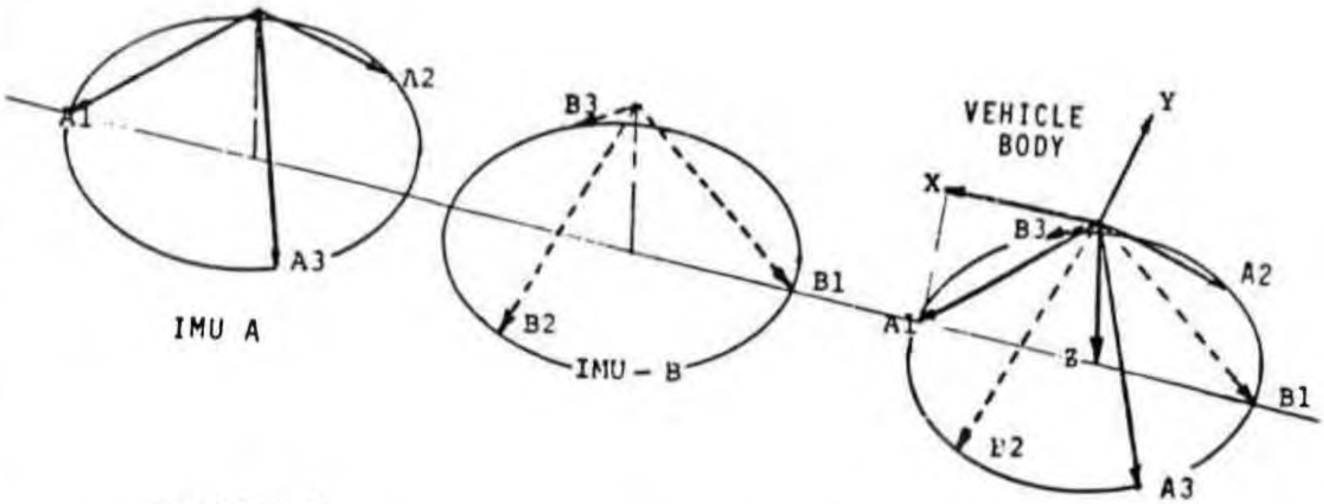


FIGURE 8-3
RELATIONSHIP OF SENSOR INPUT AXES TO BODY AXES FOR EWED IMU'S

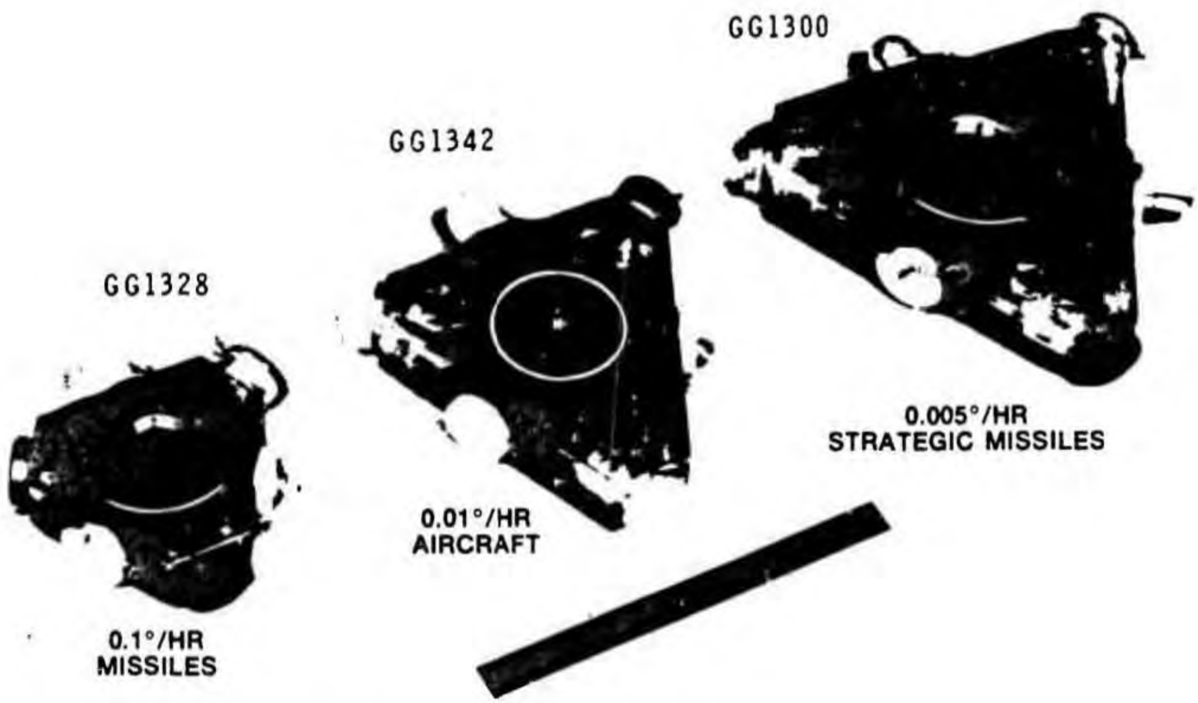
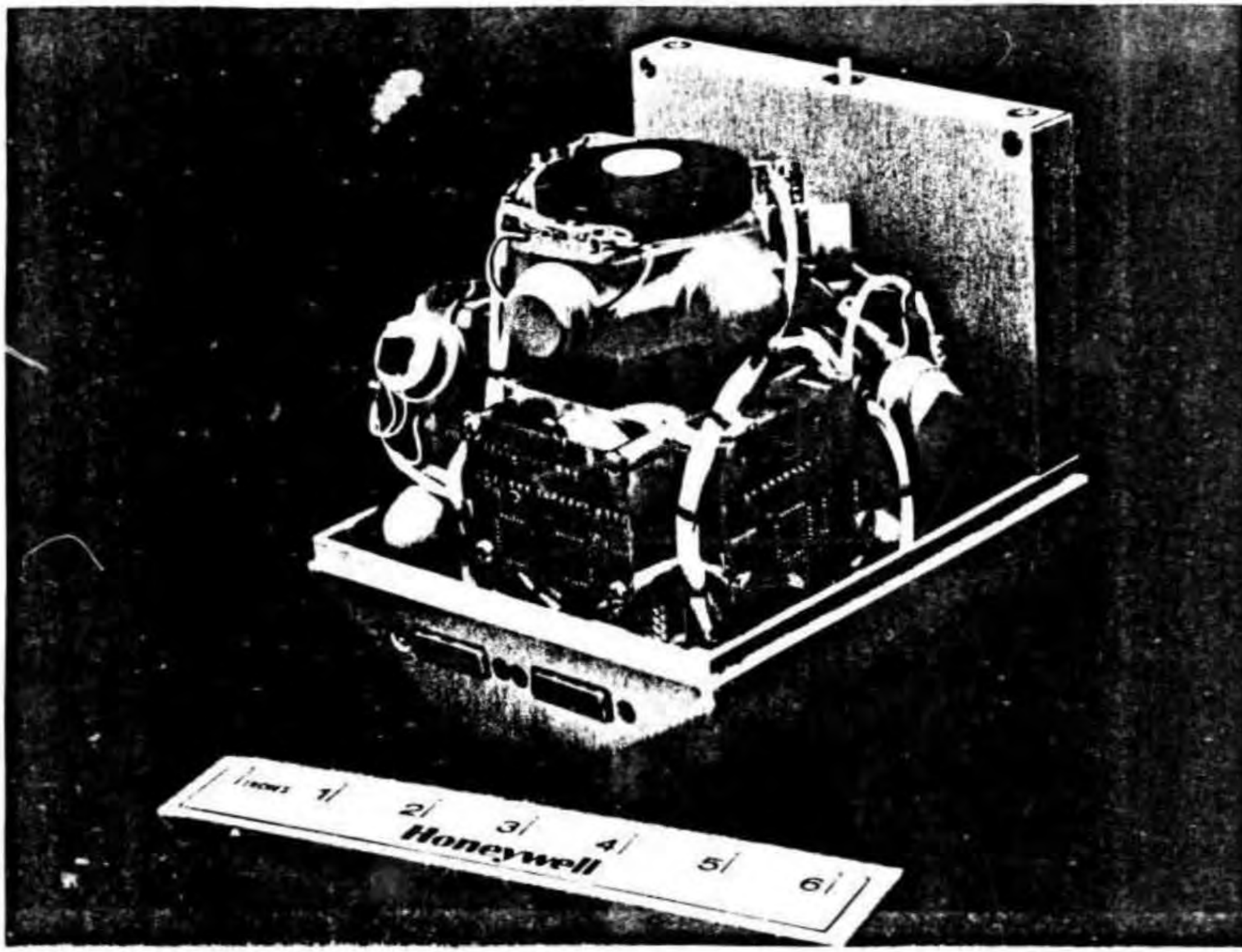


FIGURE 8-4 HONEYWELL LASER GYRO FAMILY

FIGURE 8-5

LASER GYRO INERTIAL ATTITUDE REFERENCE



capability could also be provided. Therefore, the baseline design was changed to a solid state tracker, representation of which is the Ball Bros. unit shown in Figure 8-6. The actual field of view, star sensitivity and sunshade design need to be optimized in future specific mission studies. It is anticipated that a field of view of $10^{o}x10^{o}$ with a sensitivity to 5th magnitude stars would be more than sufficient.

8.3.3 Horizon Sensor

With a normally operating system the attitude relative to the Earth will be known accurately at all times. As a backup to insure a safe entry in the event of only a partially operating system, Earth horizon sensing is provided. The major trade-off is between scanning and static Earth sensors. While the latter are the most accurate and reliable, they are also only suitable over limited altitude ranges such as are encountered by satellites in circular orbits. The scanning type Earth sensor can provide modest attitude information over a wide range of altitudes of two units are used, one for pitch and the other for roll. Ithaco horizon sensors as shown in Figure 8-7 are baselined for the spaceplane.

8.3.4 GPS Navigator & Precision Clock

During peacetime and medium to low Earth orbit operation, a GPS navigator will be the primary subsystem for providing the spaceplanes position. Since at least 3 electronic companies appear to be firmly committed to providing space qualified GPS equipment, it did not appear necessary to make a close study at this time. The equipment appears to be getting smaller all the time, so the recommended strategy is to make the selection downstream when other users have developed the optimum equipment. An example of current hardware available is shown in Figure 8-8.

For sizing, power, and weight estimates a vendor was chosen, in this case Texas Instruments because the microprocessor implementing their GPS system is the same as baselined for the spaceplane processor. The processor board of their GPS equipment is shown in Figure 8-9.

A trade-off not settled is the location and type of antenna required for GPS. The preferred location would be in the rear vehicle area and to survive through re-entry may need to be extendable. It is certainly desirable to have GPS capability during the recovery phase after re-entry to facilitate a precision touchdown.

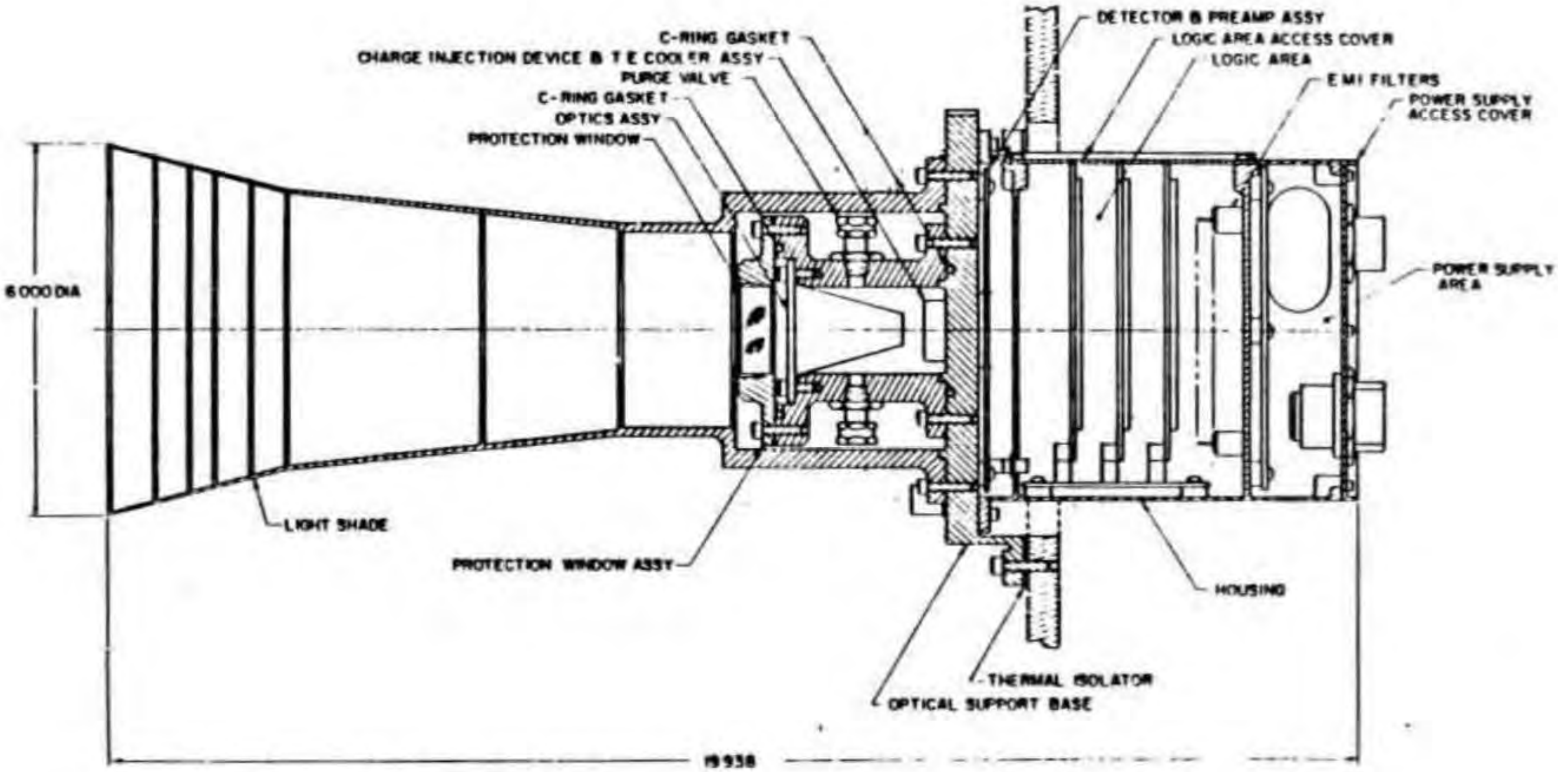
A precision clock which is usable by other avionics subsystems would normally be a component of the GPS hardware. For redundance purposes a backup lower accuracy clock will probably be required.

8.3.5 Tracking, Telemetry, and Command (TTC)

As with the GPS equipment there is a reasonable choice of TTC equipment, although some modifications will no doubt be required when the spaceplane missions are better defined, particularly what control network will be used such as existing ground systems or future satellite relay system. Physical characteristics were established that should be representative of the hardware avilable in the needed time period.

FIGURE 8-6

SECTIONED VIEW OF ADVANCED ELECTRO-OPTICAL SENSOR



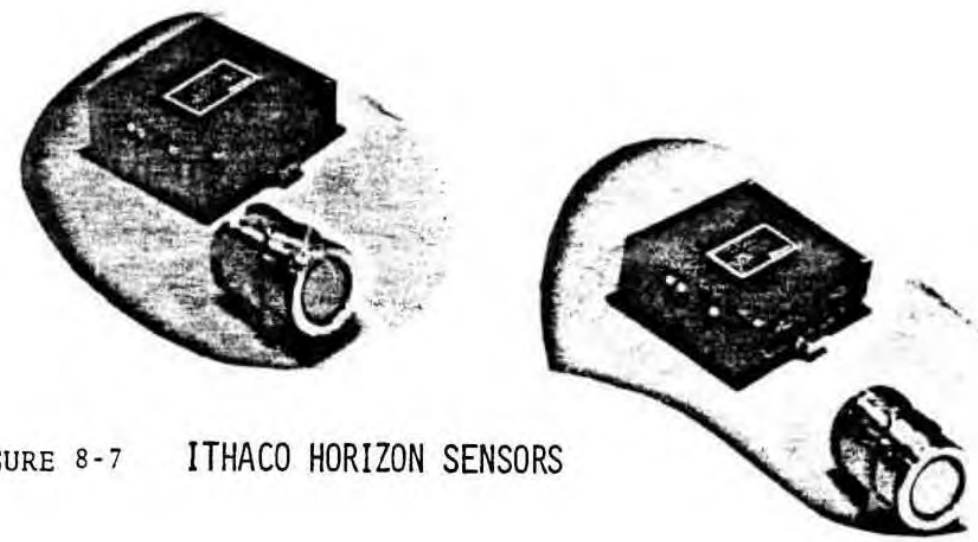


FIGURE 8-7 ITHACO HORIZON SENSORS

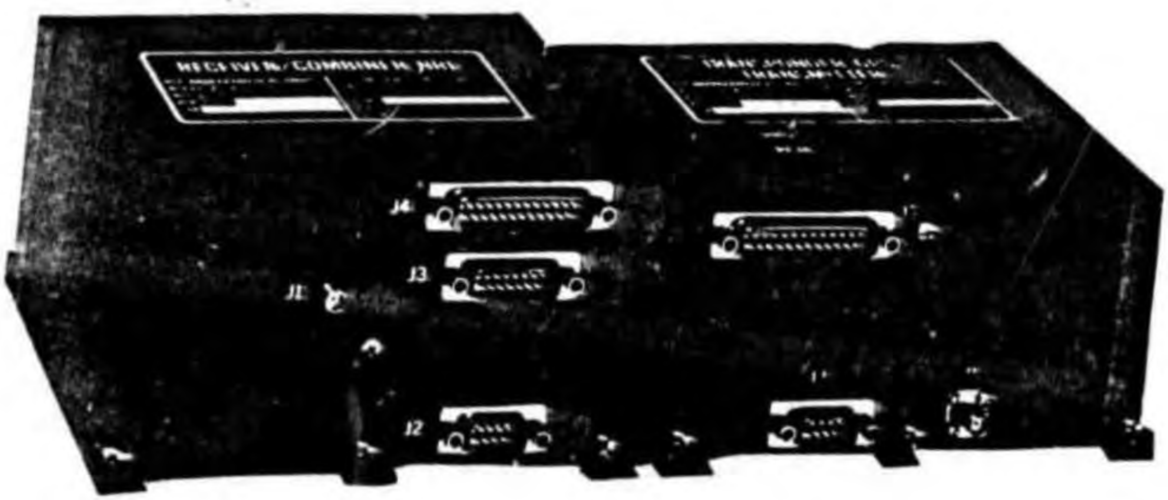
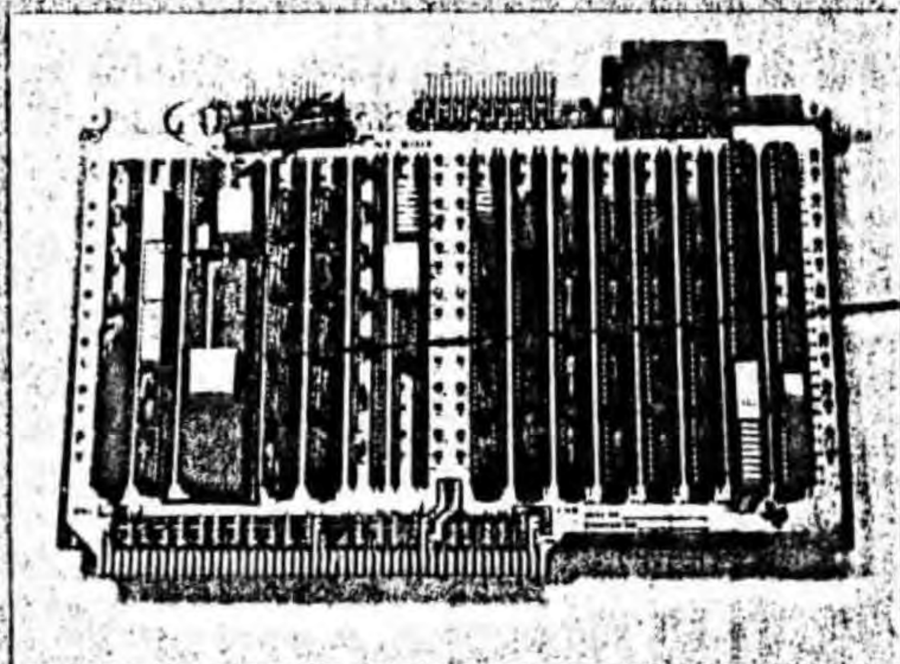


Figure 8-8 - Typical GPS Equipment

Honeywell Figure 8-9 AVIONICS DIVISION

SECOND GENERATION GPS MICROPROCESSOR MODULE



- SBP 9989 16-BIT MICROPROCESSOR
- I^2L BIPOLAR TECHNOLOGY
- PROTOTYPE TESTED AT 4.2 MHZ IN GPS ADVANCED DIGITAL RECEIVER BREADBOARD (ADRB)

- GPS ADRB MICROPROCESSOR MODULE
- SPACE QUALIFIED 6.56 " X 6.89" MODULE UNDER DEVELOPMENT

Rockwell on the GPS satellite program has recently issued RFQ's and will be embarking on a "Micromin Hardware" program for the GPS TTC which should insure that a next generation miniatured system will be in being.

As with the GPS system suitable antennas will be required, again probably of the retractible type located at the rear of the vehicle.

8.3.6 Radar Altimeter

Whereas a normal re-entry will be implemented by navigation utilizing the GPS and inertial subsystems, a radar altimeter is required as part of the back-up re-entry system to be used in conjunction with the horizon sensor. The ideal configuration would be to use the rendezvous millimeter wave radar in a altimeter mode. Future trade-offs are required to determine whether a door or radome can be appropriately located so that the radar altimeter can be operated with the nose access closed. One alternate would be to use the gimbal mounted millimeter wave radar transmitter with a second antenna built flush into the spacecraft structure, or separately deployed in the rear area. The worst case would be if a completely separate unit is required. The baseline assumes that at a minimum the rendezvous radar could be utilized but with a separate small antenna. Since the second antenna would not be gimbaled, during radar altimeter operation the vehicle would need to be oriented to some nominal atitude to insure an adequate signal return.

8.3.7 Momentum Controls

For precision pointing and low jitter operation during inspection and/or surveillance missions, a precise attitude control subsystems is required which is satisfied by momentum reactions wheels. Four Sperry wheels were initially selected for baseline.

Since the wheels are not required for any mission mode involving safety, it was decided to eliminate the redundant wheel in the initial baseline and provide only three wheels. Should a wheel fail, the pilot can switch that channel to jet control and continue operation with degraded performance. Three momentum wheels made by Sperry of the type contemplated are shown in Figure 8-10.

The units selected would need to have more momentum than those shown, although further study is required to determine the exact sizing. Wheels weighing 10 lbs. each can supply about 13 oz. in torque.

8.3.8 Reaction Control Jets

While the reaction control jets are not part of the Avionics system proper, close coordination is required to assure a good spaceplane control system. The most critical modes where the reaction control jets are used are rendezvous and docking. Jet sizing and location must be such as to provide pure rotation, translation, and combinations of both about and along all three axis. The level of acceleration in thrusting and angular acceleration required will be determined in the next phase as the result of study of rendezvous and docking by Lin Com. The pulsing length must be capable of short durations in order to insure reasonable control

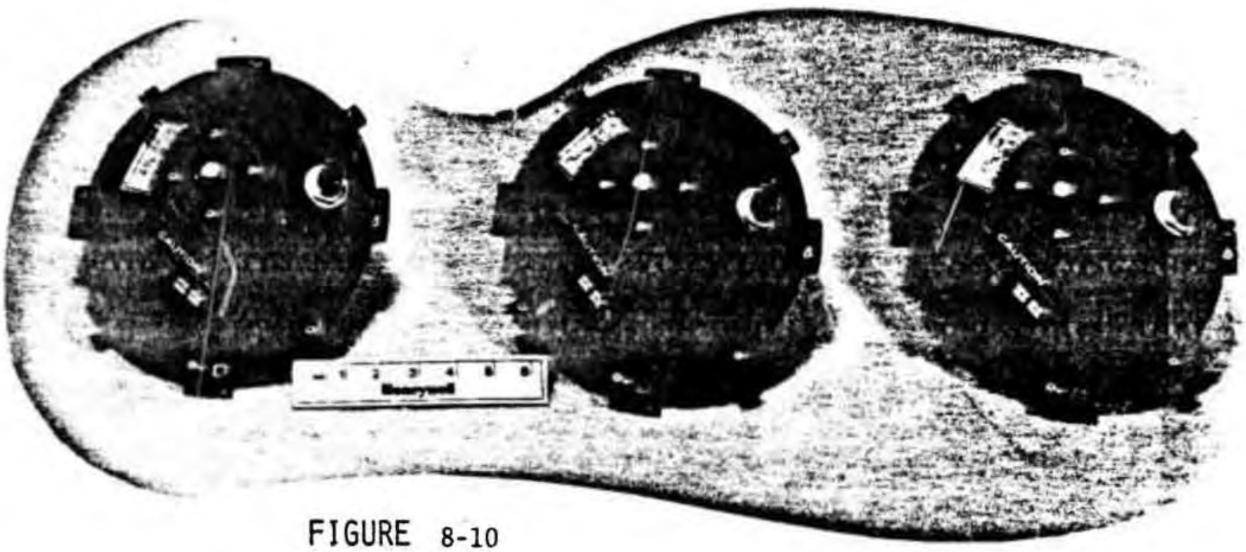


FIGURE 8-10

SPERRY MOMENTUM REACTION WHEELS

without large amplitudes of limit cycling as well as inefficient depletion of fuel. Sizing must also be adequate to control the vehicle during the transition from space to the upper atmosphere where the aerosurfaces take over.

8.3.9 Aerosurface Electric Drives

Honeywell is responsible for the flight control system on the space shuttle and considerable trade information was available because the space shuttle had recently completed a study of hydraulic versus electric drivers for the primary aerosurfaces.

The motivation for the study was to eliminate the contamination of the vehicle by hydraulic fluid, reduce weight, save power, and improve reliability. The study concluded that all four of these objectives could reasonably be expected to be met. An examination of the shuttle data indicated an electric drive for the spaceplane should provide the same advantages, and hence an electric drive was baselined. Experience with the electric drive is actually quite mature, some examples being shown in Table 8-2.

Table 8-2

INDUSTRY EXPERIENCE WITH ELECTRIC DRIVES

- O NASA DRYDEN PROGRAM USING AIR RESEARCH "POWER HINGE"
- O NASA AMES/BOEING LINEAR ACTUATOR ON THE QSRA AIRCRAFT
- O NASA'S ALL ELECTRIC ACTUATOR TRANSPORT
- O BOEING E727 FOR AN ALL ELECTRIC RUDDER
- O F14 FLIGHT TESTS WITH ELECTRIC RUDDER ACTUATOR

There are two basic types of electric drivers--the linear actuators designed as one-on-one replacements for hydraulic piston actuators, and rotary drivers. Because electric motors are naturally rotary devices, for new designs, the rotary type device will weigh less for a given horsepower rating.

Figure 8-11 depicts an electric actuator being considered for future application to space shuttle and of course is much larger and more powerful than required for the spaceplane. The solution for spaceplane cannot be made without hinge moment data but it can be assumed that duty cycle experience with space shuttle will be similar on spaceplane.

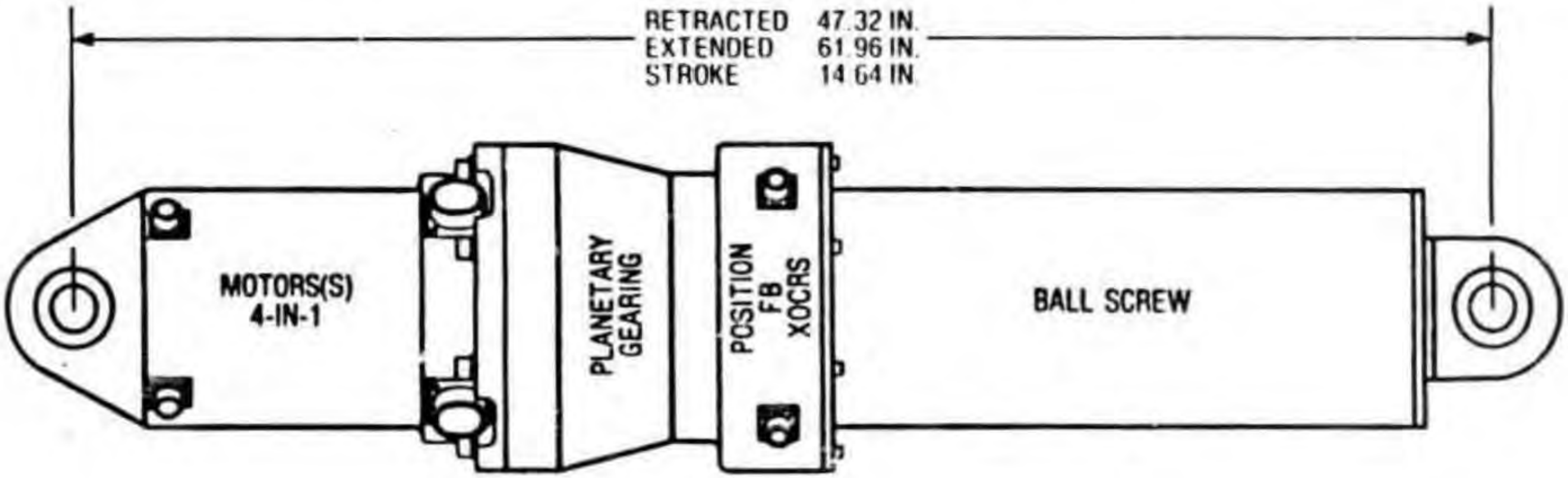
On that basis, an Air Research EM type actuator called the Electric Hinge appears attractive, as shown in Figure 8-12.

FIGURE 8-11

SHUTTLE

Honeywell

EM DEVELOPMENT ACTUATOR CHARACTERISTICS

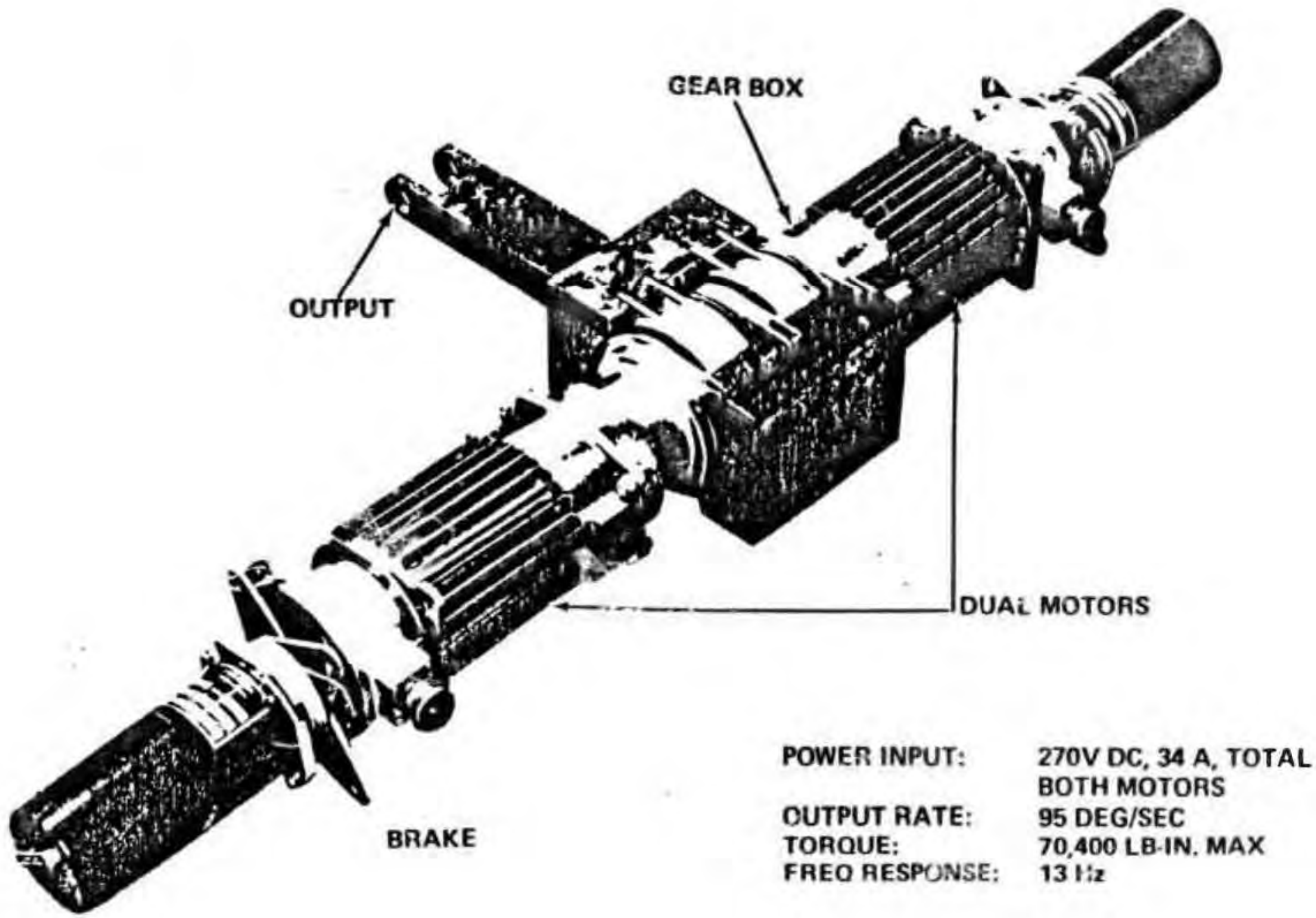


CHARACTERISTICS

- Quad redundant motor; brushless PM, rare Earth magnets
- Quad redundant rotary position sensors
- Quad redundant linear position sensors
- Force: 50,100 pound minimum at stall with two faults
- Rate: 7.84 inch/second minimum at 19,000 pound force with two faults
- Power: 270 vdc
- Cooling: passive mass heatsink

FIGURE 8-12

AIRESEARCH EM ACTUATOR



This device is capable of providing 4 H.P. on a continuous basis and a peak of 12 to 20 H.P., depending on the duration of the peaks. It weighs approximately 12 lbs. A lighter, lower power version should be capable of satisfying Spaceplane requirements when they are firmed up.

It would be packaged with both motors on the same side with a planetary gearbos to give a more compact package as is shown in the Hamilton Standard layouts.

Because aerospace control is required during the critical re-entry period, full redundancy is required with a monitoring means to be able to disable a malfunctioning drive. Such a redundant drive system is shown in Figure 8-13.

8.3.10 Rendezvous and Docking Subsystem

Three general types of rendezvous can be envisioned:

1) Rendezvous with friendly targets in reasonably well known orbits carrying corner reflectors and/or transponder beacons.

2) Uncooperative targets in well known orbits.

3) Uncooperative targets in poorly defined orbits.

In conjunction with Lin Com it was concluded that modest instrumentation would allow rendezvous and docking for the first two conditions, whereas very special bulky sensors such as LWIR telescopes would be required for the third case. It was decided that the standard complement should address rendezvous capability for the first two cases and that rendezvous in the third case would not be attempted unless a special mission payload for this purpose was carried in the payload bay. Having arrived at this distinction, further effort was expended on the first two types of rendezvous only. While not effecting the rendezvous sensors, it is of extreme importance to recognize for control purposes that many rendezvous would be made while attached to a large, probably cryogenic stage.

Functions required for rendezvous and docking are shown in Table 8-3.

Table 8-3

RENDEZVOUS AND DOCKING FUNCTIONAL REQUIREMENTS

Detection and Acquisition

Range and Range Rate Tracking

Angular Tracking and Angular Rate

Visual Inspection both Day and Night

Close Range Very Precise Range and Range Rate.

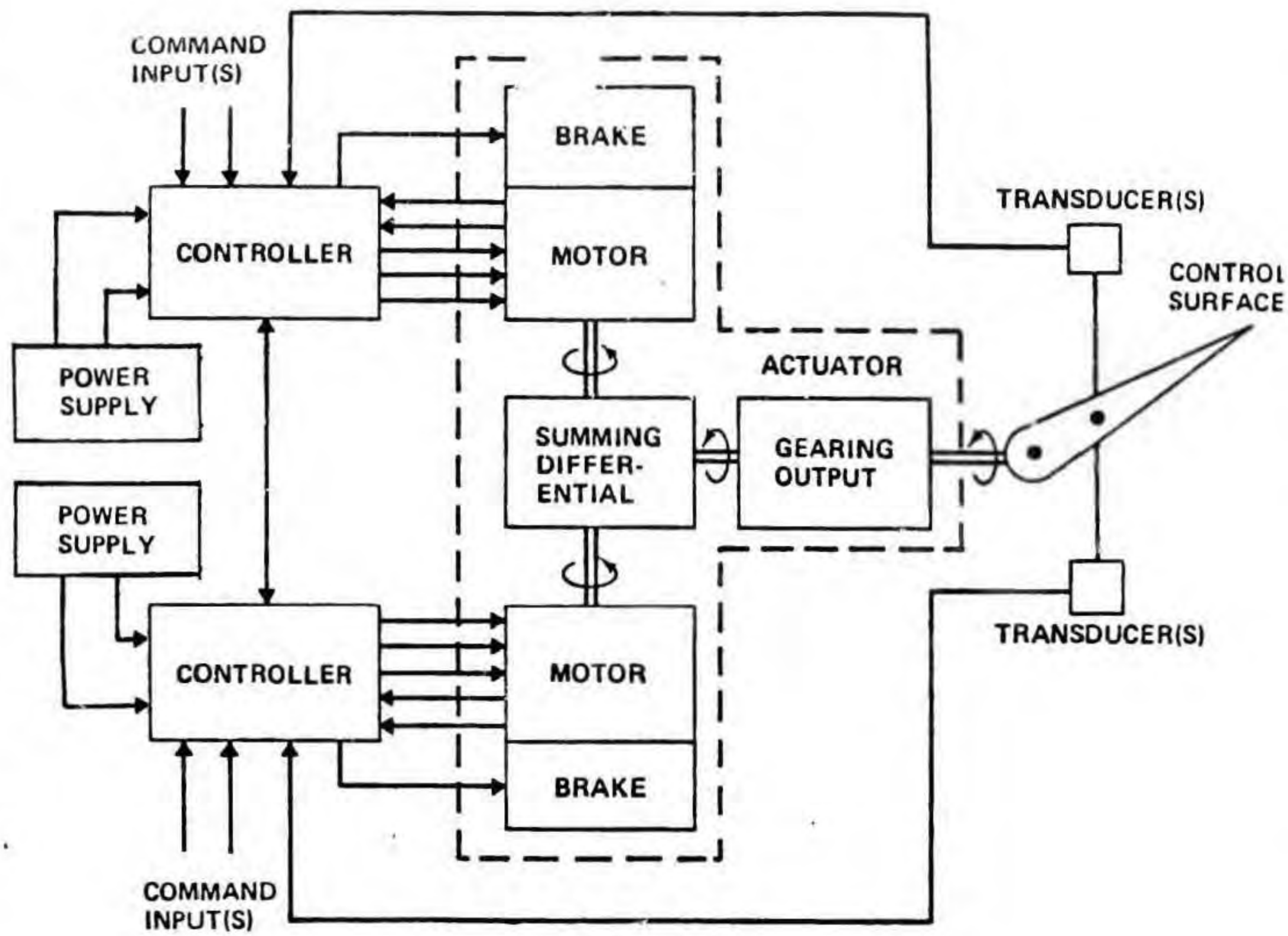


FIGURE 8-13

AIRESEARCH ACTUATION SYSTEM CONFIGURATION

Because of the ranging and range rate requirements it was early on recognized that active systems would be required, I.R. would not suffice.

To provide the functions of Table 8-3 it became clear that a gimbal system would be required to isolate the sensors from the vehicle motion and at the same time allow for the target to be located in a large segment of the forward sky. To provide such access obviously requires the vehicle nose to be folded back as illustrated in Figure 8-14 showing the baseline rendezvous and docking sensor assembly.

The actual gimballing and mounting is clearly subject to change depending on the viewing angle available and based on the recommended simulation, the direction the rendezvous targets would generally appear from. The configuration of the laser illuminator and ranger will depend on what sort and location docking system is selected.

While an antenna dish nearly 8 inches in diameter is shown in the sketch, a millimeter wave radar providing the performance shown only requires a 6-inch diameter dish at 2KW peak power. The radar can double as an altimeter if earth viewing is provided, or if a separate antenna is provided which illuminates the ground in the nominal re-entry attitude of 2° pitch down.

8.3.11 Displays and Controls

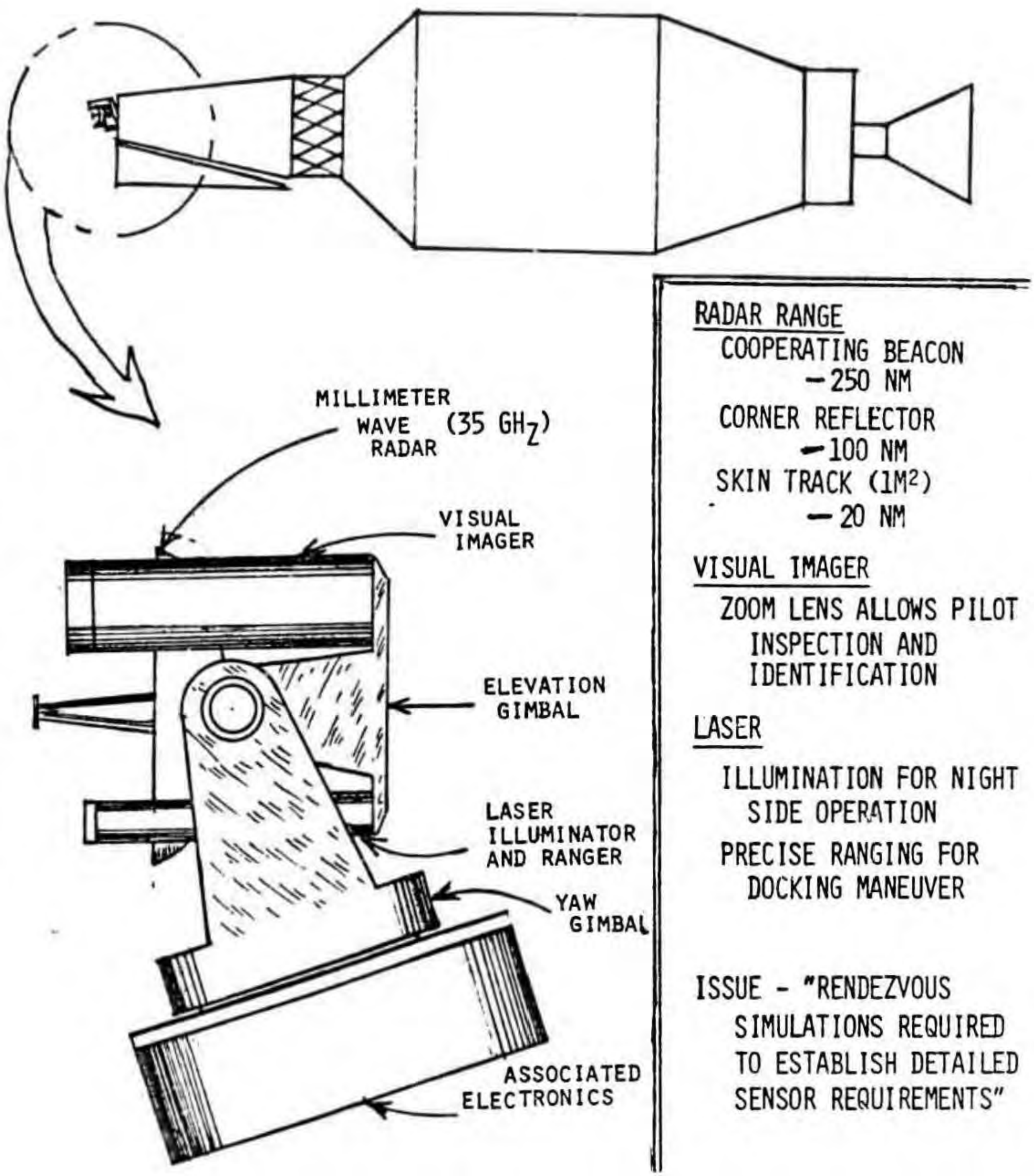
Until the missions are better defined and the pilot man-machine interface is further studied only some broad characteristics of the displays and controls can be ascertained. The fact that the pilot will be fully suited for the full mission duration is a significant factor.

Because of sun glare, and poor viewing attitude when the pilot has his head up through the access hatch, it is essential that at least critical displays be projected inside his helmet. Also because he may have his hands occupied it is very desirable for critical commands to be voice activated. This would allow some degree of display and control during EVA, should that be necessary.

The panel mounted displays should preferably be solid state for high reliability and general flexibility should be provided so that any critical display function could be moved to a secondary display should the primary fail. An interactive, digitally controlled display and control system is anticipated where the desired functions are called up by the pilot using a menu type of system.

It is anticipated that the TV type visual images located on the gimballed rendezvous sensor assembly would be projected when desired both on the display panel area and on the helmet sight, with wide range zoom, angle, and focusing controls that would provide a significant capability for outside viewing when required.

FIGURE 8-14- RENDEZVOUS & DOCKING SENSOR ASSEMBLY



8.3.12 Processors and Input/Output

Of all the overall requirements, the need for fail operational capability has the most significant implications. Triply redundant computation is required with voting between the three computer outputs. Each computer also needs to receive inputs from redundant sensors so as to be able to discriminate against failed inputs. The degree of cross-strapping involved is illustrated in Figure 8-15 which shows just the interfaces with the laser IMU's and the celestial sensor. The same degree of cross-strapping would be involved with all the other time critical subsystems where fail operations is required.

Lack of resources did not allow a trade-off to be performed in the interface/input-output area specifically for spaceplane. The need for the same degree of redundance, however, has been recently studied on other programs and the attractiveness of a standard bus system such as MIL-STD-1553B must be evaluated. With the maturing of microprocessors over the last several years the computing functions can best be performed by a microprocessor. If required, a 32 bit, floating point microprocessor capable of being programmed in a higher order language (HOL) should be available, although possibly it would be an overkill for spaceplane. The Air Force seems to be converging on a standard architecture for its future computers in operatoinal systems, the MIL-STD-1750A architecture, and during the next phase the advantages of using a 1750A processor needs to be examined.

Also very key to the spaceplane computer system is the computer memory to be used. For flexibility to make changes and still prevent memory mishaps, past experience has shown the need in operational systems, besides the scratch pad memory, for the program memory to be reprogrammable, non-volatile with power interruption, and using a non-destruct read system. The wide ranging operational modes and missions, particularly in the pilot display area, will probably also require some form of bulk memory such as disk drives for bringing in tenant programs.

Pending the future trade-offs, a curent microprocessor, the Tl 9989, has been selected along with Harris RAM's for scratch pad memory, and Honeywell/General Instrument NMOS EAROMS for program memory. These key components are shown in Figure 8-16, and have been assembled into a demonstration computer designed for the control of space vehicles.

Mockups that illustrate the type of packaging typical of the processor and I/O hardware are shown in Figure 8-17.

8.3.13 Electrical Power

Total Electrical Power requirements are broken down in Section 8.5.2 and summarized in Table 8-4.

Note the dramatic difference between the two types of power, high voltage, high peak, low duration power for the aero surface drives, and low voltage, steady drain, long durationpower for the avionics and life support system.

FIGURE 8-15 - TRIPLY REDUNDANT PROCESSOR ILLUSTRATING CROSS-STRAPPED INPUTS AND OUTPUTS

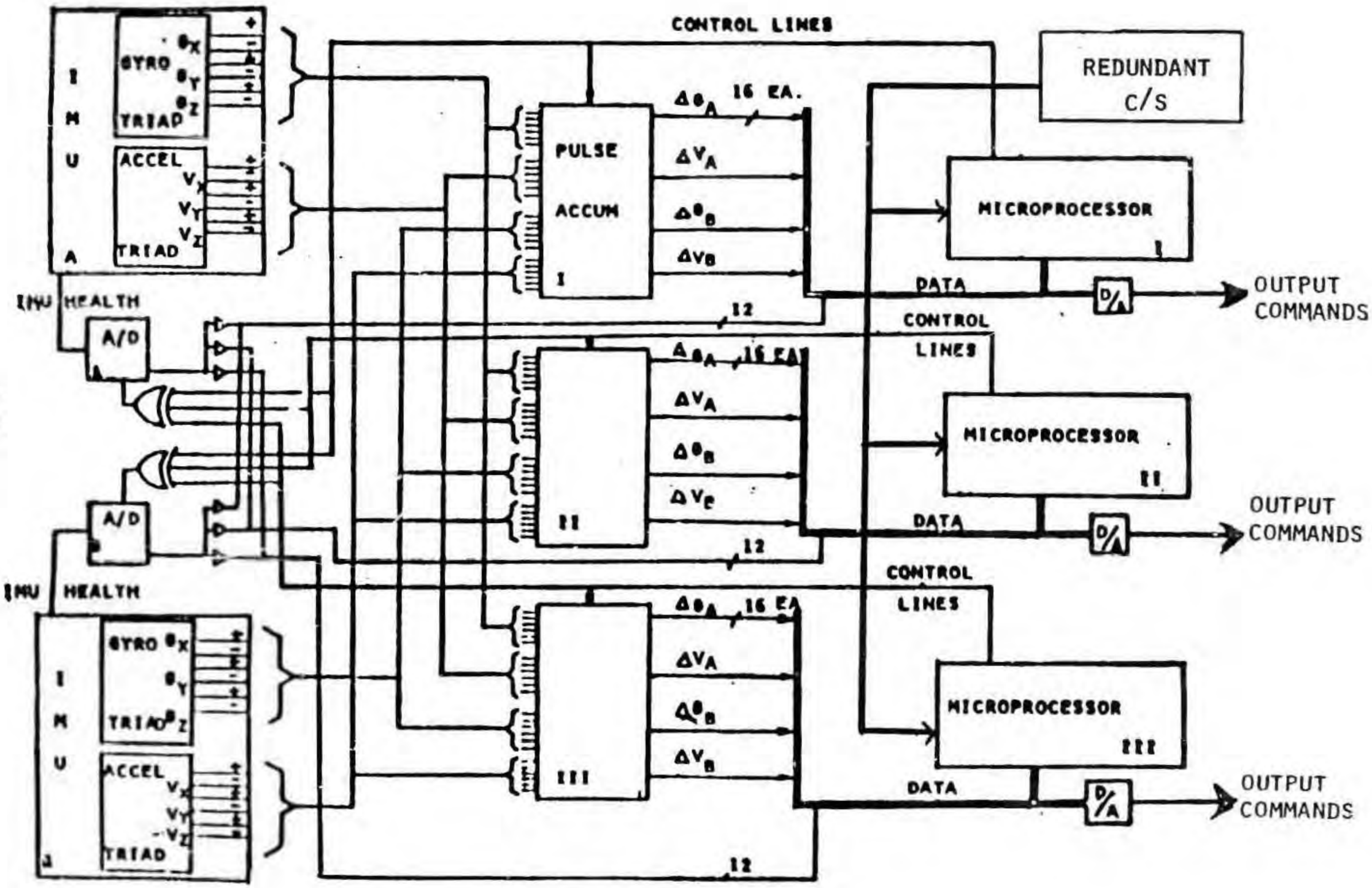
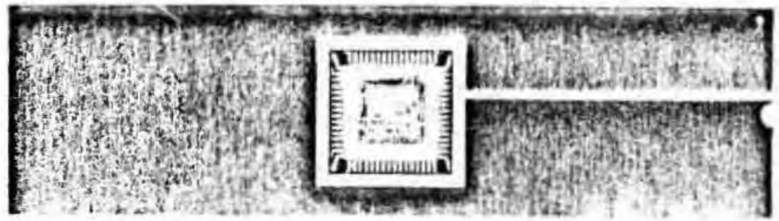


FIGURE 8-16

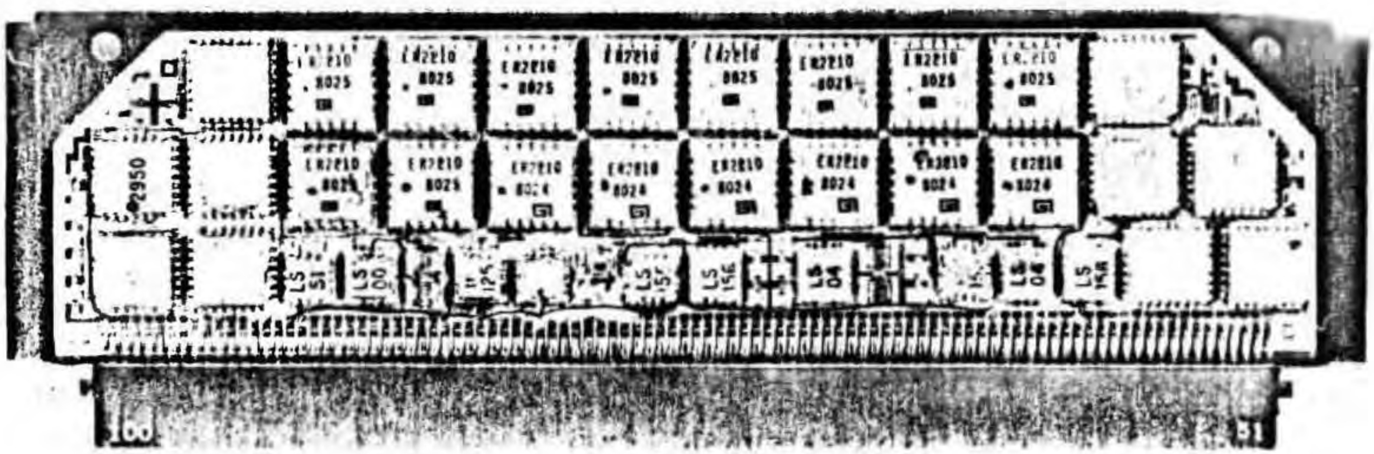
KEY COMPONENTS OF NEW RAD HARD SPACE COMPUTER



TI 9989 MICROPROCESSOR

HARRIS 256 WORD RAM CHIP

16K MNOS EAROM BOARD



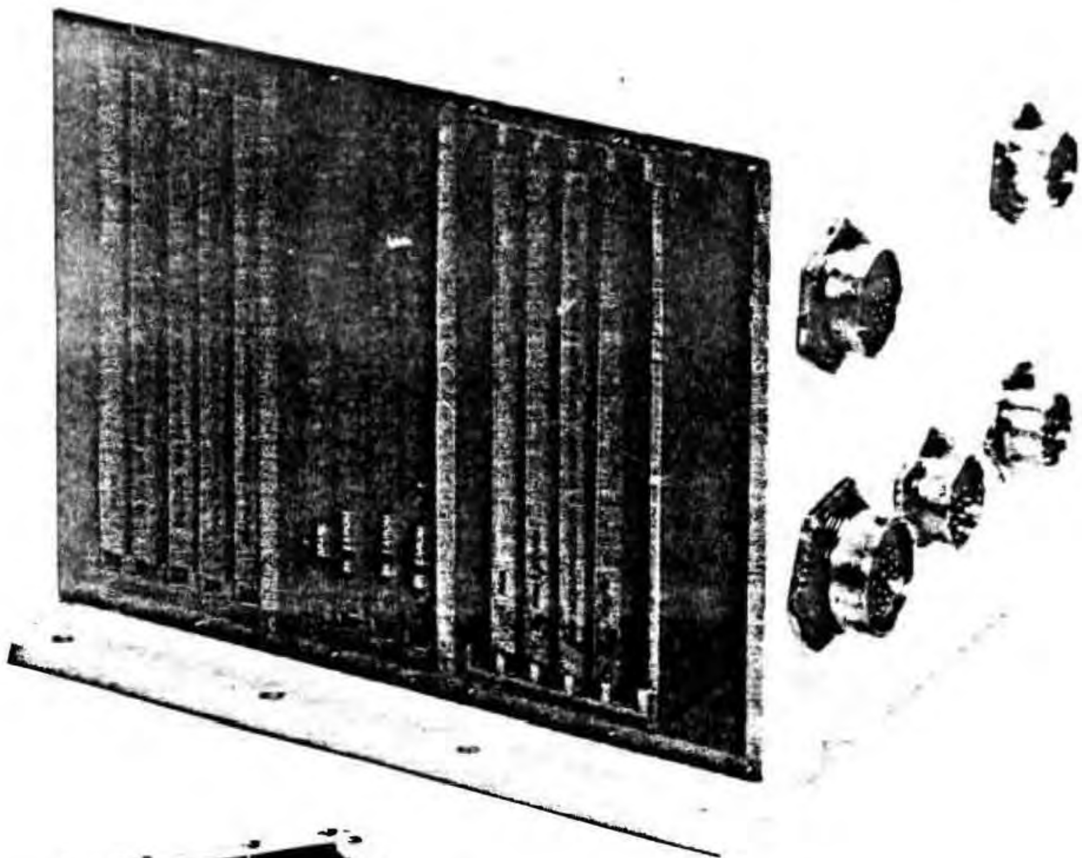


FIGURE 8-17
MOCKUPS ILLUSTRATING PACKAGING TYPICAL OF PROCESSOR AND I/O HARDWARE

Table 8-IV

POWER REQUIREMENTS

Required Characteristics

	Voltage	Average Power-KW	Peak Power-KW	Energy-KW-Hrs
o Avionics	28V	.25 For 24 Hours	.80	6.00
o Life Support	28V	.30 For 24 Hours	.30	7.20
o Aero-Surfaces	270V	2.10 For 1/2 Hour	42.00	1.05

(Peak HP - 27, Average 5%, Drive Efficiency 65%)

- o Fail Operational

Desirable Characteristics of Power System

- o No Additional Tankage
- o Easily Serviced in Space (Fluid Replacements Only)
- o Reasonable Weight and Volume
- o Reserve for Contingencies

These drastic differences will require a combination auxiliary power system meeting both requirements. Fail operational is a requirement for pilot survival. Other desirable characteristics which will shape the configuration selected, are listed in the table also.

In the conflicting opinions about the electrical power system it quickly became apparent that a rigorous, systematic approach would be required to make an objective trade-off between the various candidates. Fortunately, a similar trade-off had already just been finished by the Honeywell Shuttle Flight Control System group looking at upgrading the Shuttle to all-electric drives. This group had converted all the data on power systems they had collected into a very useful form, a plot of power density versus energy density and shown in Figure 8-18. It should be recognized that these are smoothed curves and that some suppliers products may be "better" or "worse" than depicted due to other consideration such as designs for severe environments, long life, ease of maintenance, low cost, etc.

As a reminder it should be noted that at this time Lithium batteries are not considered rechargeable and due to their use of non-water electrolytes have lower power densities than some other power sources. Silver-Zinc batteries can only be considered as semi-rechargeable because of a tricky charge/discharge profile requirement. Practical fuel cells for space utilize LOX and LH_2 the latter not otherwise being required for spaceplane. Some of the systems such as moltan salts require continuous heat inputs which would have to be charged as an energy penalty and limit quick use capability from a dormant state. The I.C. Engine is shown for reference, but is not a practical space candidate since its plot does not account for the weight of O_2 consumed. Also note that the fuel cell curve never gets very high on the chart representing its characteristic of being unable in a reasonable size and weight to handle the peak load demands of the aerosurface drives.

Note on the curves that the weight of a system varies considerably, depending on the time duration of use and power drawn, a fact many vendors are not quick to volunteer. For instance, the Lithium battery provides nearly 400 watt-hours per pound at low current drains such as 6 watts per pound but when pushed to the maximum power drain would weight nearly 10 times as much to provide the same total energy.

With all this information in hand a number of power systems were configured wherein the desired characteristics after acceptable redundancy were to be minimum volume and weight. The best three of these are shown in Figures 8-19, 8-20 and 8-21. For comparison purposes the optimum electric/hydraulic system is shown in Figure 8-22. To be on the same basis actuators weights and volumes are included in the totals. For these four systems there was not great differentiation on the basis of volume and weight only, although the all battery system had the slight edge on volume. The only one of the four which did not have some other serious drawback however, was the continuous power generation, load leveling system, and it was selected for the baseline. In this selected system the power generation is provided by a redundant turbo-generatory system, continuously outputting both 270V and 28V. Mission time or the number of re-entries only affect the amount of hydrazine consumed, estimated at 6 lb/KW-hr.

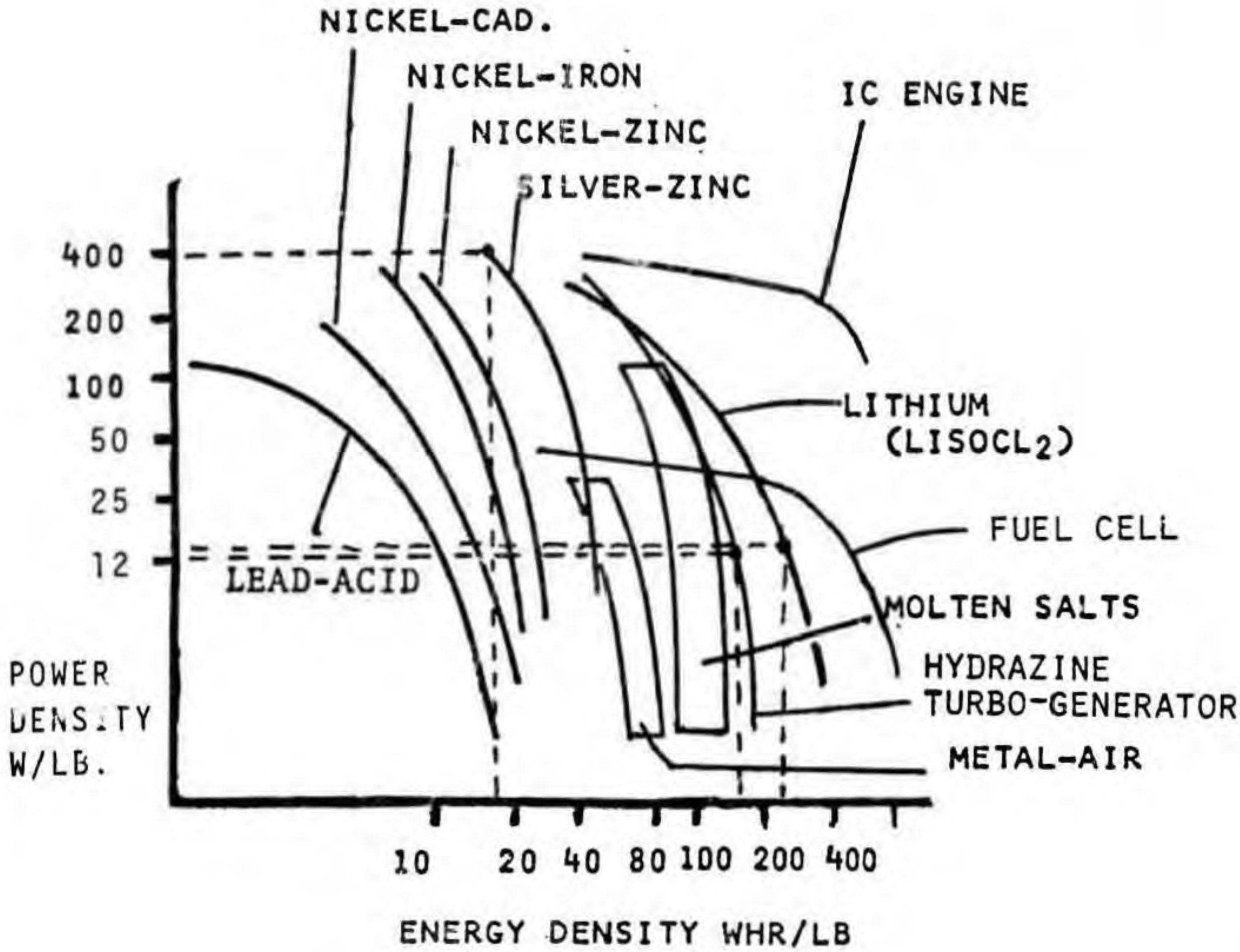
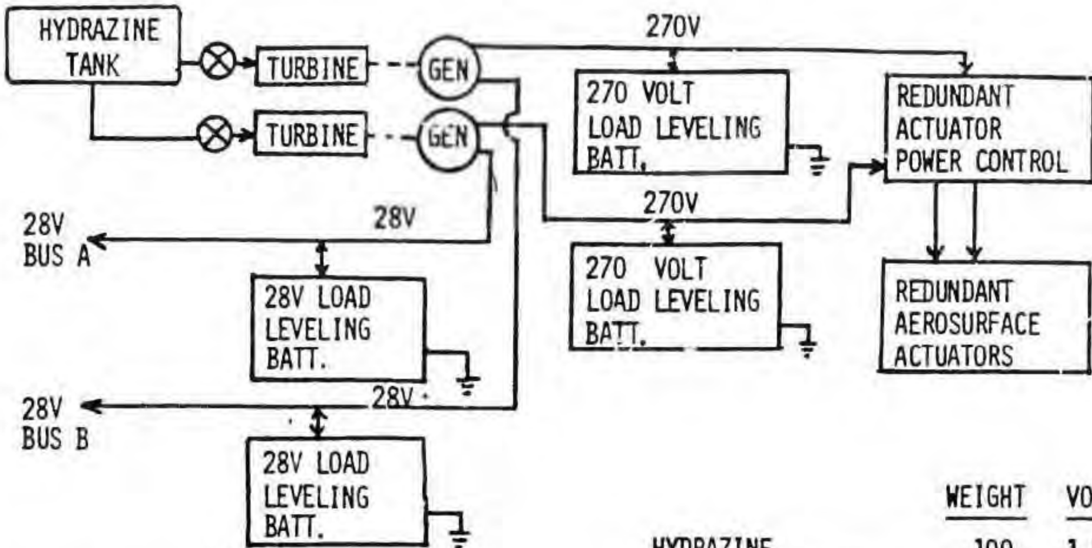


FIGURE 8-18 POWER DENSITY Vs ENERGY DENSITY

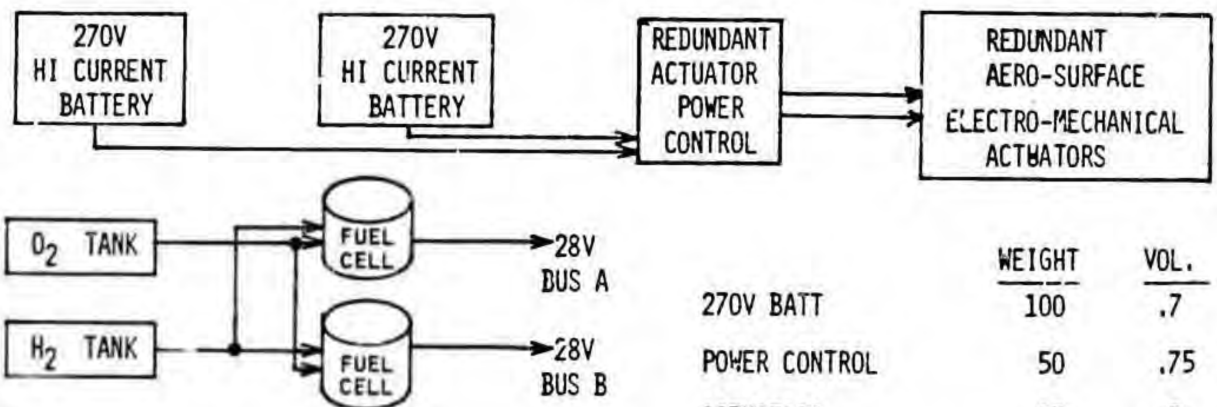


	WEIGHT	VOL.
HYDRAZINE	100	1.5
TURBO-GENERATORS	30	.8
POWER CONTROL	50	.75
ACTUATORS	25	.5
270V BATTERIES	80	.5
28V BATTERIES	15	.16
	300	4.21

COMMENTS

- MISSION TIME OR NUMBER OF ENTRIES EXTENDED BY INCREASING HYDRAZINE CONSUMPTION
- NO LOGISTIC PROBLEMS WITH SPACE BASING
- HI DENSITY COMPONENTS USED

FIGURE 8-19 CONTINUOUS POWER GENERATION WITH LOAD LEVELING- SYSTEM SELECTED



	WEIGHT	VOL.
270V BATT	100	.7
POWER CONTROL	50	.75
ACTUATORS	25	.5
FUEL CELL	150	2.5
H_2 TANK	10	.1
	335	4.05

COMMENTS

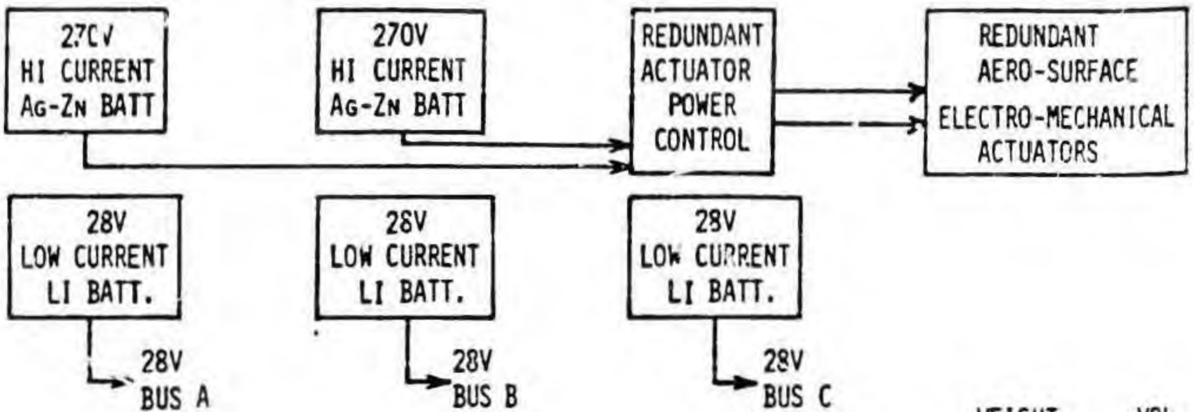
MULTIPLE ENTRY WOULD REQUIRE MORE BATTERY CAPACITY

LH_2 POSES LOGISTICS PROBLEMS AND ADDS ANOTHER EXPENDABLE

FUEL CELLS ADD SIGNIFICANT VOLUME

PEAK POWER REDUCED WITH SINGLE BATTERY LOSS

FIGURE 8-20 - FUEL CELL/BATTERY COMBINATIO AUXILIARY POWER SYSTEM



COMMENTS

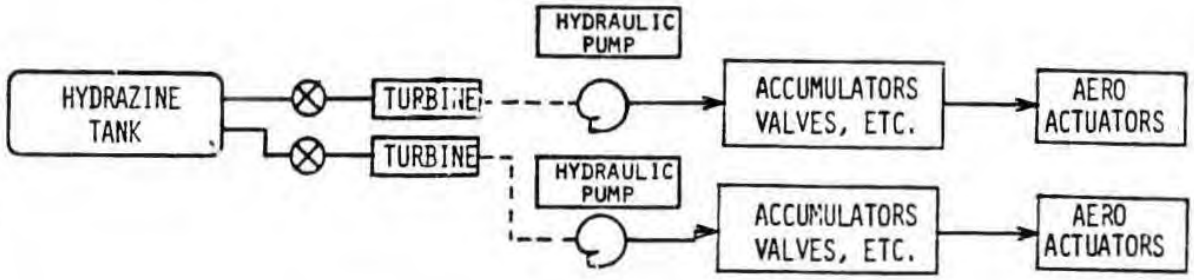
PRIMARY LI BATTERIES WOULD PREVENT SPACE BASING

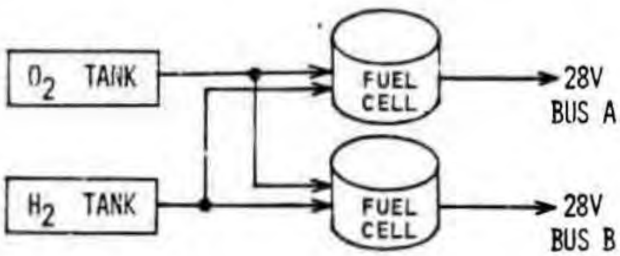
MULTIPLE ENTRY WOULD REQUIRE MORE BATTERY CAPACITY

PEAK POWER REDUCED WITH SINGLE HI VOLT BATTERY LOSS

	WEIGHT	VOL.
270V HI CURRENT BATTERIES	100	.7
28V LOW CURRENT BATTERIES	120	1.0
POWER CONTROL	50	.75
EM ACTUATORS	25	.5
	295	2.95

FIGURE 8-21 - ALL BATTERY AUXILIARY POWER SYSTEM





	WEIGHT	VOL.
HYDRAZINE	25	.3
PUMPS, TURBINE, ACCUM, VALVES	85	2.0
VALVE DRIVES	10	.15
AERO ACTUATORS	25	.5
FUEL CELLS	150	2.5
H_2 TANK	10	.1
	305	5.55

COMMENTS

HYDRAULICS FLUID LEAKS

LH_2 POSES LOGISTICS PROBLEMS AND ADDS ANOTHER EXPENDABLE

FUEL CELLS ADD SIGNIFICANT VOLUME

FIGURE 8-22 - HYDRAULIC ACTUATOR/AUXILIARY POWER SYSTEM
(SHOWN FOR REF.ONLY)

Batteries are used for load leveling. Very small batteries on the 28V buses help suppress switching transients and hold up the power long enough to make an emergency re-entry from low earth orbit in the event of both turbo-generators failing. Also, the high power density of the 270V supply insures a battery of sufficient energy density to be able to support aerosurface needs during such a re-entry. Silver Zinc batteries would be today's choice; however, in the future, some battery weight might be saved by using Silver Hydrogen.

Note this system is reasonably dense, is weight competitive, has growth potential, high reliability, and is compatible with Space basing. It also uses the same fuel as is contemplated for the Spaceplane rockets and reaction control jets, thus being able to share tankage.

The all battery system would have been prohibitively heavy without using lithium batteries for the 28V buses and hence it is disqualified for space basing, since it is not rechargeable. The fuel cell system requires the use of liquid H_2, not otherwise required on the spaceplane, and a difficult expendable to make available in space. In the future fuel cells may be available using hydro carbons in place of liquid H_2 which might be more attractive.

The electric/hydraulic system shown for reference is primarily rejected because it violates the desire for all electric vehicle to avoid the severe contamination problems encountered with hydraulic systems.

8.4 PERFORMANCE ANALYSES

A limited number of analyses were performed in support of the trade-off studies and to help in accessing the capabilities of the system baselined. While the analyses were predominately in areas concerned with navigation, several other analyses were performed.

8.4.1 Navigation Performance to Geosynchronous Earth Orbit (GEO)

While normal peacetime navigation accuracy can be expected to be as good as that provided by GPS, it is of importance to know the expected error to GEO, the most severe case considered when required to use pure inertial navigation. If the SANS concept is implemented as recommended in the baseline, autonomous position errors should be bounded around one or two nautical miles, depending largely on the number and time separations of celestial observations made. Pure inertial navigation performance to GEO is shown in Table 8-5.

8.4.1.1 Shuttle Deployed

The shuttle deployed bi-elliptic encounter transfer to GEO was particularly analyzed. The conditions for the bi-elliptic encounter transfer are given in Figure 8-23. The breakdown of error sources and their contribution to navigation errors in in-track, cross-track and radial directions is given in Table 8-6. The clear lesson is that the poor shuttle altitude handoff is unacceptable and that one and preferably two attitude updates should be performed to insure thrusting is precisely along the desired direction.

8.4.1.2 Air Launched from Aircraft Mothership

Pure inertial navigation performance for an air launch from an aircraft mothership is shown in Figures 8-24 and 8-25. The spaceplance inertial system is aligned by velocity matching during flight prior to separation from the mothership. Two mothership navigation type systems were assumed, giving significantly different results. Even with a precision mothership navigator, performance results are poorer than would be considered acceptable for most missions. Normally both position and attitude updates would be performed during parking and orbit transfer modes to reduce the pure inertial errors to acceptable levels.

8.4.1.3 Ground Launched

While a specific ground launched pure inertial performance analysis for spaceplane was not made, the performance to GEO given in Table 8-5 is that expected, based on the experience with Atlas Centaur for which Honeywell is the inertial equipment supplied. Assuming an azimuth optical alignment as is currently done on the Atlas Centaur, the performance of 10 N.M. and 8 ft./sec. can reasonably be expected.

8.4.2 Stellar Autonomous Navigation System (SANS)

It is well known that if a space vehicle accurately knows its celestial attitude, its Earth local vertical attitude, and precise time at

Table 8-V

SUMMARY OF PURE INERTIAL NAVIGATION PERFORMANCE TO GEO

		36 SPERICAL ERRORS AT APOGEE*		
Condition		Pos Error N.M.	Vel Error Ft/Scc	Dominant Remaining Error Source
1.	Ground launched by expendible launch vehicle.	10	8	IMU Instrument Errors
2.	Shuttle deployed with Celestial Attitude update before both perigree and apogee burns.	35	27	Shuttle Pos and Vel Handoff Errors
3.	Mother aircraft air launched with on C/S update.	70	35	Transfer Alignment From Mother Aircraft

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* Performance for all cases would be significantly improved with GPS or SANS updates and trim maneuvers during transfer orbits.

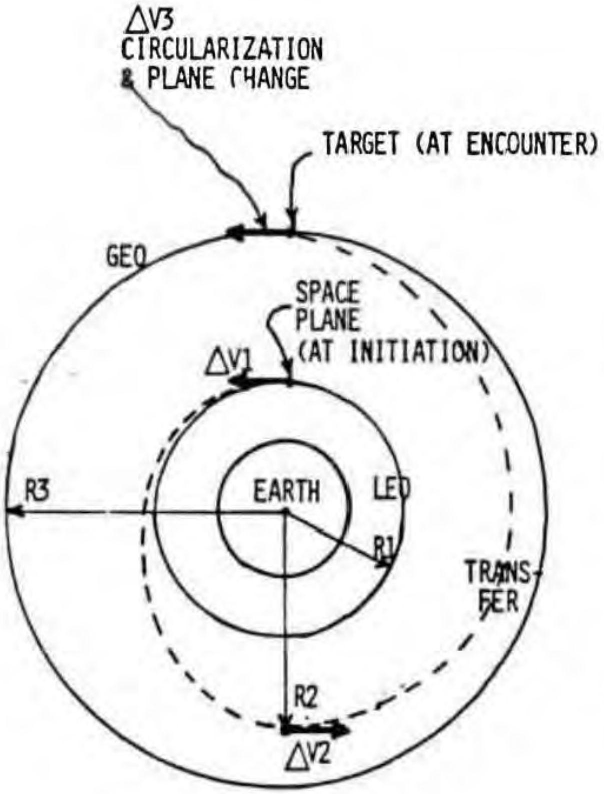


FIGURE 8-23 BI-ELLIPTIC ENCOUNTER TRANSFER TO GEO

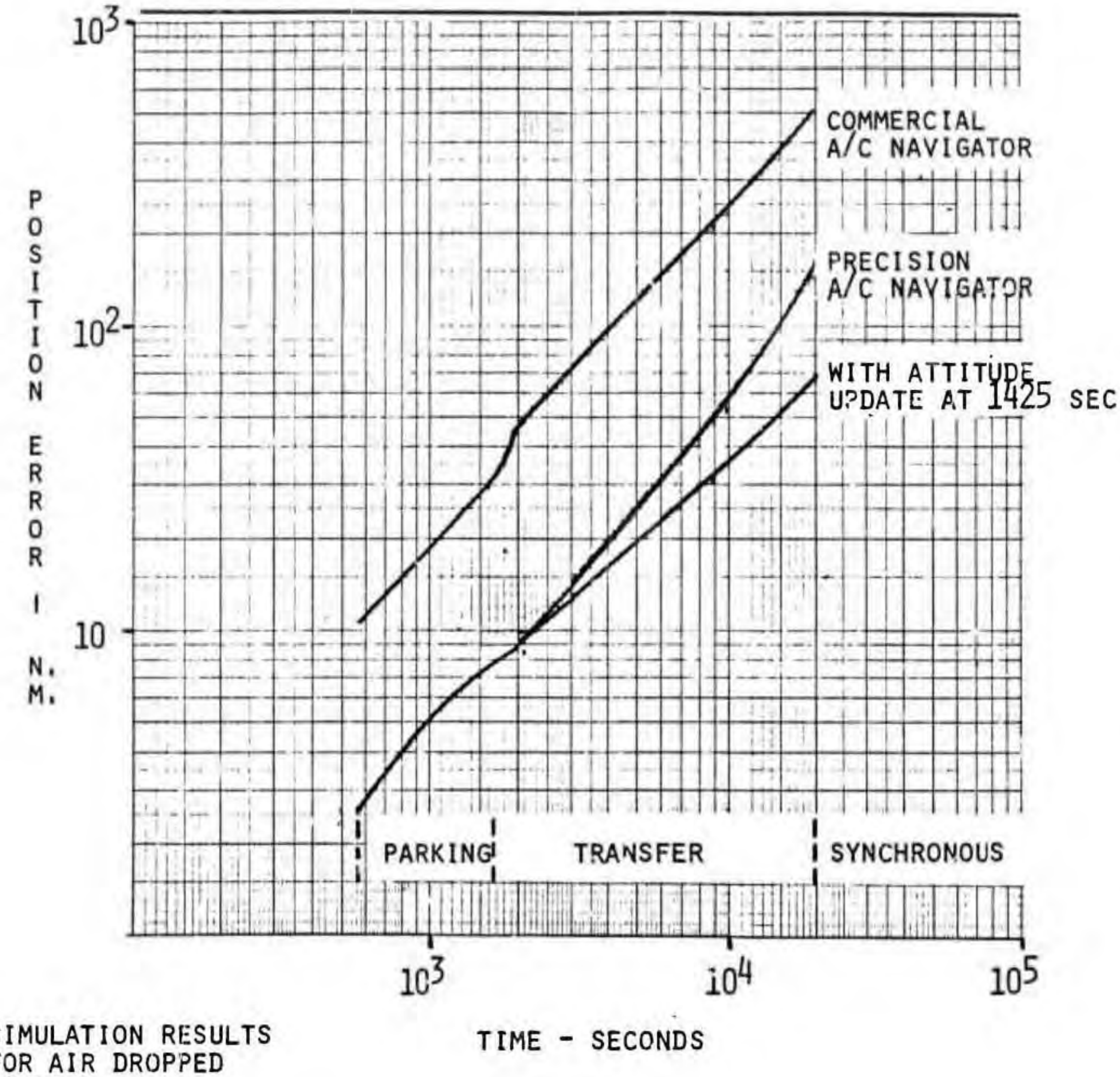
$\Delta V1$	6672.6 FT/SEC
$\Delta V2$	1344.1 FT/SEC
*$\Delta V3$ (CIRC.)	1198.7 FT/SEC
(PLANE CHG. 28.5 DEG)	4961.8 FT/SEC
T_{1-2}	3 HOURS
T_{2-3}	6.177 HOURS
R1	200 NMI
R2	14468.5 NMI
R3	22808 NMI

*WHEN ΔV_3 CIRCULARIZATION AND PLANE CHANGE ARE DONE TOGETHER ΔV_3 = 5104.5 FT/SEC

Table 8-VI

PURE INERTIAL
NAVIGATION ERRORS AT ENCOUNTER

Error Source	Error Magnitude	Position Error, NMI In Track	Position Error, NMI Cross Track	Position Error, NMI Radial	Velocity Error, Ft/Sec In Track	Velocity Error, Ft/Sec Cross Track	Velocity Error, Ft/Sec Radial
Laser Gyro Random Walk	0.02 Deg/Hr	4.3	-	-	5.5	2.1	6.1
Laser Gyro Bias	0.05 Deg/Hr	18.7	-	-	38.6	14.6	41.6
IMU Alignment	0.2 Millirad	1.6	-	-	1.0	0.4	1.1
Accel. Bias	5 G	0.2	-	0.4	0.2	0.01	0.2
Accel. Scale	45 PPM	3.9	-	7.8	4.3	0.2	3.5
Accel. Alignment	0.2	1.6	-	-	1.0	0.4	1.1
RSS Sensor Errors		19.7	-	7.8	39.3	14.8	42.2
Initial Attitude Error From Shuttle Handoff	1 Deg	139.6	-	-	87.3	34.9	96.0
RSS Error with one Attitude Update Prior to Burn 1	45 ARCSEC Update Accuracy	19.8	-	7.8	39.3	14.8	42.2
RSS Error with Attitude Updates Prior to Burn 1 and Burn 2	45 ARCSEC Update Accuracy	4.8	-	7.8	4.6	0.7	3.8



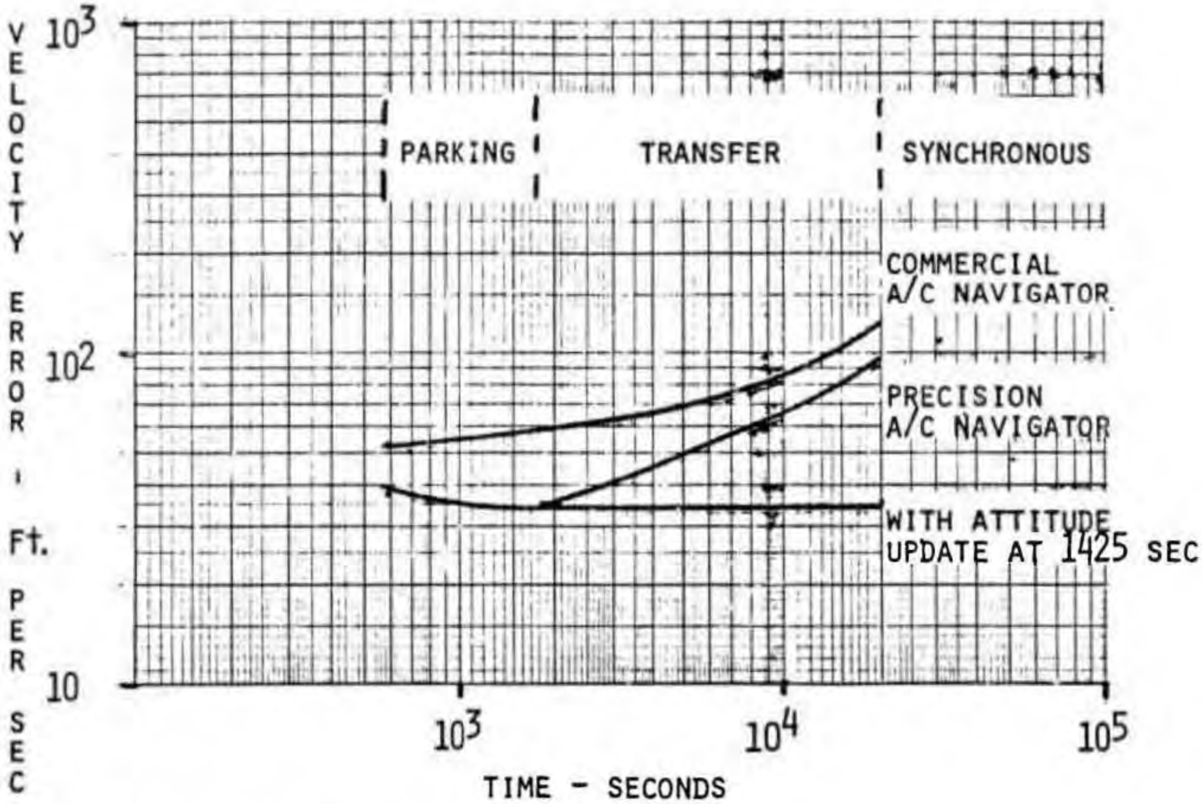
SIMULATION RESULTS FOR AIR DROPPED LAUNCH.

SPACE-PLANE NAVIGATOR ALIGNED BY VELOCITY MATCHING TO NAVIGATOR OF MOTHER AIRCRAFT.

TWO MOTHER AIRCRAFT TYPE NAVIGATORS COMPARED - 747 COMMERCIAL AND PRECISION MILITARY ESG.

FIGURE 8-24

PERFORMANCE WITH AIR DROP POSITION ERROR - 3σ



SIMULATION RESULTS FOR AIR DROPPED LAUNCH.

NAVIGATOR ALIGNED BY VELOCITY MATCHING TO NAVIGATOR OF MOTHER AIRCRAFT.

TWO MOTHER A/C TYPE NAVIGATORS COMPARED -747 COMMERCIAL AND PRECISION MILITARY ESG.

FIGURE 8-25

PERFORMANCE WITH AIR DROP

VELOCITY ERROR - 3σ

a number of points on an orbit, position and velocity can be autonomously determined. Using the celestial sensor and the horizon sensors, the capability is in fact planned as a back-up navigation mode.

Greater precisions in autonomous navigation of this type can be achieved if the stellar sensor is capable of accurately tracking stars as their light posses through the atmosphere. A system based on this concept, named SANS, is proposed as part of the spaceplane avionics baseline, and its operation is illustrated in Figure 8-26. As a star sets or rises, its apparent position is shifted by atmospheric refraction, which effect can be used for indirectly determining the Earth's location. A sample star setting as measured by the HEAO-2 is shown in Figure 8-27. If precise atmospheric models are used, the ultimate potential of SANS shows promise of autonomous navigation, even with a 1 arc second star tracker, of around one-tenth of a mile. Such accuracy would not likely be achieved in a typical spaceplane mission because to get the levels of accuracy shown in Figure 8-28 requires smoothing and averaging over several orbits. Even for a partial orbit, performance of a mile or so achieved autonomously should be regarded as quite acceptable.

8.4.3 Re-Entry Navigation

Re-Entry navigation performance was not investigated in detail because re-entry corrider and window tolerances had not been established. However, in preparation for such an investigation, re-entry error sensitivities were investigated. Re-entry from GEO geometry and associated orbital parameters are illustrated in Figure 8-29.

A significant finding from this re-entry analysis is the critical sensitivity of the perigee altitude to the velocity error after de-orbit burn. It is unlikely that the velocity error can be held to one foot per second although it is likely the re-entry altitude tolerance derived will be around ± 10,000 feet. Therefore, position updates with associated trim burns can be expected to be performed during the transfer orbit. Considerable study needs to be done in this area in validating backup navigation concepts as suitable for re-entry since they would have larger associated initial velocity errors. A likely scenario is to fly an acceleration profile after initial re-entry which would not depend on knowledge of the actual altitude. Wind tunnel tests for varying altitudes should provide the data needed to generate stored acceleration Vs re-entry time profiles which would yield safe re-entries. The steering law utilized would be simply to pitch up if the longitudinal acceleration were greater than nominal and pitch down if less than nominal.

8.4.4 Synergistic Plane Change

While it was intended to evaluate the navigation accuracy needed while dipping in and out of the upper atmosphere in order to use aerodynamic lift to make plane changes, lack of aerodynamic data precluded this. Several observations are useful guides.

In the ideal limit, as a first approximation, the ΔV achieved in a plane change is equal to the L/D time the velocity loss. In the practical case additional velocity loss due to nonlifting drag entering and departing

SANS
AUTONOMOUS NAVIGATION CONCEPT:

MEASURE ATMOSPHERE'S REFRACTION OF STARLIGHT AS A REPLACEMENT FOR DIRECT EARTH SENSING

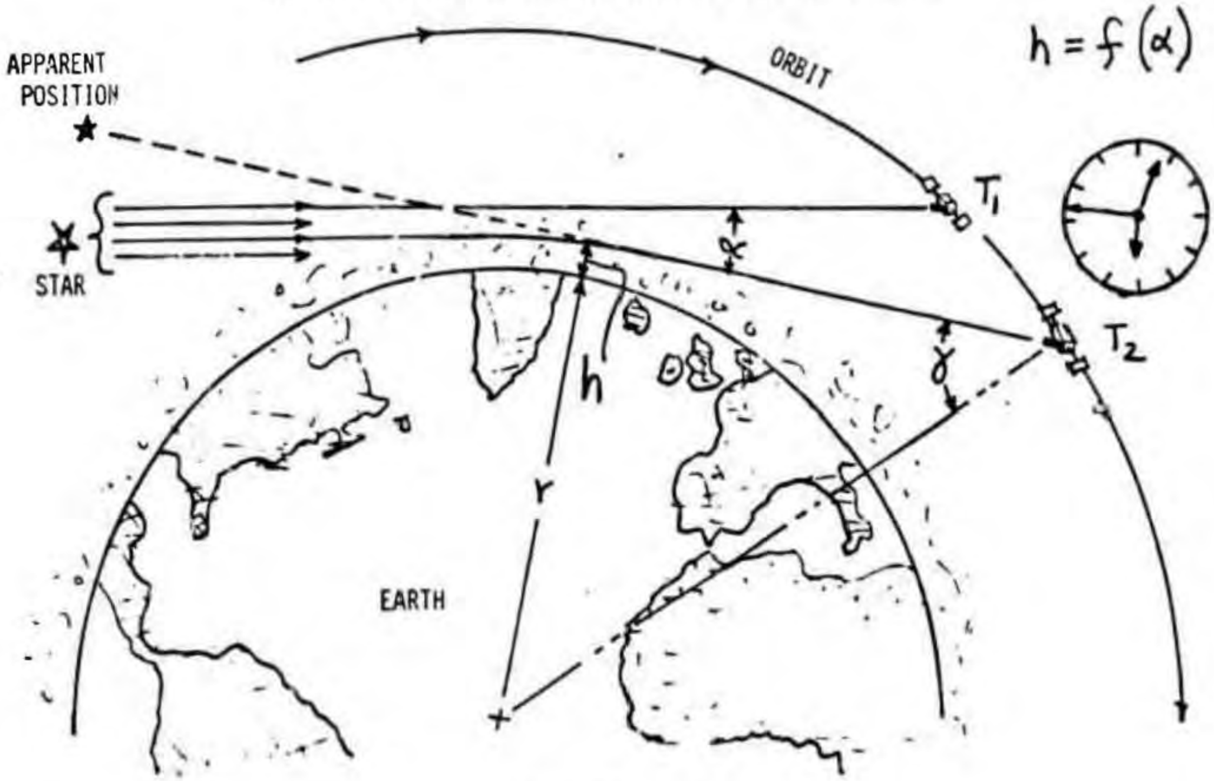
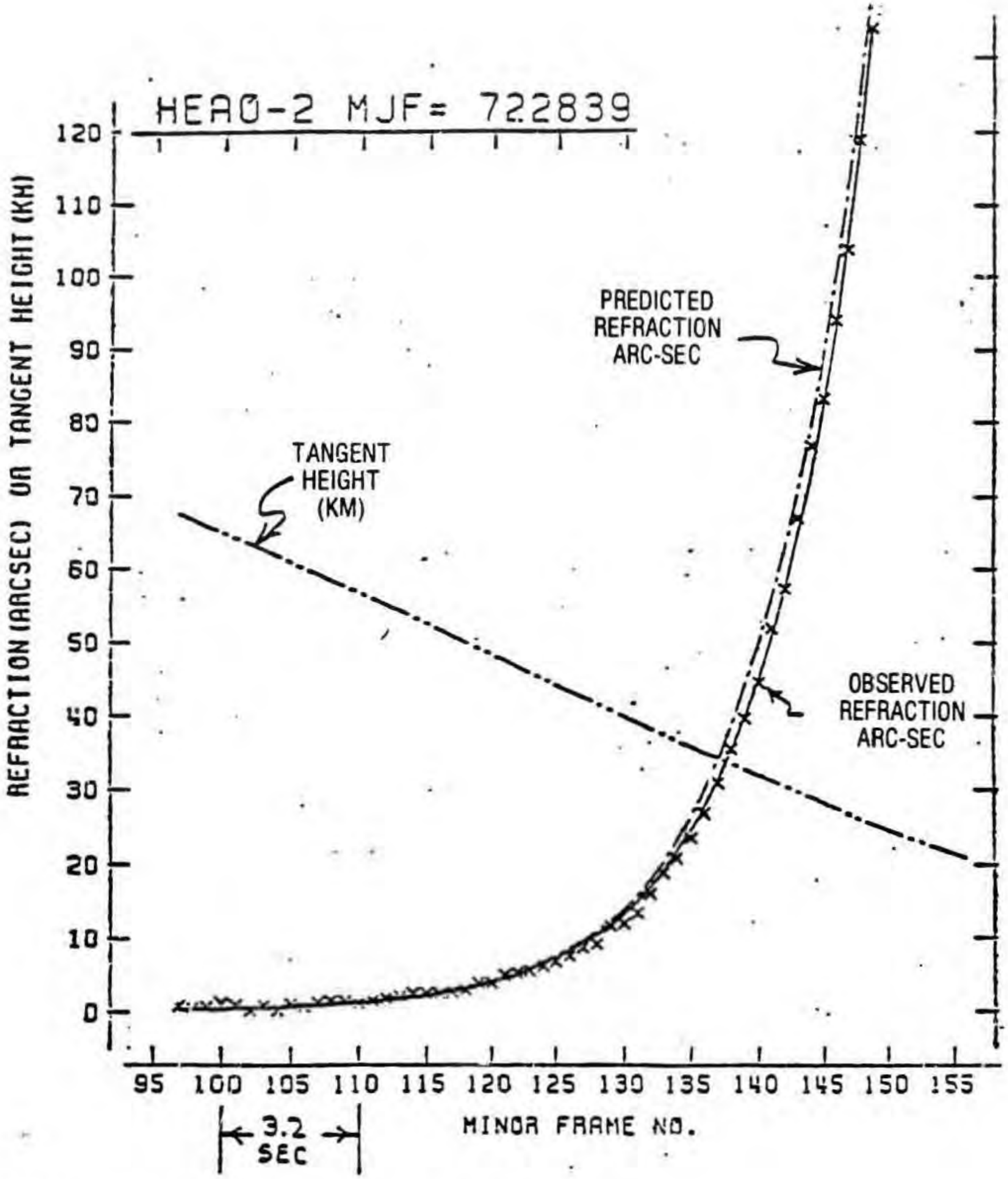


FIGURE 8-26



DATE : 10/21/79
GMT : 12: 5:50.33
STAR A MAGN : 5.26
LATITUDE : 17.05 S
LONGITUDE : 166.51 E
SIGMA : 0.38 ARCSEC
CHI-SQUARED : 6.32

FIGURE 8-27
TYPICAL STAR REFRACTION CAUSED BY PASSING THROUGH THE EARTH's ATMOSPHERE

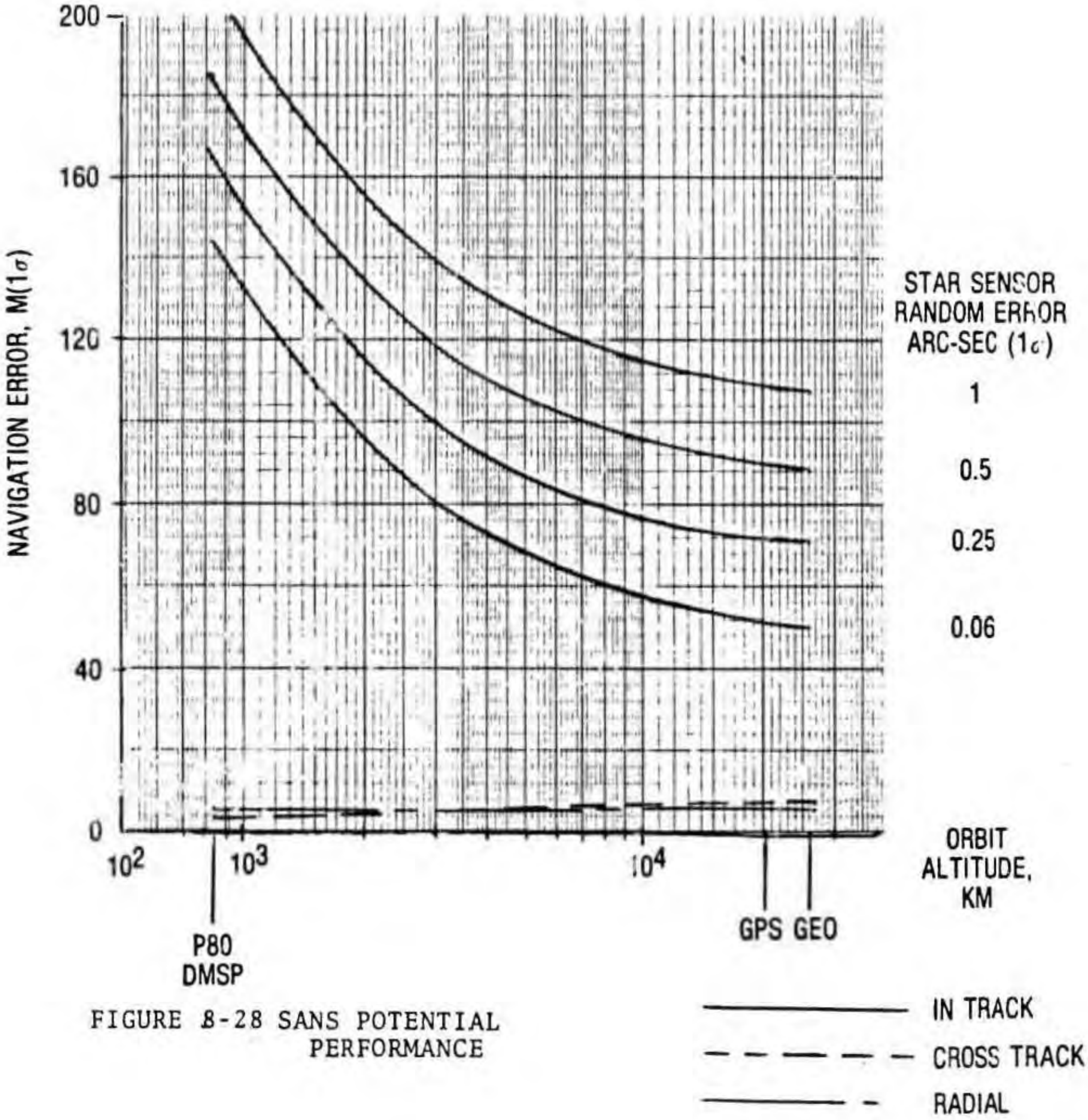
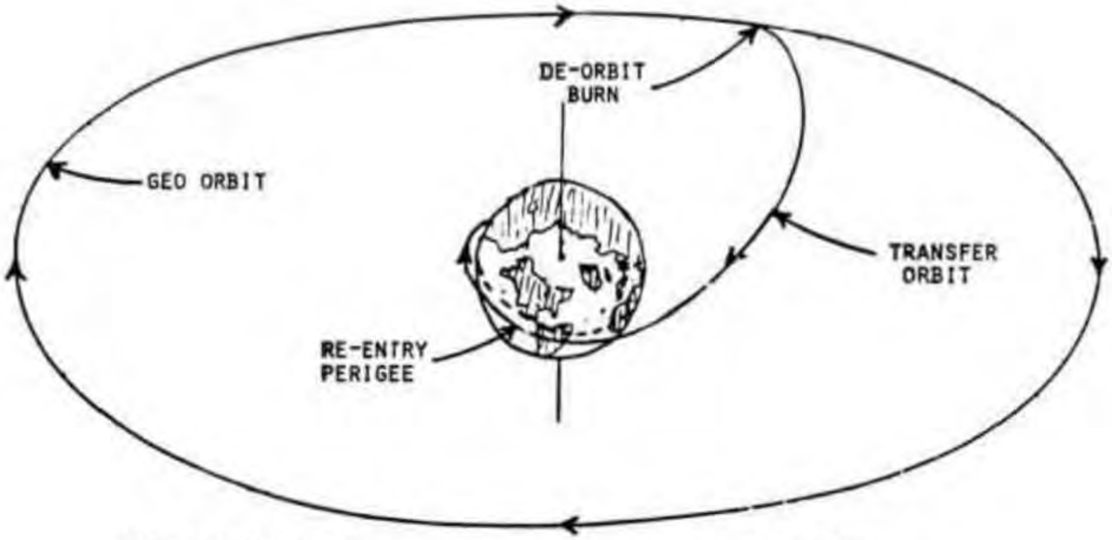


FIGURE 8-28 SANS POTENTIAL PERFORMANCE



TRANSFER ORBIT TIME	--	5.25 HRS
VELOCITY BEFORE DE-ORBIT BURN	--	10,078 FPS
VELOCITY AFTER DE-ORBIT BURN	--	5,185 FPS
RE-ENTRY VELOCITY	--	33,998 FPS
LIFT REQUIRED TO STAY IN ATMOSPHERE	--	1.7G
RE-ENTRY PERIGEE ALTITUDE SENSITIVITY TO VELOCITY ERROR AFTER DE-ORBIT BURN	--	9,396 FT/FPS

FIGURE 8-29 - RE-ENTRY GEOMETRY FROM GEO AND ASSOCIATED ORBITAL PARAMETERS the atmosphere prior to and after the plane change must be considered. It would be expected that at the higher altitudes, drag would be encountered where effective lift would not exist. Therefore, evaluating the synergistic plane change and what best navigation strategy to employ with it requires simulation with realistic parameters.

8.4.5 Aero Surface Power

Since the power used by the aerosurface during re-entry strongly effects the Spaceplane power requirements and they in turn, many of the other Spaceplane systems including weight and V performance, it was essential to evaluate the aerosurface power needs even though accurate hinge moment data and angular rates were not available.

Based on tentative maximum hinge moment torques of 1500 ft-lbs and possible aerosurface angular rates of 150°/sec, a total peak power reflected back to the power source, after allowing for 65 percent motor efficiency, was calculated at 42.KW, assuming all surfaces were driven at once. This established the power density requirements for the aerosurface drive power source which was selected at 270 VDC, a standard voltage.

Based on knowledge of similar systems it was obvious that the average power would be significantly lower. Again, lacking power time history data for spaceplane, it was assumed that the shuttle power time history might be reasonably close, and so it was used. The inboard elevon was chosen as representative, at least as to ratios of peak power to average power, and this data is shown in Figure 8-30 for a nominal re-entry. This time history gives hinge moment loads, aerosurface angular rates; and instantaneous horsepower. Note that the horsepower trace is mostly around zero.

This horsepower time history was analyzed for the full re-entry period as to percentage time at various horsepower levels, which data is presented in the chart of Figure 8-31. For a 25 H.P. drive it can be seen that only two percent of the time did the horsepower exceed one horsepower.

Scaled down to spaceplane needs, this would indicate a need for only 2.0 KW for a 1/2 hour period, or 1.00 KW-hrs total. Because of the uncertainties in the assumptions of the Spaceplane hinge moment and angular rates, a safety factor of 3 was used, sizing the aerosurface energy requirements at 2.1 KW-hrs.

No additional energy was determined to be required for a synergistic plane change and subsequent re-entry because the power system selected has the capability to recharge the aerosurface batteries between the synergistic plane change maneuver and the final re-entry.

8.4.6 Rendezvous and Altimeter Radar Power

Calculations showed that the rendezvous radar required 2 KW peak power at 25db margin in order to detect a $1M^2$ target in an unaided mode at 20 N.M. A coherent, low pulse rate millimeter wave radar was assumed with a dish size of 6 inches.

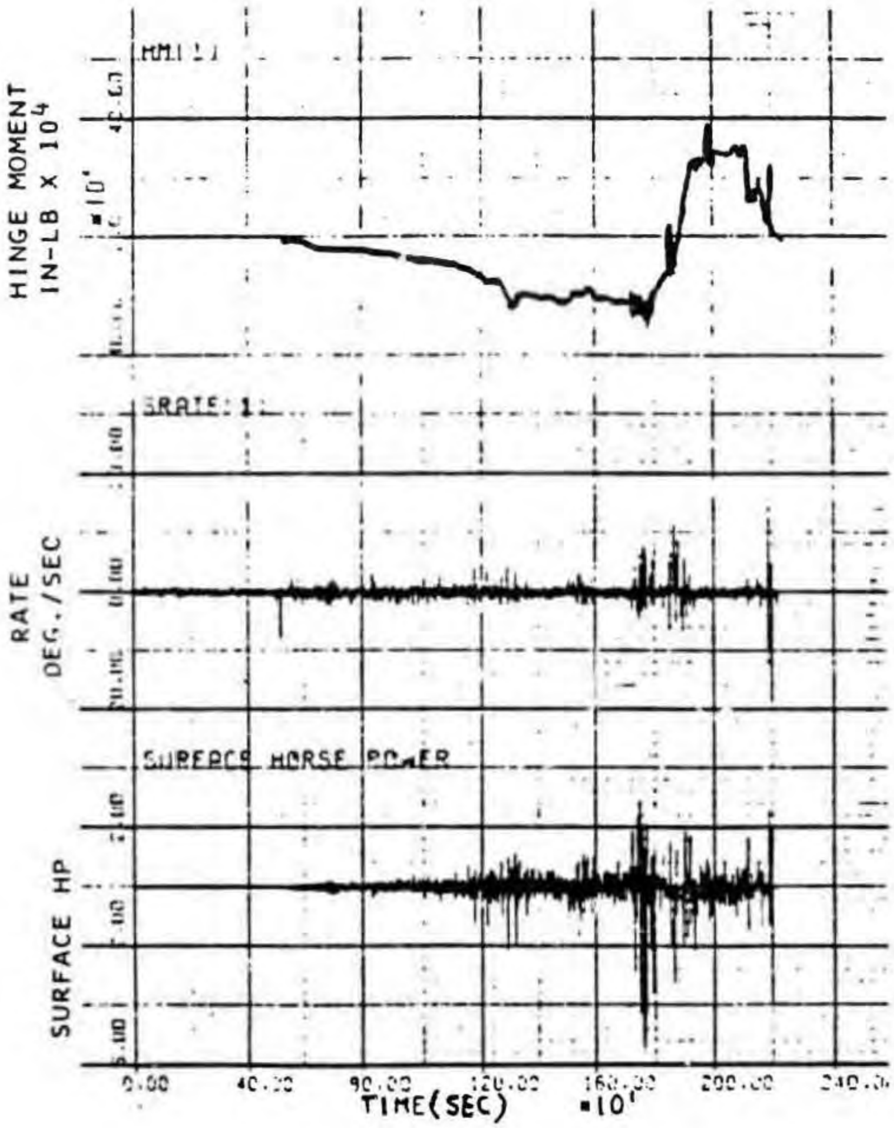


FIGURE 8-30 - SHUTTLE INBOARD ELEVON HINGE MOMENT, ANGULAR RATE, AND HORSEPOWER DURING A NOMINAL RE-ENTRY.

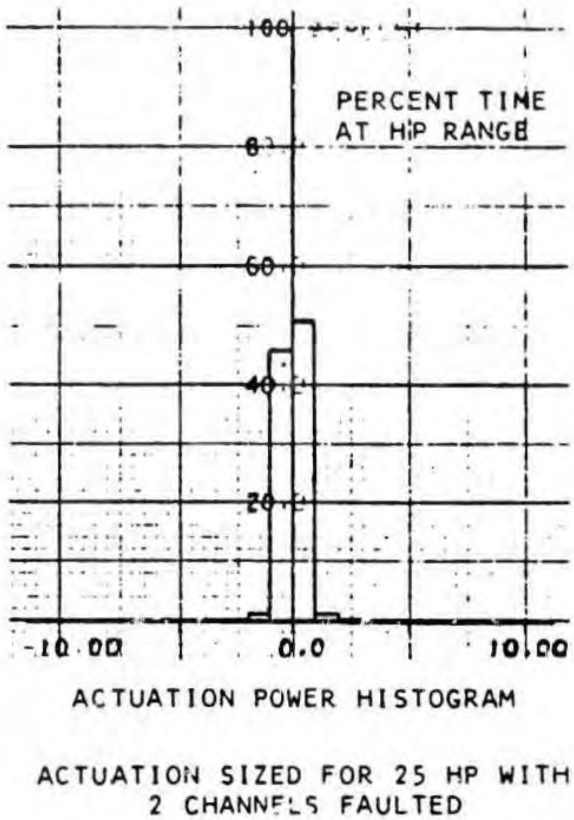


FIGURE 8-31
SHUTTLE INBOARD ELEVON PERCENTAGE TIME AT VARIOUS HORSEPOWER LEVELS

In the radar altimeter mode, allowing for 20 db of atmospheric attenuation and 10 db for ground reflection at 100 miles (assumes beam pointed straight down), the same power of 2 KW peak was required with a 25 db margin. Considerable amount of this margin or a reduced altitude range could result if an auxiliary antenna were used during re-entry approach that had less gain and wasn't pointed straight down due to limitation of siting in the aft area of the spaceplane.

8.5 AVIONICS SYSTEM PHYSICAL CHARACTERISTICS

Weight, size, power, and form factor data were generated for each component of the Avionics System for use in vehicle layouts by Hamilton Standard. Also any special installation considerations such as viewing were identified. These are covered in detail in the vehicle layout section, but summarized here for reference use.

8.5.1 Special Installation Requirements

A few of the avionics subsystems as noted below require particular attention regarding their integration into the vehicle.

8.5.1.1 IMU/Stellar Tracker

The two IMU's and the stellar tracker need to be rigidly mounted together and are so shown in Figure 8-32 representing one layout arrangement. The mounting planes of the two IMU's are approximately 70^{o} apart to provide the needed skew angle between them. The stellar tracker shown is the Ball Bros. tracker that replaces the originally baselined slit detector sensor. An access door along the field of view of the stellar tracker is required to provide star viewing.

8.5.1.2 Gimballed Rendezvous Sensors

Also shown in Figure 8-32 are the gimballed rendezvous sensors positioned so that with the nose retracted and folded back that as much of the forward hemosphere as possible can be viewed without any interferences.

8.5.1.3 Horizon Sensors

The two horizon sensors which are approximately 2 inches in diameter need an unobstructed view downward during the re-entry mode. Since viewing through the vehicle thermal protection system is not practical they may need to be mounted on a retractable arm in the rear are of the vehicle.

8.5.1.4 Momentum Wheels

The only restriction on the momentum wheels is that each should be mounted with its torque axis along one of the principal vehicles axes, roll, pitch, and yaw.

8.5.1.5 Antennas

The TTC and GPS antennas are probably of the whip type and could be mounted in the rear area for protection during re-entry. Retraction may be required when attached to a lower stage vehicle in order to stay within the allowed spaceplane envelope.

The antenna for the millimeter radar in the altimeter mode when the nose is in place would probably have to be mounted in the rear, and likely should be fastened to the same retractable arm as the horizon sensors. This antenna would be a small dish for direction gain, but would

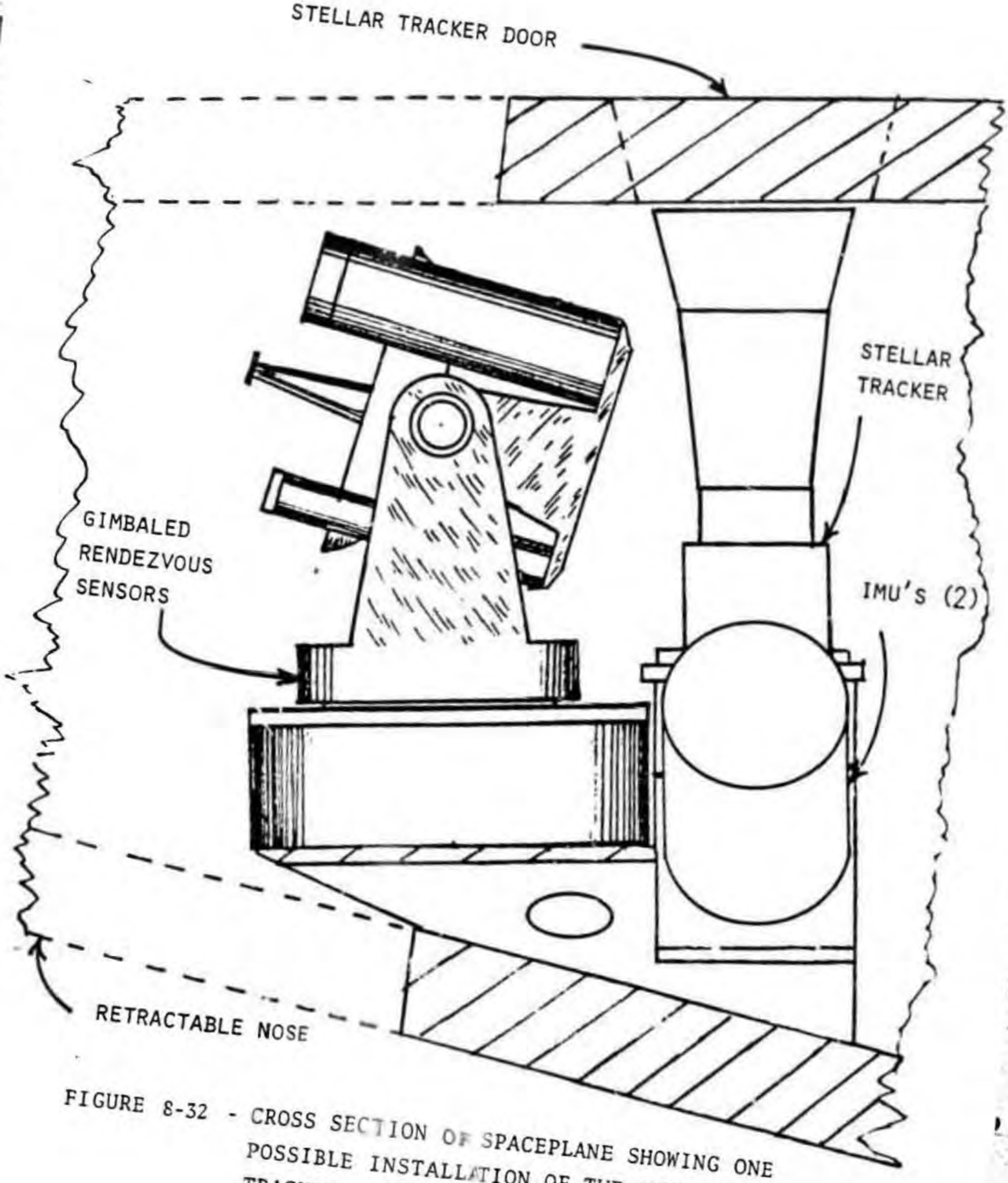


FIGURE 8-32 - CROSS SECTION OF SPACEPLANE SHOWING ONE POSSIBLE INSTALLATION OF THE IMU's, STELLAR TRACKER, AND GIMBALLED RENDEZVOUS SENSORS not be gimballed; the spacecraft would be oriented such that this dish would face downward when an altimeter reading was desired.

8.5.2 Avionics System Estimated Weight, Size and Power Usage

The avionics system estimated weight, size and power by subsystem, as well as totals are shown in Table 8-7. These values need to be confirmed during the next study phase when requirements are more specific.

Table 8-VII

AVIONICS SYSTEM ESTIMATED WEIGHT, SIZE, AND POWER USAGE

Components	Weight - lbs	Volume - ft^3	Aver. 28V Power-Watts (24 Hrs)
2 Laser Gyro IMU's	16	.34	30
1 Redundant Celestial Sensor & Shield	8	.55	2
1 Set Aerosurface Actuators & Controls*	75	1.25	10
2 Horizon Sensors	6	.05	5
3 Reaction Wheels	30	.30	15
1 Redundant Computer Assemply	40	.60	45
1 Redundant Control Electronics	30	.50	40
1 Redundant Command/TM Data Link & Ant.	23.2	.25	25
1 Redundant GPS Receiver & Ant.	30	.386	25
1 Redundant Clock	5	.05	5
1 Viewing Sensor, Radar Ranger, & Laser	40	1.00	25
1 Viewing Sight, Nav. & Situation Display	20	.32	25
1 Data Entry and Controls	12	.20	5
1 Earth Location Display	6	.086	5
1 Local Vert. Att. Display	6	.086	5
1 Aux. Power System	225	2.71	0
TOTALS	572.2 lbs	8.678 ft^3	267 Watts

* Aerosurface Controls also use 2.1 KW of 270 Volt Power During the 1/2 Hr. Re-Entry

8.6 CRITICAL TASKS RECOMMENDED FOR NEXT PHASE

- Task 1 - Avionics Systems Requirements:

  With a proper design reference mission and its variations as an input, a top avionics system performance and functional specification will be generated which will establish the operating modes, functions, mission duration, probability of mission success, etc. for the Spaceplane Avionics System. Critical interfaces to the Avionics System and initialization will be defined.

- Task 2 - Avionics Subsystem Requirements:

  The Avionics System will be partitioned into appropriate subsystems for which individual performance and functional specifications will be generated. This will establish detailed modes and functions, budget power weight and volume, and specify other critical characteristics such as MTBF.

  Currently identified subsystems are:

  - Power
  - Inertial Reference and Navigation
  - Celestial Sensor
  - Horizon Sensor
  - GPS Navigation
  - Tracking Telemetry and Command
  - Radar Altimeter
  - Momentum Controls
  - Reaction Jet Controls
  - Aerosurface Electric Drives
  - Rendezvous Sensors
  - Displays and Controls
  - Processors and Input/Output

- Task 3 - Baseline Definition:

  The existing baseline Avionics System will be reviewed and updated as appropriate with regard to specific subsystem hardware. Functional diagrams as well as latest size, weight, installation and interface data will be provided.

- Task 4 - Supporting Studies:

  In order to properly accomplish Tasks 1, 2 and 3 above which are to be considered the primary outputs of the design study the following supporting studies must be accomplished. (It is assumed that rendezvous simulation to help establish the requirements of the rendezvous sensors subsystem will be accomplished by Lin Com in a timely manner and Honeywell involvement will be limited to coordination with Lin Com.)

- Task 4.1 - Navigation Performance Analysis:

  Navigation performance analyses for critical mission phases will be performed. These include:

  - Boost from shuttle, airlaunch and booster deployment.
  - Rendezvous
  - Normal re-entry
  - Synergestic plane charge with second re-entry.

  Re-entry windows, corridors and profiles will need to be supplied to Honeywell in a timely manner. Prior to that time the minimum heating profile used on Shuttle will be assumed.

- Task 4.2 - Pointing Accuracy and Control Stability:

  With mission payload pointing accuracy and control stability requirements furnished to Honeywell, a control analysis will be performed to insure the adequacy of the momentum control system, with particular attention to wheel sizing. The effects of pilot motion and fuel slosh if any will be included.

- Task 4.3 - General Control Analysis:

  For normal control in space, not including rendezvous or precision payload pointing, when control is by reaction jets, an analysis will be performed to determine control characteistics, such as limit cycling, that will effect operation of such devices as the celestial sensor and horizon sensor.

- Task 4.4 - Optimization of Power Generation and Storage:

  Further study (probably subcontract) of the power generation subsystem will be accomplished to confirm the characteristics of the assumed load leveling system. Efficiencies, life, size, and weight of the assumed mono-propellant prime-mover needs to be confirmed. Battery type selection also needs to be confirmed for the avionics battery and the aerosurface power battery.

- Task 4.5 - Aerosurface Power Requirements:

  From re-entry and synergistic place change hinge moment time history data to be supplied to Honeywell, peak and average power required to drive the aerosurfaces will be determined.

- Task 4.6 - Processor Architecture:

  A trade-off will be made on the Processor architecture particularly with regard to using DoD standards such as 1750A and implementation of flight software in higher order languages.

- Task 4.7 I/O Configuration:

  A trade-off will be made to establish an optimum I/O system to interface the various subsystems. The main trade-off will be between a subsystem dedicated I/O versus a universal bus system allowing more flexibility such as the military standard 1553B.

- Task 4.8 - Man machine Interface:

  A man machine interface trade-off will be made to establish which functions should be automatized and which others can be performed by the pilot with a goal in view of achieving as much simplification as possible without burdening him. Display/Control functions will be coordinated with Hamilton Standard.

- Task 4.9 - Redundancy Management:

  In consort with the other systems of the Spaceplane, a redundancy management philosophy will be established. Generally, time critical safety critical functions will be reconfigured automatically, with other functions relying on pilot actions.

8.7 HONEYWELL CAPABILITIES APPLICABLE TO SPACEPLANE

The Honeywell Aerospace and Defense Group has a diverse and wide ranging spectrum of capabilities which have the potential to make a valuable contribution in bringing Spaceplane into being. The Spaceplane applicable capabilities are located in four operations of the Group as depicted in Figure 8-33.

Honeywell has a broad spectrum of current ongoing programs closely related to the needs of the spaceplane. The programs are illustrated in Figures 8-34 and 8-35.

On the Space Shuttle we have both the Flight Control System and the Main Engine Controllers. Honeywell has had more experience than any other company on boost guidance systems as currently represented by Centaur and Agena. On the DMSP-Block 5D, Honeywell has orbital injection as well as on orbit attitude determination responsibility. On the P80-1, Honeywell, in addition to complete orbit injection, on orbit attitude determination and control, is involved in Shuttle deployment of the vehicle. The recent work on Centaur in Shuttle will add to this experience.

The Dormant Inertial Navigation System represents the guidance responsibility for a re-entry vehicle and at the same time provides Honeywell experience in applying Laser Gyros in space. The Electrostatically Supported Gyro system Honeywell has been applying for a decade in the area of Precision A/C Navigation provides a valuable background in initialization for aircraft launched missions. Honeywell's Homing Overlay Experiment program provides a wealth of capability including long wave infrared sensing, laser ranging, precision pointing, intercept guidance and on-board data processing. The Forward Acquisition System program will further Honeywell's capabilities in on-board data processing.

The resulting systems capabilities that Honeywell has available applicable to tackling the Spaceplane program are shown in Table 8-8.

FIGURE 8-33

AEROSPACE & DEFENSE GROUP

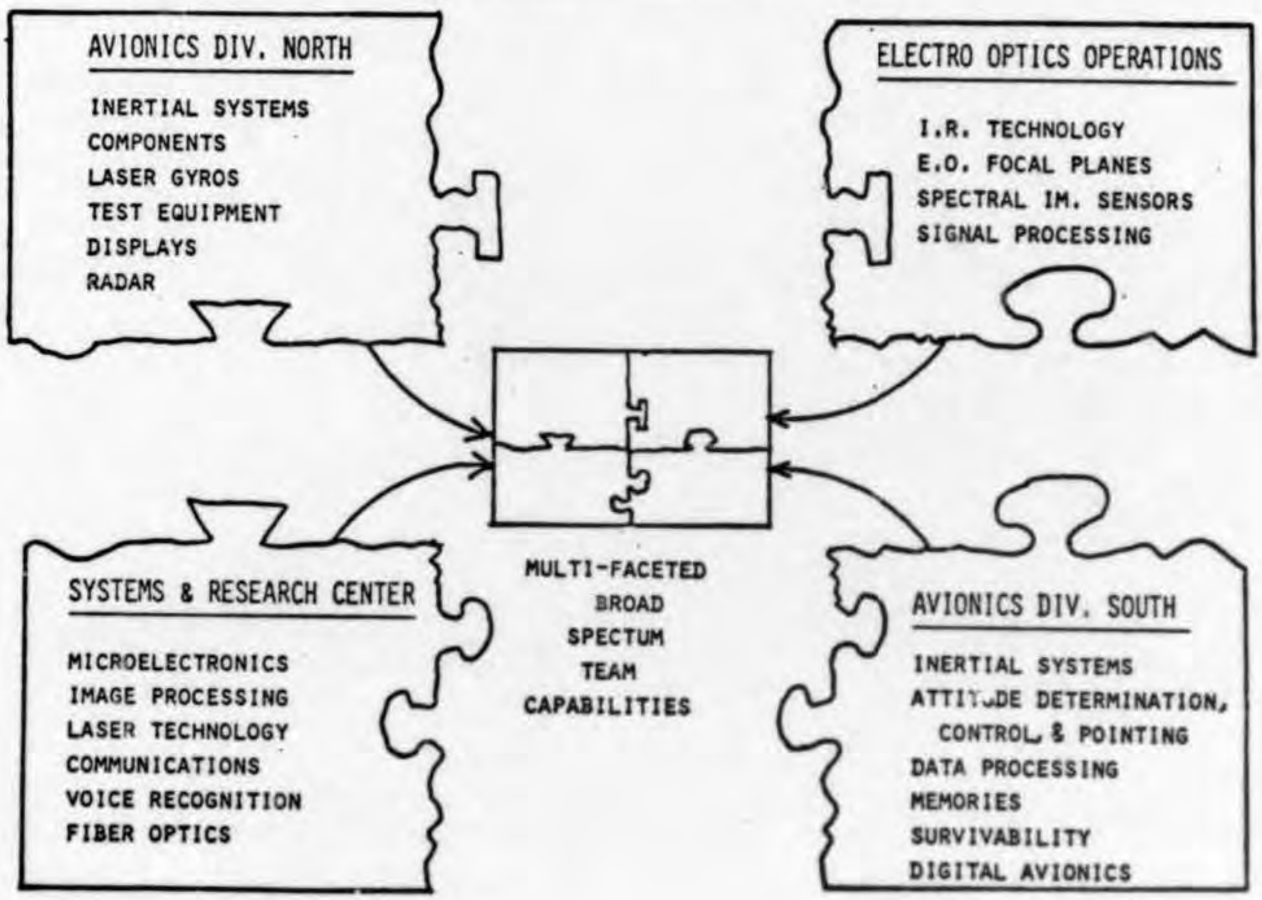
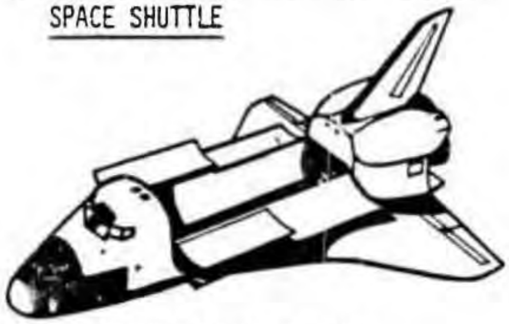


FIGURE 8-34

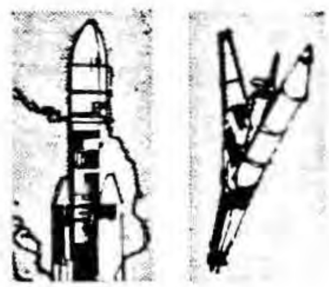
FOUR CURRENT RELATED PROGRAMS

SPACE SHUTTLE



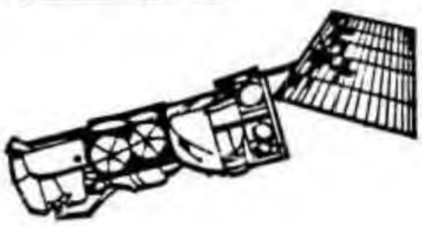
FLIGHT CONTROL SYSTEM
MAIN ENGINE CONTROLLER

CENTAUR AND AGENA LAUNCH VEHICLES



GROUND-
LAUNCHED
PRECISION
INERTIAL
GUIDANCE
SYSTEMS

DMSP - BLOCK 5D



ORBITAL INJECTION
ON-ORBIT PRECISION
ATTITUDE DETERMINATION

P80-1 SATELLITE

SPACE SHUTTLE LAUNCHED
ORBITAL INJECTION
ON-ORBIT PRECISION
ATTITUDE DETERMINATION & CONTROL

FIGURE 8-35

FOUR ADDITIONAL CURRENT RELATED PROGRAMS

DORMANT INERTIAL

NAVIGATION SYSTEM

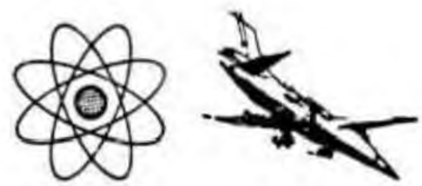
REENTRY
GUIDANCE
& CONTROL

ELECTROSTATICALLY SUPPORTED GYRO

AIRCRAFT NAVIGATORS

PRECISION MOTHERSHIP
NAVIGATION

ALIGNMENT TRANSFER



HOMING OVERLAY EXPERIMENT

DETECTION

ON-BOARD DATA
PROCESSING

LASER RANGING

INTERCEPT
GUIDANCE

FORWARD ACQUISITION SYSTEM

ON-BOARD
DATA PROCESSING

Table 8-VIII

RESULTING APPLICABLE SYSTEM CAPABILITIES

• Inertial Guidance:	Ground Launched Air Launched Orbital Re-Entry
• Control:	Manned Unmanned Atmospheric Space
• Intercept/Rendezvous:	Detection On-Board Data Processing Ranging Pointing

- Systems Engineering and Integration
- Software Management
- Real Time Redundance Management
- Nuclear & Radiation Hardening