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NUCLEAR SHUTTLE SYSTEM DEFINITION STUDY

PHASE III

PERFORMANCE REVIEW

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GEORGE C. MARSHALL SPACE FLIGHT CENTER

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LINE ITEM 2

LOCKHEED MISSILES & SPACE COMPANY

Sunnyvale, California

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FOREWORD

This report contains copies of the material presented by Lockheed Missiles & Space Company at the Second Performance Review of Phase III, Nuclear Shuttle System Definition Study, Contract NAS8-24715. It represents a portion of the reporting requirements listed on MSFC-DRL-197, Line Item 2, for this study contract. The meeting was held at the George C. Marshall Space Flight Center, Huntsville, Alabama, December 17, 1970.

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PRESENTATION OUTLINE

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INTRODUCTION

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PRESSURIZATION ANALYSIS

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STUDY OBJECTIVES

- COMPLETE RNS PHASE A ACTIVITIES
 - ESTABLISH RNS OPERATIONS REQUIREMENTS
 - REFINE RNS SYSTEMS DEFINITION
 - DEFINE RNS DEVELOPMENT REQUIREMENTS
- PERFORM DISCRETE ENGINEERING TRADE STUDIES
 - ENGINE INTERFACE
 - TANK PRESSURIZATION
 - E/STS-2 DESIGN SUPPORT
 - SAFETY

PHASE III STUDY PLAN

REUSABLE NUCLEAR SHUTTLE

The basic dimension drawing for the Lockheed Reusable Nuclear Shuttle is shown on the facing chart. This represents the assembled configuration as currently defined in the ongoing study.

The RNS consists of a central space frame, six propellant modules, and a propulsion module. The NERVA engine is a part of the propulsion module. The propellant modules are mounted circumferentially around the space frame, which contains the necessary fluid manifolding to feed the propulsion module.

The propulsion module is mounted on the aft end of the space frame. Each propellant module contains 38,676 lb of liquid hydrogen, and the propulsion module contains 37,371 lb of liquid hydrogen, thus providing a total propellant loading of 269,427 lb. Each of the basic elements is sized within a 15 ft. x 60 ft. envelope for compatibility with the Space Shuttle cargo bay. The NERVA engine and the orbital assembly adapter are delivered to orbit separate from the propulsion module, and final assembly occurs there.

The overall RNS length is 1964 in., including the length of the equipment module which is deployed on orbit. The current estimate for the inert weight of the assembled RNS is 81,241 lb, including a 5 percent contingency. The RNS performance for the reference lunar mission is 85,965 lb delivered to the moon and 20,114 lb returned from the moon. These weights and performance data include a 5900-lb supplemental shield for crew protection.

REUSABLE NUCLEAR SHUTTLE

RNS ORIENTATION REQUIREMENTS

The modular RNS is designed to contain all principal elements of the integrated avionics system on the equipment module portion of the space frame. These include the solar array portion of the power system, the antenna array portion of the communication system, and the star tracker elements of the navigation, guidance and control system.

The facing chart indicates the orientation requirements for the RNS on the lunar mission. The equipment installation is such that all elements can function throughout the mission with no interference. Principal orientation during coast is with the roll axis pointing toward the sun. This permits continuous communication and provides for periodic positional updating via the star tracker. During powered flight, the vehicle is oriented along the flight path and rolled as necessary to align the solar arrays.

RNS ORIENTATION REQUIREMENTS

INTERIM CONCLUSIONS

- PHASE II SYSTEM AND SUBSYSTEM SELECTIONS VERIFIED
- DESIGN AND OPERATIONAL FLEXIBILITY OF MODULAR RNS CONTINUE TO BE ATTRACTIVE
- ORBITAL ASSEMBLY FEASIBILITY DEMONSTRATED
- INITIAL DEFINITION OF NERVA ENGINE INTERFACE ESTABLISHED
- NERVA ENGINE OPERATING POINT IN THE 24-T0-26-PSIA RANGE

REMAINING ACTIVITIES

- PERFORM ADDITIONAL EVALUATION OF RADIATION ENVIRONMENT PROTECTION
- COMPLETE DEFINITION OF RNS SYSTEMS
- REFINE RNS DEVELOPMENT PLAN
 - TEST
 - MANUFACTURING
 - FACILITIES
 - TECHNOLOGY
 - SCHEDULE
 - COST

SYSTEMS DEFINITION

TASK 3 OBJECTIVES

- DEFINE THE SELECTED RNS CONCEPT IN DEPTH
 - IDENTIFY PERTINENT DESIGN CRITERIA
 - CONDUCT SUBSYSTEM DESIGN
 - ESTABLISH A DETAILED CONCEPTUAL DESIGN OF THE BASELINE RNS

TASK 3 REQUIREMENTS

DESIGN CRITERIA

Design criteria, including both structural and environmental design criteria, have been established for the RNS. In conjunction with the study guidelines and requirements, these criteria have been compiled into a report that serves as the basis for vehicle design and operation.

DESIGN CRITERIA

- DESIGN CRITERIA
 - STRUCTURAL
 - ENVIRONMENTAL

- GUIDELINES
 - PROGRAM
 - MISSIONS
 - RNS
 - INT-21
 - SPACE SHUTTLE
 - PROGRAM ELEMENT INTERFACES

- REQUIREMENTS
 - PROGRAM
 - MISSIONS
 - RNS

WORK BREAKDOWN STRUCTURE

The Work Breakdown Structure has been developed in accordance with a procedure agreed upon by NASA and the contractors. The Reusable Nuclear Stage has been assigned the third level. In the multiple-tank system, the three modules -- control and assembly structure module, propulsion module, and propellant module -- are at the fourth level, and the basic subsystems are at the fifth level.

WORK BREAKDOWN STRUCTURE

STRUCTURE SYSTEM

The structure system includes structural elements of the propulsion module, propellant module, and control and assembly structure. The various structural elements within each module are shown in conformance with the WBS format for this vehicle.

STRUCTURE SYSTEM

SPACE FRAME DESIGN CONDITIONS

The space frame design conditions were based upon full thrust engine loads during the various mission phases with appropriate payload combinations. Those loading conditions which represent critical conditions in the design are shown in the table. A summary of the areas considered include:

- o Payload/equipment module interface
- o Equipment module/space frame interface
- o Space frame structure
- o Propellant module/space frame forward interface
- o Propellant module/space frame aft interface
- o Propulsion module/space frame interface

A space frame computer program was used for establishing the loads in the elements. An ultimate factor of safety of 1.4 was used for all conditions examined.

SPACE FRAME DESIGN CONDITIONS

	CRITICAL DESIGN CONDITION	DESIGN LOAD *
PAYLOAD INTERFACE (P/L)	LOI, P/L 146 K, 0.266 g	AXIAL, 38,800 LB SHEAR, 1410 LB MOMENT, 545,600 IN-LB
EQUIPMENT MODULE INTERFACE	SAME AS P/L	X- 8922 LB } EACH Y- 99 LB } LONG- Z- 544 LB } ERON
BASIC SPACE FRAME STRUCTURE	SAME AS P/L + 0.69-DEG C.G. OFFSET	AXIAL, 40,595 LB SHEAR, 1,868 LB MOMENT, 1,888,506 IN-LB
PROPELLANT MODULE FWD SUPPORT	TEI, P/L 26K, 0.421 g	AXIAL, — SHEAR, 263 LB
PROPELLANT MODULE AFT SUPPORT	SAME AS FWD SUPPORT	AXIAL, 18,410 LB SHEAR, MOMENT, 1,841,000 IN-LB
PROPULSION MODULE INTERFACE	TLI (T=0) P/L 0, 0.218 g	AXIAL, 58,815 LB SHEAR, 1,358 LB MOMENT, 991,677 IN-LB

* ALL DESIGN LOADS INCLUDE PITCH DUE TO 1.5-DEG. GIMBAL WHERE APPLICABLE

STRUCTURAL FRAME -- SPACE FRAME MODULE

The hexagonal core of the space frame forms the basic load-carrying structure of the RNS. This space frame serves as a thrust structure, propellant module, support structure, forward docking structure, aft propulsion module docking structure, and vehicle alignment system. An equipment module is located in the forward end of the space frame and is stowed inside the space frame during launch in the Space Shuttle. The equipment bay is extended 80 in. forward with the bi-stem boom system in orbit.

The critical design condition of the structure is compression buckling. The individual sections being considered are welded, tubular, truss structures of aluminum alloy, 6061-T6. This material was chosen for the space frame because of the ease and reliability of its weldments. However, the particular alloy and temper are not of great significance, because of the relatively high slenderness ratio of most of the members; i.e., they are designed by column buckling rather than material strength. Other material considered were magnesium, titanium, beryllium, Lockalloy, boron epoxy, and graphite epoxy. The costs of these materials are significantly higher, and a detailed design is required to make a selection.

The space frame was analyzed with an LMSC space frame computer program incorporating an ultimate factor of safety of 1.4.

STRUCTURAL FRAME— SPACE FRAME MODULE

PROPELLANT TANK WEIGHT

Propellant tanks have been designed both for the launch environment within the Space Shuttle and for all operational phases. The launch loads are based on the following Space Shuttle loading conditions:

Flight Phase	Maximum g Loading			
	①	②	③	④
Launch Liftoff	<u>+1.5</u>	0.3	0.3	<u>+0.3</u>
Maximum α Q	1.7	0.5	0.5	<u>+0.5</u>
Stage Burnout	3.0	0.3	0.3	<u>+0.3</u>
Entry	-1.5	-2.0	--	--
Approach-Landing	-1.0	0.5	2.7	<u>+0.5</u>

The launch loads dictate the minimum pressures and consequently the minimum weight of the 2021 aluminum tanks. Operating pressure during the mission increases the tank weight according to the graph on the facing page.

PROPELLANT TANK WEIGHT

* TOTAL TANK WEIGHT
 = MEMBRANE WEIGHTS
 PLUS WELDLINE WT (175#)
 PLUS TANK SUPPORT, PENETRATIONS (367#)
 PLUS 10% CONTINGENCY

ULLAGE PRESSURE AT 1.0g ENGINE BURN - PSI

TANK -- PROPELLANT MODULE

The tank size is based on the 15 x 60-ft Space Shuttle cargo bay allowing for a 3-in. radial clearance around the tank for insulation and meteoroid shielding. The tank is a cylinder with elliptical $\sqrt{2}$:1 bulkheads, which are girth-welded to the cylinder. The forward bulkheads incorporate an access cover and the aft bulkheads a 75-deg sump. The tank has a volume of 9,190 ft³, which allows a propellant capacity of 38,676 lb with 5-percent ullage. The tank material is 2021-T81 aluminum.

TANK-PROPELLANT MODULE

TANK -- PROPULSION MODULE

The tank size is based on the 15 x 60-ft Space Shuttle cargo bay, allowing for a 3-in. radial clearance around the tank for insulation and meteoroid shielding and a 25-in. axial clearance for the propellant feed system installation in the aft end. The tank is a cylinder with elliptical $\sqrt{2}:1$ bulkheads which are girth-welded to the cylinder. The forward bulkhead incorporates an access cover. Also, a skirt is located forward and aft of the tank. The tank has a volume of 8,880 ft³, which allows a propellant capacity of 37,371 lb with 5-percent ullage. The tank material is 2021-T81 aluminum.

TANK-PROPULSION MODULE

METEOROID PROTECTION

Double-wall shielding has been evaluated on the basis of no damage to tank surfaces.

The probability of no tank damage has been assessed for both $P_o = 0.995$ and 0.950 -- using the average meteoroid environment containing sporadic and shower particles, with appropriate corrections for operation near planets.

Bumper thickness is established by a thickness-to-particle-diameter ratio of 0.10 as the minimum material that will effectively destroy particles. The particle size is dictated by exposed area and time of exposure. A portion of each clustered tank is inherently shielded, and two-thirds of the total area is used for both 45-day and 365-day missions. As shield standoff distance decreases, the second sheet must become thicker to avoid rupture from liquid and vaporous spray behind the bumper sheet. The variation of second-sheet thickness with spacing is founded on current ballistic limit data for lower-speed laboratory experiments that are more damaging than meteoroid conditions at the second sheet. The data showing thickness and spacing variation for two mission times and two probabilities strongly indicate that a single shielding element may provide the necessary protection for 45-day missions employing clustered tanks.

METEOROID PROTECTION ASSUMPTIONS

- DOUBLE WALL ALUMINUM METEOROID SHIELDING
- PROBABILITY OF NO TANK SURFACE DAMAGE = .995/.950
- AVERAGE TOTAL METEOROID ENVIRONMENT USED
- BUMPER THICKNESS BASED ON SURFACE AREA AND EXPOSURE TIME
- INDIVIDUAL TANKS ARE PROTECTED ON $\frac{2}{3}$ OF THE TANK SURFACE
- MISSIONS ARE 45 DAYS AND 365 DAYS
- SECOND SHEET THICKNESS VARIES WITH SPACING
- BALLISTIC LIMIT OF BOTH SHEETS CONSIDERS:
 - METEOROID VELOCITY OF 20 KM/SEC
 - METEOROID PARTICLE DENSITY OF 0.5 GM/CC

METEOROID SHIELDING

Meteoroid shielding requirements are very sensitive to shield spacing, surface area, and exposure time. For single tanks, to provide 0.995 or greater probability of no system failure due to meteoroid puncture of the vehicle tank, the shielding must be adequate to provide for a $P_o = 0.995$. For multiple-tank systems, with the loss of any two tanks with a $P_o = 0.950$ each (except the propulsion module) system reliability still is greater than for a $P_o = 0.995$ single-tank system. Consequently, the multiple-tank system has been provided with shielding having a $P_o = 0.950$ for the propellant module and a $P_o = 0.995$ for the propulsion module -- because this tank corresponds to a single-tank system. In addition, the initiation of each mission was considered an independent event and consequently the exposure time and shielding were minimized.

METEOROID SHIELDING

DOUBLE WALL ALUMINUM

PROPULSION SYSTEM

The propulsion system includes propulsion elements of the propulsion module, propellant module, and control and assembly structure. The various propulsion elements within each module are shown in conformance with the WBS format for this vehicle.

PROPULSION SYSTEM

PROPELLANT FEED SYSTEM

The RNS propellant feed system is designed to combine simplicity, high reliability, and minimum pressure drop in controlling the flow of LH_2 to the engine. Plumbing for each of the six propellant modules is identical. Each tank, including the propulsion module, is equipped with an inflight vent system that can operate with any combination of liquid and gas at the inlet and an emergency vent system to safeguard the tank against abnormal pressure rise. All feed valves, with the exception of the two TSOVs at the propulsion module outlet, are designed to fail open in order to preclude loss of liquid availability from any tank. The propulsion module TSOVs must be designed to fail closed in order to be consistent with the NERVA engine redundant feed system and to permit either leg to be isolated in the event of a TPA or open line failure.

All tanks are connected to the space frame manifold by means of remotely operated disconnects. To avoid undesirable pressure drop, disconnect checks are not used in these lines. Connector leaks can be isolated with the space frame isolation valves, normally open during flight. Except when propellant outflow is required, propellant module shutoff valves are held closed.

Ground fill and drain valves on each tank provide a common interface with the Space Shuttle that will not interfere with tank deployment. They are capable of actuation only when the tank is secured in the Space Shuttle cargo bay.

PROPELLANT FEED SYSTEM

FEED SYSTEM LINE SIZE OPTIMIZATION

The objective of this analysis was to optimize the feed system line size for transfer of LH_2 from the propellant modules to the propulsion module. In this analysis it was necessary to consider the following:

- Weight of the line for the various operating pressures and line sizes
- Propellant losses during chilldown of the line prior to transfer
- Propellant losses during chilldown of the valve on the space frame
- Increase in tank weight resulting from pressure losses in the various line sizes at various operating pressures

The results of these analysis, which are presented on the opposite page, indicate that the optimum feedline size is 6 inches. Should an increase in operating pressure be anticipated, an increase in system weight and line size can also be anticipated.

FEED SYSTEM LINE SIZE OPTIMIZATION

PROPELLANT FEED LINE CONFIGURATIONS

In addition to the propellant manifold optimization, an analysis of the propulsion module propellant feed system was conducted. The primary concern in this analysis was the basic configuration of the feedline and valves, as well as interface complexity. The configurations analyzed included the three concepts shown on the facing page.

Configurations No. 1 and No. 2 were evolved in an attempt to simplify the remote interface and the number of tank penetrations. The disadvantage of these configurations as compared with Configuration No. 3 are: (1) a high ΔP (and consequently a heavier system) and, (2) an extra valve (capability of shutting off each line on each side of the remote interface is required). Configuration No. 1 has less propellant capacity in the tank than Configuration No. 2 (approximately 350 lb of LH_2 due to shortened distance between the tank bottom and remote interface).

Configuration No. 3 provides a longer tank (as does No. 2) with a lower ΔP and one less valve. The two remote interfaces for the propellant feedline do not present any additional design problems, and the thermal heat load into the tank does not appear to be a significant problem.

PROPELLANT FEED LINE CONFIGURATIONS

TANK TOP

- CONFIGURATION #1**
- SINGLE TANK PENETRATION
 - SINGLE LINE ACROSS REMOTE INTERFACE
 - 3 VALVES

60'

- CONFIGURATION #2**
- SINGLE TANK PENETRATION
 - SINGLE LINE ACROSS REMOTE INTERFACE
 - LONGER TANK (350 LB LH₂)
 - 3 VALVES
 - FEED LINE ASYMMETRY

- CONFIGURATION #3**
- LONGER TANK (350 LB LH₂)
 - 2 VALVES
 - MINIMUM PRESSURE DROP (SYSTEM PENALTY = 600 LB/PSI)
 - TWO TANK PENETRATIONS
 - TWO REMOTE INTERFACES

ENGINE FEED SYSTEM SELECTION

After selecting a basic propellant-feedline configuration, the line size was optimized. The sensitivity of valve type to line size and system penalty was also evaluated. The types of valves considered were the "poppet", or center-body valve and the "eyelid" ball valve.

Parameters in these trade analyses included line weights for lines from 5 to 10 in. in diameter; valve weights for corresponding line sizes; line chilldown losses (four startups); valve chilldown losses (four startups); and increase in tank weights resulting from pressure losses in the various line sizes and at three basic operating pressures (22, 24, and 26 psia). Additional pressure losses from the valves were not included although, of the two valve types considered, the poppet valve inherently has a higher ΔP .

The results shown on the facing page represent the summation of weight penalties associated with the various line sizes, valve types, and operating pressures. For the higher-weight poppet valve, the optimum line and valve size for all operating pressures is approximately 7 in. The eyelid valve system is optimum for line diameters between 7 and 8 in., with weight advantage over the poppet valve of 100 (22 psia) to 150 lb (26 psia). An 8-in. line size is recommended.

ENGINE FEED SYSTEM SELECTION

CONFIGURATION 3
(2 LINES - 2 VALVES)

PRESSURIZATION SYSTEM

The pressurization system is designed to permit pressurization of any combination of propellant modules and propulsion module, using high-pressure gas bled from the NERVA engine. All valves, with the exception of the manifold vent, are located on the individual modules. Redundant pressurization-control valves and pressure sensors ensure a fully operational system after any double-component failure. The gas cooldown valve provides the capability of using vent gas from any module as an alternate coolant medium for the engine.

Filters at each tank inlet guard against contamination from engine sources. The orifice in the propulsion module inlet permits a differential pressure to be maintained, using the propulsion module vent valve, in the event one of the control valves fails to open.

PRESSURIZATION SYSTEM

PURGE SYSTEM

For multiple re-use tankage, a seal-reseal function is required on the LH₂ tank purge bag for the ascent (and possibly for reentry cycles). Helium gas is used to remove the air initially from the insulation during ground hold. The purge bag is then vented during ascent so the insulation pressure will reach $< 10^{-5}$ torr in space. If the tank is brought back into the Earth's atmosphere, either the insulation must be repurged if the tank is still cold or the tank must be warmed to prevent air or moisture condensation in the insulation (moisture has been shown to remove aluminized coating but not gold coating). Use of gold in place of aluminum alleviates the moisture problem but drives the insulation material cost up by a factor of about 100.

Either of two approaches can be taken to providing the venting reseal function. First, butterfly valves can be used at different locations on the bag and cycled open or closed as required. However, unless a large number of valves are used, the time to reach 10^{-5} torr will be long, and boiloff will be high in the interim.

The second approach is to reduce the vent-path length, and consequently vent time, by opening the purge bag with tubes located over the insulation joint areas as shown on the chart. Inside each of these tubes is another tube that collapses into a flattened figure-eight shape when not pressurized. When the flattened tube is pressurized, it expands and fills the outer tube, sealing off the vent holes (for ground hold or reentry phases of flight if required). When it is desired to vent the insulation again (ascent), the three-way valve releases the gas pressure in the inner tube.

The GHe used to purge the insulation during ground hold (or prior to reentry if desired) is introduced at the top and vented out of the bottom of the purge bag (ground hold only), using the same He supply source as the vent tube mechanism. Also indicated in the chart are He purge requirements for various hold times.

PURGE SYSTEM

RCS CONCEPT COMPARISON

The RNS Reaction Control System (RCS) provides attitude control (rotation about pitch, yaw, and roll axes), as well as vernier translation in the plus or minus direction along each of the three vehicle axes. Major impulse in the forward direction (as might be required during orbit phasing or midcourse correction maneuvers) as well as TVC during main-engine operation is provided by the NERVA engine. Factors considered in system selection were performance, cost, availability, safety, reliability, and maintainability. Due to the large impulse requirement, all system concepts other than mass expulsion using bipropellant thrusters were discarded on the basis of performance.

Earth storables ($\text{N}_2\text{O}_4/\text{MMH}$) were chosen as the baseline RCS propellant combination. From the weight-sizing trade analyses, earth storables provide major advantages over cryogenic systems with specific impulse up to 1.2×10^6 lb-sec. The most sensitive parameters in this selection were minimum impulse bit (1 lb-sec as against 3 to 4 lb-sec for cryogenic $\text{H}_2\text{-O}_2$ systems) and total impulse. Space storage systems such as those using $\text{OF}_2\text{-B}_2\text{H}_6$ (Isp = 380 to 410 sec) offer attractive performance, but were eliminated on the basis of possible safety problems and lack of data on long-term corrosive effects.

RCS CONCEPT COMPARISON

	PERFORMANCE	COST	AVAILABILITY
EARTH STORABLE N_2O_4/MMH	<ul style="list-style-type: none"> • MEDIUM I_{sp} • LOW INERT WEIGHT • SMALL MIN. IMPULSE BIT (1 LB-SEC) • <u>HYPERGOLIC IGNITION</u> <p>SYSTEM WEIGHT 2840 LB</p>	SYSTEM SIZING BASED ON FLIGHT- PROVEN HARDWARE, NO TECHNOLOGY DEVELOPMENT REQUIRED	FLIGHT-PROVEN ON AGENA & APOLLO. HARDWARE AVAILABLE
CRYOGENIC O_2/H_2	<ul style="list-style-type: none"> • HIGH I_{sp} • HIGH INERT WEIGHT • LARGER MIN. IMPULSE BIT (3-4 LB-SEC) • <u>IGNITER REQUIRED</u> <p>SYSTEM WEIGHT 3045 LB</p>	NO FLYABLE HARD- WARE HAS BEEN BUILT. A 50-250 LB THRUSTER WOULD REQUIRE DEVELOP- MENTAL FUNDING	NOT PRESENT STATE- OF-THE-ART. SPACE SHUTTLE REQUIREMENTS NOT APPLICABLE. NO SMALL MOTOR DEVELOPMENT PLANNED
SPACE STORABLE OF $2/B_2H_6$	<ul style="list-style-type: none"> • HIGH I_{sp} • <u>LOW INERT WEIGHT</u> <p>SYSTEM WEIGHT 2400 LB</p>	MAJOR DEVELOPMENT OF A NEW TECHNOLOGY	NO LARGE-SCALE TECHNOLOGY DEVEL- OPMENT PLANNED

RCS SYSTEM SCHEMATIC

The RCS system was sized at 630 K lb-sec to meet lunar mission requirements, with growth capability to 1×10^6 lb-sec. Total system weight is 2,840 lb, of which 2,200 lb is impulse propellant. Thirty thrusters at 100 lb each are required to meet the following reliability goals:

- Attitude Control: dual redundancy -- fail operational, fail operational, fail safe.
- Axial ΔV : single redundancy -- fail operational, fail safe.
- Lateral ΔV : zero redundancy -- fail safe.

This system is designed for on-orbit maintenance and has dual RCS propellant modules capable of exchange or refueling. Failed or degraded thrusters can be replaced in orbit. The engine module RCS system can be charged from the RNS RCS system or through the common fill lines.

RCS SYSTEM SCHEMATIC

RCS THRUSTERS -- ARRANGEMENT

The thrusters have been arranged to provide the required redundancies and to minimize propellant consumption during all mission phases. Since the RNS is a flyable, autonomous system when only the space frame is deployed, RCS capability must be provided for this configuration. The RCS thrusters facing the payload are mechanically locked out when the payload is in position.

RCS THRUSTERS-ARRANGEMENT

ASTRIONIC SYSTEM

The astrionic system includes astrionic elements of the propulsion module, propellant module, and control and assembly structure. The various astrionic elements within each module are shown in conformance with the WBS format for this vehicle.

ASTRIONIC SYSTEM

PROPULSION MODULE

PROPELLANT MODULE

CONTROL & ASSY STRUCTURE (SPACE-FRAME)

INTEGRATED AVIONICS SYSTEM

An integrated avionics system was utilized in order to integrate all of the astrionic subsystems physically and functionally. The enclosed schematic shows the functional interaction as well as the physical location of the various elements of the system.

The data management subsystem integrates the RNS avionics system, providing for the control of signal flow between subsystems. The data management subsystem consists of a central processor, a high-speed memory, and bulk storage. A data bus is used for signal distribution. Local processors are utilized in each module in order to reduce the data flow on the bus and perform the maximum number of computations in situ.

The signals are further distributed within each module on a data bus to the respective standard interface units. The indicated input/output boxes may be integrated into the respective components or may act as separate units for interface between the components and the standard interface units. The diagram indicates not only the data management distribution system throughout the RNS but also the interaction of the power system and communications system with the data management system.

The use of this integrated approach with a local processor on each module allows for the monitoring and checkout of all subsystems not only on the RNS, but also when the module is in the Space Shuttle. Checkout on the ground and during ascent is feasible with minimal software updating by the Space Shuttle data management system.

INTEGRATED AVIONICS SYSTEM

COMMUNICATION SYSTEM

The communication system provides for the assembly of signals, conversion to r-f, command decoding and generation, and reception and transmission between the RNS and all space communication network elements, including ground communication elements where applicable.

The S-band equipment provides both the primary and secondary mode communication links. Four omnidirectional antennas are used and a 4-ft-diameter, steerable, stowable antenna is used for long-range communication or for high-data-rate transmission.

The UHF equipment is the primary voice link for RNS to the other space vehicles. The UHF system may also be used as a wake-up command. Two omni antennas are located 180 deg apart on the vehicle. An emergency transceiver operating at 4-GHz is incorporated and operates in a manner similar to the primary system. Two omni antennas are incorporated for this system.

COMMUNICATION SYSTEM

NAVIGATION, GUIDANCE, AND CONTROL -- FUNCTIONAL DIAGRAM

The NG&C system is an explicit inertial navigation and guidance system operating in a closed-loop strapdown configuration. It provides steering signals, attitude control commands, and vernier translation commands during the mission. In addition, the system generates real-time discrete signals for engine start and cutoff and provides control of the variation in occurrence and duration of cooling pulses.

The basic strapdown system is augmented by equipment peculiar to the mission phase. The inertial system can be updated by DSIF to improve the onboard estimate of the state vector and by the onboard celestial sensors to update the attitude estimate.

NAVIGATION, GUIDANCE, & CONTROL

FUNCTIONAL DIAGRAM

NG&C / ENGINE INTERFACE

A most important interface exists between the RNS NG&C system and the NERVA digital I&C electronics assembly (NDICE). While the NDICE provides primary control of engine functions, the NG&C system must provide time-critical initiation and termination commands with the ability to alter the nominal operating profile in order to minimize trajectory errors related to variations between (1) the predicted nominal operating profile, using the program stored in the guidance computer; and (2) the measured performance delivered by the engine. Preliminary evaluation has identified the following NG&C/NDICE command signal interfaces:

- o Initiate startup -- based on predicted operating profile
- o Enable thrust buildup -- NG&C and NDICE must both indicate "go"
- o Begin shutdown -- based on delivered performance and predicted shutdown-cooldown profile
- o End throttle hold -- based on updated prediction
- o End pump tailoff -- assumes PTO can be extended using NERVA "idle mode" to minimize impact of prediction errors
- o Pulse initiation and termination -- initiation and termination by NG&C command would be the normal mode, with temperature demand control optional. Limit temperature override would insure safe conditions at all times.

There will be additional NG&C/NDICE command signal interfaces for malfunction-mode and emergency-mode operation, but they have not yet been defined.

NG & C / ENGINE INTERFACE

POWER PROFILE -- LUNAR MISSION

A power profile was established for the lunar shuttle mission. The profile is based on individual subsystem requirements throughout all mission phases. Power requirements during the assembly phase were also established. The NERVA engine is the most demanding for both average and peak power requirements. The average power level is below 1 kw throughout the mission, with peaks to over 3 kw during engine burn.

POWER PROFILE-LUNAR MISSION

POWER SYSTEM COMPARISON

The power system sources were reexamined in the light of the reduced NERVA engine requirements. Primary batteries, fuel cells, RTG-secondary battery, and solar-array/secondary-battery systems were evaluated comparatively. The primary emphasis of reusability and autonomy indicates a preference for either RTG or solar arrays. A considerable weight advantage accrues with the use of solar arrays, which also have the advantage that considerable experience has been gained in their employment. Fuel cells must be reintroduced, however, if payload life-support functions are integrated.

POWER SYSTEM COMPARISON

POWER SYSTEM	FUNCTIONAL	PERFORMANCE			SAFETY	RELIABILITY
		COST 10 MISSIONS	WT 10 MISSION BASIS, LB	AVAIL- ABIL- ITY		
PRIMARY BATTERIES	SIMPLEST SYSTEM	\$ 2.0 M \$ 5.0 M	51,300 34,200	1971 1978	EXCELLENT	HIGH, AS REDUNDANCY IS EASY TO PROVIDE
FUEL CELLS H ₂ - O ₂	COMPLEX SYSTEM, REQUIRES RADIATOR AND REACTANT RESUPPLY	\$ 0.7 M	5,085	1973	GOOD, WITH NORMAL PRECAUTIONS IN HANDLING HYDROGEN AND OXYGEN	GOOD, AS REDUNDANCY CAN BE PROVIDED AT RELATIVELY SMALL WEIGHT PENALTY
RTG SECONDARY BATTERIES	POWER MANAGEMENT REQUIRED	\$ 1.0 M HDWE + \$ 6.0 M ISOTOPIC FUEL	1,200	1975	GOOD, BUT PRECAUTIONS MUST BE TAKEN WITH RADIOISOTOPES	GOOD, AS REDUNDANCY CAN BE PROVIDED IN THE RTG DESIGN AND REDUNDANCY IN BATTERIES IS NO PROBLEM
SOLAR ARRAYS - SECONDARY BATTERIES	POWER MANAGEMENT REQUIRED. EXTENSIVE EXPERIENCE	\$ 0.9 M	560	1972	EXCELLENT	GOOD, BASED ON EXPERIENCE & EASE OF PROVIDING CONSERVATIVE DESIGN FACTORS

POWER SYSTEM CONCEPT

The selected power system concept is based on a redundant, flexible, solar-array power source with redundant battery sources to handle peak loads as well as mission phases during which the solar array is inoperative. The power is distributed on an unregulated bus with junction boxes for the equipment bay, propellant module, and propulsion module. The solar arrays are of the flexible roll-up type that can be extended and retracted. Nickel-cadmium batteries are currently envisioned in this support role.

POWER SYSTEM CONCEPT

ENVIRONMENTAL CONTROL SUBSYSTEM

The environmental control subsystem rejects heat generated by components within the equipment module and maintains equipment temperature within required limits during all mission phases. Internal heat loads vary with duty cycle conditions, and external environmental heating rates vary with orientation relative to the Earth, Moon, and Sun.

Candidate thermal-control concepts considered were passive/heaters, louvers, fluid-loop, and heat pipe systems. Ranking of the concepts relative to the most significant parameters led to selection of a passive/heater approach.

The chart shows the heat-rejection capability of equipment-module radiating skins and the area required to reject the module heat load. The module has adequate radiating surface area with either the 15- or 33-ft-diameter payload.

ENVIRONMENTAL CONTROL SUBSYSTEM

VEHICLE INTEGRATION

Vehicle integration was initiated subsequent to the subsystem concept selection. The subsystem concepts were evaluated for interaction with each other in addition to satisfying operational requirements. The effects of the launch vehicle, launch operation, mission operation, and payload were considered.

The integrated vehicle was subjected to a detailed thermodynamics analysis and a radiation effects analysis. The effects of reliability and maintainability are included in the integration process. Subsequent to these analyses, the Nuclear Shuttle definition can be accomplished.

VEHICLE INTEGRATION

MAINTAINABILITY APPROACH

The basic maintainability approach is inherent in the vehicle design. At the lowest level, self-repair techniques such as voting logic are incorporated in the design. Where this is not feasible, redundancy is employed, especially in components such as valves.

At the next level, replacement by EVA becomes the maintainability approach. This can be an item such as a thruster or a sub-module that may be in the equipment bay.

The final step is the replacement of a complete module. All of the modules -- propellant propulsion, and control and assembly structure--can be replaced in this manner.

MAINTAINABILITY APPROACH

- MODULE REPLACEMENT
- SUB-ASSEMBLY REPLACEMENT
- REDUNDANCY
- SELF - REPAIR

SUBASSEMBLY REPLACEMENT

The replacement of subassemblies can be accomplished when the replacement of a total module is not warranted. This approach allows the replacement of a module in an expedient manner when a critical component in the assembly has failed. These modules are sized so that they can be handled by the crew in an EVA mode. Typically, the subassemblies are modularized and slide on a track into their supports. The rotary latching mechanism makes the functional connections and locks the subassemblies in place.

SUBASSEMBLY REPLACEMENT

**RCS PROPELLANT
SUPPLY**

ELECTRONIC MODULE

SOUTH ATLANTIC ANOMALY

Members of the RNS orbital-operations-support or Space Shuttle crews and Space Station personnel will be exposed to natural nuclear-radiation environments in performance of their missions. These hazards are due to trapped protons, trapped electrons, and galactic cosmic radiation (GCR). The peculiarity of this phenomenon is that exposure is experienced only when the orbiting body passes over the South Atlantic regions. An isoflux plot of this anomaly at an altitude of 240 nm is shown on the facing page. The radiation intensity increases with increasing altitude and is influenced by the orbital inclination angle. Peak intensities occur around the 35-deg inclination orbits for all altitudes but decay very slightly up to 90-deg inclinations.

As the flux levels of the anomaly are high in the 31-deg, 260 nm orbits, concern has been expressed for the radiation exposure that RNS support and maintenance personnel would experience. Analyses were undertaken to determine the number of passes through the South Atlantic Anomaly, the anticipated levels and exposures, and the impact on RNS orbital operations.

SOUTH ATLANTIC ANOMALY

EXPOSURE HISTORY FROM SOUTH ATLANTIC ANOMALY

In the low earth orbit, the trapped radiation exposure is time-dependent, and essentially all of this radiation is received during the passage through the South Atlantic magnetic anomaly. The time-dependence of proton exposure in the 31-deg, 260-nm orbit is presented on the facing page. Electron exposure would show a similar time-dependence for this orbit but would be significantly lower in intensity. For higher orbit inclinations, additional small electron pulses would appear as the orbiting body approached the 50-deg latitudes.

Note that exposure occurs in pulses during a 12-hr period each day. The pulses will exhibit variations each day due to the 6-deg regression rate, but generally will be the same. With this characteristic, it can be seen that there is a 14-hr period each day during which no exposure is experienced -- permitting a significant work period without any hazard from natural nuclear radiation. However, as use of this period might not always be timely or cost-effective, analysis of the "worst case" exposure was undertaken. These data are presented on the facing page.

EXPOSURE HISTORY

FROM SOUTH ATLANTIC ANOMALY

TOTAL SPACE RADIATION EXPOSURE

The radiation exposure to personnel in orbit is dependent upon nuclear radiation flux and dwell time in the radiation field; in addition, the total exposure is dependent on the shielding afforded by the spacesuit and/or spacecraft structure. The integrated exposures presented on the facing page indicate the significance of this shielding effect. The current spacesuit design would provide shielding equivalent to 0.5 gm/cm^2 of aluminum, whereas the 10-to-20- gm/cm^2 aluminum shielding data are typical of current Skylab designs.

The allowable exposure for support personnel was established by MSFC as 40 rads/year. This included a 25-rad allowable from the RNS, leaving 15 rads from all other sources. Doses were computed assuming a full 24-hour stay time on orbit. Even taking this conservative approach, a total of 15 days/year of EVA time per man could be permitted without overexposure. Should the allowables be increased to 25 rads, a significant increase in mission time could be anticipated. On-orbit personnel could retreat to a shielded rest station for even lower exposures.

TOTAL SPACE RADIATION EXPOSURE

INDUCED RADIATION ENVIRONMENT

Gamma dose rates arising from the neutron activation of materials in the propulsion-module/engine-adaptor and tank-bottom regions have been estimated at various detector locations for various decay times after a 10-mission life. Mission frequency was assumed to be eight missions per year. These dose rates are shown on the facing page. The fractions of total dose due to each type of material are as follows:

<u>Material</u>	<u>Wt (lb)</u>	<u>Hours After Shutdown</u>			
		<u>10</u>	<u>100</u>	<u>1,000</u>	<u>10,000</u>
SS Alloy	341	29.3	32.1	77.3	8.6
Al Alloy	1,215	43.9	19.0	1.6	6.2
Inconel 718	50	0.4	16.3	21.1	7.8
Copper	100	26.4	32.6	--	--

At 10 hr after shutdown, Mn-56 with a half-life of 2.58 hr and Cu-64 with a half life of 12.8 hr account for approximately 98 percent of the 10-hr total dose rate. The stainless steel alloys have a maximum weight fraction of 2 percent Mn and the Al alloys have a weight fraction of 0.15 to 0.9 percent Mn and 0.4 to 6.8 percent Cu.

At 100-hr decay time, many radioactive isotopes are significant: Cu-64, 12.8 hr; Cr-51, 27.8 days; Fe-59, 45 days; Mo-99, 66 hr; Co-58, 71 days; Mn-54, 291 days; and Na-24, 15 hr. The latter two isotopes are products of fast-neutron reaction. The Na-24 isotope results from an (n, α) reaction with Al, while Co-58 and Mn-54 isotopes result from (n, p) reactions with Fe-54 and Mn-58, respectively. The dose rates from the stainless steel alloys are dominated by Cr-51 and Fe-59 isotopes, while Cu-64 and Na-24 dominate the dose rates from Al alloys.

At 1,000 hr, the predominant isotope is Co-58, followed by Mn-54; Cr-51 and Fe-59 isotopes are also significant. At 10,000 hr, the Mn-54 isotope becomes predominant, followed by Co-58. The Zn-65 isotope emerges as relatively significant, while Fe-69 is relegated to a role of minor importance.

INDUCED RADIATION ENVIRONMENT

DOSE RATE, * (RADS/HR)

DECAY TIME HOURS	10	100	1000	10,000
DETECTOR				
①	.038	.00024	.00006	.000005
②	3.097	.019	.005	.0004
③	36.80	.212	.108	.010
④	16.04	.075	.011	.001
⑤	15.75	.068	.008	.0009

* FOLLOWING 10 MISSIONS

COMPONENT RADIATION EFFECTS

A radiation effects analysis of the RNS insulation material and instrumentation (pressure, temperature transducers) was made during Phase II activities. This analysis indicated that no radiation effects problems could be anticipated in these areas. However, the equipment located down in the adapter and remote interface area was not sufficiently defined to include in this analysis.

Considering the criticality of the flyaway components and engine electronics, analyses were undertaken to determine the anticipated lifetime of these items. This analysis was made considering the lifetime requirements (10 missions) and the currently defined environment received from the reference lunar mission.

As anticipated, no significant radiation effects from gamma radiation is predicted for the components shown on the facing page. However, the flight electronics (semiconductor devices) will be affected by the neutron environment. These components, which include the multiplexers, decoders, etc., could not survive a complete mission even if the current designs were radiation-hardened. By using a neutron shield (polyethylene) of the dimensions and weight shown on the facing page, the sensitive components can perform for the required number of missions. Shielding in the sides is required because of scattering from the thrust structure. Reflections from the tank establish the top shielding requirements.

COMPONENT RADIATION EFFECTS

ADAPTER SECTION

	RADIATION ENVIRONMENT RADS/HR	INTEGRATED DOSE/MISSION RADS	DAMAGE THRESHOLD RADS	MAXIMUM ALLOWABLE RADS	SHIELDING REQUIREMENTS 10 MISSIONS	COMMENTS
BATTERIES	1×10^6	6×10^5	7×10^7	10^8	—	
STORABLE PROPELLANTS	1×10^6	6×10^5	$> 10^7$	$> 10^7$	—	2×10^3 BTU/LB/SEC
METAL SEALS	5×10^5	3×10^5	$> 10^8$	$>> 10^8$	—	
VALVE SEALS	5×10^5	3×10^5	$> 10^7$	$>> 10^8$	—	ASSUMING URETHANE OR ESTHANE TYPE SEAL
WIRING INSULATION	1×10^6	6×10^5	$> 10^8$	$>> 10^8$	—	ASSUMING ORGANIC INSULATION
TRANSDUCERS	1.5×10^6 $2 \times 10^{11} *$	1×10^6 $5 \times 10^{14} **$	$> 2 \times 10^7$ $> 10^{16} *$	$> 2 \times 10^7$ $>> 10^{16} *$	—	
INTEGRATED CIRCUITS	$2 \times 10^{11} *$	$5 \times 10^{14} **$		$2 \times 10^{14} **$	BOTTOM, 9.6" SIDE, 3.8" TOP, 0.67"	TO SHIELD RNS COMPONENTS WOULD REQUIRE ~320 LBS

* NEUTRON DOSE RATE, N/CM²/SEC

** NEUTRON INTEGRATED DOSE, NVT

EQUIPMENT ARRANGEMENT

The self-contained approach requires a separable engine interface adapter as an integral part of the engine thrust cone structure. All power, propulsion, control, and communications subsystems required for engine handling would be included in this adapter and separated with the engine when removal is required. The following design goals will govern concept analysis:

- Totally autonomous operation (i.e., no EVA or manual control)
- Simple maintenance/resupply
- RNS Systems commonality
- Power/weight efficiency
- Technical feasibility

EQUIPMENT ARRANGEMENT

INFLIGHT RADIATION ENVIRONMENT

The operating nuclear environment (both neutron and gamma) was defined for the RNS configuration shown on the facing page. This calculational model is typical of a manned lunar mission vehicle with a manned spacecraft forward of Sta. 2240, a payload frame forward of six propellant modules, and a propulsion module. Detector points were established in the centerline, and off-axis at locations coinciding with equipment locations or reference spacecraft locations.

As shown in the table on the facing page, the dose was computed as a function of propellant drain time. These data are presented by major source--e.g., direct and scatter. Direct dose d_2 is that dose received during the draining of the upper propellant with the propulsion module full. Direct dose d_1 is that received during the drainage of the propulsion module.

As can be seen, dose d_1 is affected directly by the residual in the propulsion module and the separation distance between the reactor and the detector point. Dose d_2 is only affected by separation distance.

The scatter contribution is a result of the side PVARA leakage streaming past the propulsion module and scattering on the upper propellant modules. As can be seen, the scatter contribution is not affected by a change in the direct dose d_1 .

To reduce the direct component, shielding between the reactor and the detector point must be added. To eliminate or reduce the scattered radiation component, a modified RNS configuration may be required--i.e., additional tankage below the propulsion module.

INFLIGHT RADIATION ENVIRONMENT

DETECTOR POINT	RESIDUAL	DOSE D ₁	DOSE D ₂	SCATTER DOSE	TOTAL DOSE
10	3000	24.0	5.0	7.0	36.0
	7280	12.4	5.0	7.0	24.6
11	3000	—	—	—	41.0
12	3000	31.8	7.3	12.7	51.8
	7280	16.7	7.3	12.7	36.7
13	3000	—	—	—	57.4
16	3000	70.0	16.0	75.0	161.0
	7280	36.3	16.0	75.0	128.0

— ALL DOSES IN REM

DETECTOR POINT	PROPELLANT MODULE					
	1	2	3	4	5	6
1	139	218	297	376	455	534
3	165	273	381	490	598	706
7	278	389	500	611	722	833
9	579	990	1402	1813	2234	2636

NOTE: D₁ - DIRECT DOSE DELIVERED DURING PROPULSION MODULE DRAIN
D₂ - DIRECT DOSE DELIVERED DURING PROPELLANT MODULE DRAIN

RADIATION ENVIRONMENT IMPLICATIONS

The baseline RNS configuration results in a 52 REM dose at the crew compartment location as shown on the previous charts. In order to reduce this dose level to the target level of 10 REM, supplemental shielding at the NERVA engine will be required.

Evaluation of this requirement has shown that a shield weight of 8100 lb would be required if the shielding were introduced as a disc shield located above the pressure vessel of the NERVA engine. Preliminary estimates show that this shield would consist of a central portion 33 in. in diameter and a peripheral region of 21.4 in. annular width. The central portion would weigh 3750 lb and the peripheral extension would weigh 4350 lb.

An alternate solution to this problem is to introduce the shield as a supplement to the current NERVA internal engine shield and add an external brim shield attached to the pressure vessel at a common plane to the internal shield. Such an installation would require 5900 lb total addition. This weight would be divided with 2500 lb allocated to the internal location and 3400 lb on the brim.

It is recognized that this internal/brim shield configuration has been dropped from current consideration by the NERVA program, but the 2200 lb weight advantage warrants its further consideration. As such, this brim shield approach is taken as the design for this study activity.

The total shielding weight can be further reduced by incorporation of supplemental tankage to decrease the subtended angle between the NERVA and the RNS. This approach would reduce the scattered contribution more strongly than the direct dose, and thus primarily effect the brim shield weight. This area will be evaluated further.

RADIATION ENVIRONMENT IMPLICATIONS

- BASELINE RNS CONFIGURATION REQUIRES 5900-LB SUPPLEMENTAL SHIELD FOR 10-REM DOSE AT CREW COMPARTMENT
- SMALL SUPPLEMENTAL TANKAGE DOES NOT AFFECT DIRECT DOSE SIGNIFICANTLY, BUT WILL REDUCE SCATTER CONTRIBUTION

THERMODYNAMIC ANALYSIS

A detailed thermodynamic analysis was conducted for the RNS. The analysis included the development of an analytical model to simulate the RNS in all mission phases. Boundary conditions were established, based on vehicle orientation throughout the mission and the effect of solar radiation and the lunar environment.

The environmental effects in conjunction with the basic vehicle characteristics were then fed into the Nuclear Vehicle Thermodynamic Optimization Program (NUTOP) in order to establish the design variables optimized for vehicle performance.

THERMODYNAMIC ANALYSIS

VEHICLE ORIENTATION

VEHICLE CONFIGURATION MODEL

BOUNDARY CONDITIONS

HEAT RATES

NUCLEAR VEHICLE THERMODYNAMIC OPTIMIZATION-NUTOP

THERMODYNAMIC MODEL

Since the external environment is the most significant boundary condition imposed on the thermodynamic analysis, an analytical model of the RNS was constructed. The model contains all external surfaces so that all external radiation and reflections can be determined.

A minimum of 17 surfaces were required to define the vehicle. Each surface was subdivided into elements so that variations in shading can be established. View factors are first obtained for the surfaces of the model, and then heat rates and radiation constants are calculated.

The internal structure was also defined and modeled into the thermal network. A thermal analyzer program was then utilized to calculate the temperature boundary conditions used in the NVTOP program.

THERMODYNAMIC MODEL

17 SURFACES
1277 ELEMENTS

VEHICLE ORIENTATION

Vehicle orientation throughout the mission is significant primarily to thermodynamics, power, communications, and trajectory characteristics. For thermodynamics and power generation, solar orientation is significant; for communication, Earth/Moon relationships are important; and for trajectory considerations, the relationship of the thrust vector to the flight path is significant.

During the main engine burns and impulse-utilized cooldown phases, the RNS longitudinal axis is along the flight path. During the impulse-spoiled phase, the RNS is oriented in such a manner that the payload alternately faces the sun and backs to the sun, unless thrust nullifiers are incorporated, and therefore the RNS flies a trajectory that undulates about an ideal path. During the linear orbit phase and during coast, the payload faces the sun.

VEHICLE ORIENTATION

LUNAR ORIENTATION

The lunar orbit environment will impose the most severe external heating on the vehicle. Studies were performed with a simplified model (cylinder) to determine which orientation would minimize the external heating. The model of the Moon assumes an albedo of 0.07 and a subsolar temperature of 250°F and a minimum cold-side temperature of -290°F .

Three vehicle orientations were investigated. In the first, the vehicle is space-oriented with the longitudinal axis parallel to the Moon-Sun line. In the other two cases, the vehicle is oriented axially and tangentially with respect to the Moon, and the longitudinal axis therefore changes with respect to the sun during orbit.

Incident heating rates from the Sun and Moon were obtained for various β angles. A calculation was then performed with different α/ϵ ratios to obtain surface temperatures; α_s/ϵ values 0.05/0.8 and 0.25/0.9 were used, representing optical surface reflectors and white paint, respectively.

Results indicate the first orientation is best for the following reasons:

- o Temperature is independent of surface properties
- o Average temperature for full range of β angles is low.
- o Solar panels will not have to track the sun since the vehicle is Sun-oriented.

LUNAR ORIENTATION

THERMODYNAMIC ENVIRONMENT

Once the model has been constructed, the heat-rate program will obtain external heating rates for the vehicle for each phase of the mission. During trans-lunar and trans-earth portions of the mission, a single instantaneous heat flux from the Sun is obtained and assumed to be constant for each of the three positions during their respective mission phases.

In lunar orbit, heat rates are obtained as a function of orbit position and then averaged over the entire orbit. All heat rates obtained take into account vehicle and lunar shading, as well as reflections of solar and lunar heat fluxes from one surface to another and solar energy reflected by the moon.

Subsequent to the heat-rate determination, the boundary conditions are established that consist of the external surface temperatures of each tank as a function of insulation thickness and mission time. The conduction heat leak boundary temperatures are the outer temperatures for the structural and line penetration connections that go to the H₂ tanks.

A set of boundary temperatures for various RNS elements of various mission conditions is presented.

THERMODYNAMIC ENVIRONMENT

	AVERAGE TEMPERATURE, °R				
	EQUIPMENT BAY	PROPELLANT TANK#3	PROPELLANT TANK#4	PROPULSION MODULE	ENGINE
EARTH ORBIT	360	408	410	400	395
TRANS-LUNAR COAST					
COOLDOWN	352	390	346	303	537
FACING SUN	372	147	164	35	111*
AWAY FROM SUN	167	147	163	253	180
LUNAR ORBIT	375	421	422	417	406

* NOZZLE IS 295°R

NUCLEAR VEHICLE THERMODYNAMIC OPTIMIZATION

This NUTOP computer program simulates a mission involving thermodynamic energy balances for coast phases, engine burns, cooldown, and afterburn equilibrium. The mission simulation is part of an optimization routine that can maximize payload or minimize launch weight. Any input can be an independent variable; however, most common variables are insulation thickness, vent pressure, design pressure, and propellant load.

The mission model that has been programmed for the nuclear vehicle can have up to 10 groups of propellant tanks with different thermodynamic states in each. Each of the groups can vent or operate independently at any pressure. During an engine burn, the tanks can drain together, and/or singly. After an engine burn, propellant can be transferred to the propulsion module and expended, tanks can be closed off and vented. Nuclear heating is included in the energy balance.

NUCLEAR VEHICLE THERMODYNAMIC OPTIMIZATION

THERMODYNAMIC OPTIMIZATION RESULTS

The thermodynamic optimization analysis utilized the detailed analytical model of the RNS. The complete mission was simulated with vehicle orientation scheduled according to the prescribed mission profile.

The first three tanks that are drained have a fixed insulation thickness and predetermined hydrogen saturation pressure. All of the other tanks have their insulation thickness individually optimized.

The vent pressure and consequently the maximum saturation pressure has also been optimized individually for each tank. The NPSP requirements, based on propellant saturation pressure at the beginning of each burn, have been determined for the malfunction mode operation of the NERVA engine.

The maximum operating pressure is based on the saturation pressure and NPSP values, and for the propellant modules, an 0.5 psi Δp value has been added for line losses. The track mode of pressurization was utilized, in which the operating pressure is a variable throughout the mission and is based on the saturation pressure of the active propellant module.

Vapor is dropped subsequent to depletion of each individual propellant tank. Consequently, the residual vapor weight is very small. A listing of total boiloff is also provided.

THERMODYNAMIC OPTIMIZATION RESULTS

LUNAR SHUTTLE MISSION

TANKS →	PROPELLANT MODULES				PROPULSION MODULE
	1-3	4	5	6	7
INSULATION THICKNESS, IN.	0.5	0.5	0.75	1.0	1.25
MAX. SATURATION PRESSURE, PSIA	18	22.5	24*	24*	24*
MAX. OPERATING PRESSURE, PSIA	23.9	24.9	25.4	25.4	24.9
DESIGN PRESSURE, PSIA	25.4	25.4	25.4	25.4	25.4
DROPPED VAPOR, LB	245	1141	1084		
RESIDUAL VAPOR, LB				927**	1089
BOILOFF, LB	2064				

* ALSO VENT PRESSURE

** DROPPED AFTER AE BURN

613

920

1227

ENGINEERING DRAWING TREE

The engineering drawing tree is the basis for providing graphic documentation with maximum visibility for the RNS in accordance with the WBS methodology. It also provides for the systematic build-up of each module and its final assembly into the RNS inboard profile.

ENGINEERING DRAWING TREE

STRUCTURAL FRAME -- INBOARD PROFILE

The space frame module serves as thrust structure, propellant module clustering structure, propellant module fluid distribution system, payload docking structure, orbit refueling docking structure and astrionic support structure. The total module is fully assembled, manufactured, and tested on the ground. The module is then brought into orbit as a unit by the Space Shuttle.

The space frame module contains the propulsive and intelligence capability and has its own station-keeping capability. The module is 15 ft. x 60 ft. in the stowed position, and extended to 15 ft. x 67 ft. with the equipment module extended by onboard module assembly equipment (bi-stem boom).

The equipment module consists of a power system, RCS system, environmental control system, guidance-navigation and control system, communications system, and data management system. The module is equipped with docking devices as well as propellant and RCS propellant transfer coupling devices. Vehicle orbital assembly equipment (bi-stem boom) is installed below the equipment module structure. Propellant module supports are located in forward and aft ends of the space frame. The six umbilical panels provide connections for the data management, propellant feed, pressurization, and power from the tank modules to the space frame. The propellant feed manifold interconnects to the propulsion module through an aft umbilical plate.

STRUCTURAL FRAME - INBOARD PROFILE

PROPELLANT MODULE -- INBOARD PROFILE

The propellant module consists of a propellant tank, propellant, insulation system, meteoroid protection system, and aft skirt structure; it incorporates tank supports, orbital assembly supports, equipment supports, and umbilical interface plates. This module also incorporates pressurization and propellant feed lines as well as electrical power distribution system and data management and instrumentation sensors. A propellant management system, purge system, ground emergency vent, flight vent, thermal conditioning unit and antislosh baffles are also included.

The module is structurally supported at the forward and aft ends. The aft structural attachment is effected by means of a 10-in.-diameter rigidly mounted cone with a space frame interface pin, and the forward attachment is effected by means of a 10-in.-diameter cone with an attachment pin that can slide axially on the space frame to allow for tank thermal effects at cryogenic temperatures.

PROPULSION MODULE-INBOARD PROFILE

ENGINE SYSTEM

The NERVA/RNS interface is defined on the facing page. This interface is a single plane that will permit installation of the engine on an adapter or thrust structure with standard couplings at the ground launch facility. This permits the NERVA development program and RNS development program to proceed with a minimum of interference and interdependence, maximum flexibility in the optimization of components, and efficient development of launch preparation and checkout procedures and equipment.

This approach to the definition of the interface is consistent with proven stage design practices developed and successfully demonstrated over the years. Several of the physical and functional considerations associated with this interface have been reviewed elsewhere in this presentation. During the remaining portion of the study, all material related to the interface will be collected in appropriate interface control drawings and an interface workbook.

ENGINE SYSTEM

PROPULSION MODULE -- INBOARD PROFILE

The propulsion module consists of a propellant tank and supports, propellant, and meteoroid and thermal protection systems. It also incorporates ground and emergency venting, flight ventings, pressurization system, propellant feed system, propellant conditioning system, purge system, data management system, environmental control system, electrical power, and instrumentation. The propulsion module also includes forward and aft skirts and an external fairing for the subsystem control wires. The forward docking ring transmits all of the loads through six connecting pins to the space frame module, and is structurally mounted to the forward skirt. The engine thrust structure provides for attachment of the engine flyaway control module to the propulsion module, and also transmits all of the engine thrust loads to the propulsion module aft skirt. The thrust structure is a 30-deg cone with skin- and stringer-stiffened construction. Aluminum alloy (7075-T6) is used for the entire thrust structure.

PROPELLANT MODULE - INBOARD PROFILE

RNS WEIGHT STATEMENT

The RNS weight statement provides a detailed weight breakdown to the WBS sixth level. The weights are grouped both by module and vehicle categories. The propellant module insulation weights are averaged although individual weights vary with tank number.

A Lunar Shuttle mission weight statement is also shown.

RNS WEIGHT STATEMENT - I

CODE	DESCRIPTION	WEIGHT, LB			VEHICLE*
		MODULES			
		PROPELLANT	PROPULSION	CONTROL & ASSY STRUCTURE	
2.00	STRUCTURE	(3,370)	(3,743)	(3,348)	(27,311)
	PROPELLANT TANK ASSEMBLY	2,904	2,818		20,242
	ENGINE THRUST STRUCTURE		501		501
	FORWARD SKIRT	203	215		1,433
	AFT SKIRT	203	114		1,332
	TUNNEL & FAIRINGS		30		30
	EXTERIOR FINISH & SEALER	25	25		175
	ANTI-SLOSH BAFFLES & SCREEN	25	25		175
	EQUIPMENT SUPPORT STRUCTURE	10	15	60	135
	EQUIPMENT MODULE STRUCTURE			540	540
	SPACE FRAME			2,748	2,748
3.00	METEOROID/THERMAL PROTECTION	(1,140.83)	(2,618)		(9,463)
	INSULATION	373.83	730		2,973
	METEOROID PROTECTION	767	1,888		6,490
4.00	DOCKING/CLUSTERING	(87.5)	(122)	(1,962)	(2,609)
	FWD DOCKING STRUCTURE		122	54	176
	AFT DOCKING STRUCTURE			105	105
	CLUSTERING STRUCTURE & MODULES ASSY EQUIP	87.5		1,803	2,328

* 6 PROPELLANT MODULE 1 PROPULSION MODULE 1 SPACE FRAME

RNS WEIGHT STATEMENT - II

CODE	DESCRIPTION	WEIGHT, LB			
		MODULES			VEHICLE
		PROPELLANT	PROPULSION	CONTROL & ASSY STRUCTURE	
5.00	MAIN PROPULSION	(242.5)	(34,962.8)	(667)	(37,084.8)
	NERVA ENGINE		27,798		27,798
	EXTERNAL DISC SHIELD FOR NERVA	—	5,900		5,900
	PURGE SYSTEM	70	69		489
	PROPELLANT CONDITIONING SYSTEM	25	25		175
	PROPELLANT FEED SYSTEM	84	280	498	1,282
	PRESSURIZATION SYSTEM	30	43.5	52	275.5
	FILL & DRAIN / ORBIT REFUELING	—	—	90	90
	GROUND & EMERGENCY VENT	31.5	31.5	25	245.5
	FLIGHT VENT	2	2	2	16
	ENGINE FLYAWAY CONTROL MODULE		813.8		813.8
6.00	AUXILIARY PROPULSION		(150)	(502)	(652)
	REACTION CONTROL SYSTEM		150	502	652

RNS WEIGHT STATEMENT- III

CODE	DESCRIPTION	WEIGHT, LB			
		MODULES			VEHICLE
		PROPELLANT	PROPULSION	CONTROL & ASSY STRUCTURE	
7.00	ASTRIONIC SYSTEM	(40)	(85)	(1,533)	(1,858)
	GUIDANCE, NAVIGATION & CONTROL	—	—	296	296
	INSTRUMENTATION & SENSORS	10	10	10	80
	COMMAND & CONTROL	—	—	254	254
	ELECTRICAL POWER	5	50	700	780
	ELECTRICAL NETWORKS	—	—	—	—
	ENVIRONMENTAL CONTROL	5	5	20	55
	PROPELLANT MANAGEMENT	—	—	—	—
	ON-BOARD CHECKOUT	—	—	—	—
	DATA MANAGEMENT	20	20	253	393
8.00	SAFETY/ORDNANCE SYSTEM	—	—	—	—
9.00	CONTINGENCY 5% (EXCLUDE NERVA ENGINE & DISC SHIELD)	(244)	(399)	(400)	(2,263)
	TOTAL INERT	5,124.83	42,079.8	8,412	81,240.8
10.00	RCS PROPELLANT			2,200	2,200
14.00	HYDROGEN PROPELLANT	38,676	37,371	—	269,427
	TOTAL WEIGHT	43,800.83	79,450.8	10,612	352,867.8

LUNAR MISSION WEIGHT SUMMARY—I

INITIAL WEIGHT		438832 LB
TLI LH ₂ CONSUMPTION	150120 LB	
ULLAGE GAS DROPPED	736 LB	
WEIGHT AT TLI COMPLETION		287976 LB
MIDCOURSE LH ₂ CONSUMPTION*	1259 LB	
BOILOFF—TRANS LUNAR COAST	52 LB	
RCS CONSUMPTION—TRANS LUNAR COAST	550 LB	
WEIGHT AT LOI INITIATION		286115 LB
LOI LH ₂ CONSUMPTION	31062 LB	
ULLAGE GAS DROPPED	1141 LB	
WEIGHT AT LOI COMPLETION		253912 LB
PAYLOAD DELIVERED TO LUNAR ORBIT	<u>85965 LB</u>	
*OUTBOUND MIDCOURSE ACCOMPLISHED WITH AFTERCOOLING

LUNAR MISSION WEIGHT SUMMARY—II

WEIGHT AT LUNAR ORBIT		167947 LB
BOILOFF IN LUNAR ORBIT	1173 LB	
RCS CONSUMPTION IN ORBIT	550 LB	
PAYLOAD LOADED IN LUNAR ORBIT	20114 LB	
WEIGHT AT TEI INITIATION		186338 LB
TEI LH ₂ CONSUMPTION	26881 LB	
ULLAGE GAS DROPPED	1084 LB	
WEIGHT AT TEI COMPLETION		158373 LB
MIDCOURSE LH ₂ CONSUMPTION	507 LB	
BOILOFF IN TRANS EARTH COAST	940 LB	
RCS CONSUMPTION IN COAST	550 LB	
WEIGHT AT EOI INITIATION		156376 LB
EOI LH ₂ CONSUMPTION	52459 LB	
WEIGHT AT EOI COMPLETION		103917 LB
PAYLOAD DELIVERED TO EARTH ORBIT	<u>20114 LB</u>	
LIQUID AND GAS RESIDUALS	2012 LB	
RCS PROPELLANT REMAINING	550 LB	
RNS DRY INERT WEIGHT		81241 LB

MISSION CHARACTERISTICS -- MARS OPPOSITION-CLASS MISSION

To assess the impact of manned planetary missions on the RNS design, two representative missions were analyzed: the 1986 Venus outbound swingby, an opposition-class mission departing Earth in May 1985 and taking advantage of a favorable Venus position for a gravity kick on the outbound leg; and a conjunction-class mission departing Earth in August 1990. The mission characteristics for the opposition-class mission are presented in the facing chart together with the payload requirements as defined in the study guidelines. The Planetary Mission Module (PMM) weight includes a 13-lb/day allowance for life support and RNS expendables. The velocity requirements include appropriate gravity loss allowances at Earth and Mars. In addition, ΔV allowances of 500 fps for midcourse correction on the trans-Mars and trans-Earth legs and 150 fps for Mars orbit maneuvering were included.

The mission analysis assumed a partially expendable interplanetary vehicle consisting of a baseline, seven-tank RNS augmented with expendable tankage mounted on two added spaceframes.

The interplanetary RNS was sized for Earth arrival and departure from a 24-hr elliptical orbit. Interorbital transfers from the 260-nm circular assembly orbit to the elliptical departure orbit were evaluated using the baseline seven-tank RNS performance capability.

MISSION CHARACTERISTICS

MARS OPPOSITION CLASS MISSION

18 TANK I/P RNS

BURN	JULIAN DATE	ΔV (FPS)	REF ORBIT ARR/DEP	LEG TIME (DAYS)
LE	2446150	4177	24 HR ϵ $\oplus$	350
AM	2446500	12229	12 HR ϵ $\odot$	
LM	2446560	7422	12 HR ϵ $\odot$	60
AE	2446710	3273	24 HR ϵ $\oplus$	150

PAYLOAD ITEM	WEIGHT (LB)
PMM (LE WEIGHT)	145,000
MEM	100,000
MARS PROBES	30,000
VENUS PROBES	6,000
MEM/PROBE HANGAR	5,500
MARS SAMPLES	1,500

MISSION WEIGHT SUMMARY -- MARS OPPOSITION-CLASS MISSION

Approximately 1.5×10^6 lb of propellant must be delivered to the 260-nm assembly orbit to conduct the 1986 Venus outbound swingby mission in the selected mode, slightly more than half of which is used by the interorbital (I/O) shuttle in transferring the interplanetary (I/P) RNS and payload to the elliptical departure orbit. Boiloff, vapor blowdown, and residuals were determined using LMSC's Nuclear Vehicle Thermal Optimization Program (NVTOP) and reflect common tank and insulation design (except for the tanks depleted during the leave-earth burn). Performance analysis included startup, shutdown, and cooldown effects and assumed 75 percent recovery of the pulsed-flow cooling impulse.

The I/O propellant requirements are based on coplanar transfer by a baseline 7-tank RNS. Because of the differential regression rates of the elliptical and circular orbits, coplanar opportunities will occur at approximately 56-day intervals. Non-coplanar transfers can impose severe ΔV penalties except for transfers very near the coplanar opportunity, or for the case where the line of apsides of the ellipse is coincident with the line of intersection for the two orbit planes. This latter condition also occurs only once every 56 days, but by varying the time between the last coplanar transfer and the earth-departure burn, it can be positioned time-wise from the coplanar opportunity such that I/O trips can be accomplished in 56 days with only one non-coplanar transfer. If turnaround time for the I/O RNS is on the order of 8 - 10 days, the offloaded trip No. 2 could probably accomplish the non-coplanar transfer if its full propellant capacity were used.

If two I/O RNS vehicles are available, the complete transfer could be accomplished in the turnaround time of one RNS plus the time away from low orbit and the cooldown time after return to the assembly orbit.

MISSION WEIGHT SUMMARY

MARS OPPOSITION CLASS MISSION

TOTAL PROPELLANT REQUIRED AT 260 NM
CIRCULAR ORBIT = 1,468,545 LB

INTER-ORBITAL SHUTTLE TRANSFERS			
TRIP	PAYLOAD	P/L WT (LB)	I/O W _P (LB)
1	MEM+HGR+VENUS PROBES+6 TK TIER	374,784	260,000
2	I/P RNS (7 TANK)	357,279	249,000
3	PMM+MARS PROBES + 5 TK TIER	390,771	269,400

MARS MISSION SUMMARY			
PROPELLANT EXPENDED		NON-PROPELLANT ITEMS	
ITEM	WEIGHT (LB)	ITEM	WEIGHT (LB)
IMPULSE PROPELLANT	660,498	LE INERT WEIGHT	146,189
COOLDOWN LOSSES	6,530	LE PAYLOAD WEIGHT	286,500
VAPOR BLOWDOWN	7,033	5 TANK EXP. TIER	27,109
BOIL OFF	13,887	6 TANK EXP. TIER	31,228
RESIDUALS	2,197	LIFE SUPPORT EXP.	7,280
TOTAL PROPELLANT	690,145	MARS SAMPLES	1,500

TANK CHARACTERISTICS -- MARS OPPOSITION-CLASS MISSION

The interplanetary RNS vehicle for the 1986 Venus outbound swingby mission would require 18 propellant tanks. Eleven of these tanks will be located in two expendable tiers, the first of which contains 5 tanks and is jettisoned during the trans-Mars coast. The second tier contains 6 tanks and is left in Mars orbit. For this configuration, all propellant tanks would be common in design pressure and insulation, except for reduced insulation on the four tanks emptied during the leave-Earth burn. The remaining tanks, being common, would permit maximum flexibility in order of draining.

TANK CHARACTERISTICS

MARS OPPOSITION CLASS MISSION

EXPENDABLE
5-TANK TIER

EXPENDABLE
6-TANK TIER

BASELINE
7-TANK TIER

PROPULSION
MODULE

TANK NO.	INSULATION THICKNESS (IN.)	OPERATING PRESSURE (PSI)
1	2 1/4	24.8
2 - 5	2 1/4	25.3
6 - 13	2 1/4	25.8
14	2 1/4	26.3
15 - 18	1/2	25.0

MISSION CHARACTERISTICS -- MARS CONJUNCTION-CLASS MISSION

The 1990 conjunction-class mission also assumed a partially expendable I/P RNS operating out of 24-hr elliptical earth orbit; however, because the energy requirements for this mission are considerably less than for the 1986 swingby mission, only three expendable tanks in a single jettisonable tier are required. Delta-V requirements listed include gravity loss at Earth and Mars. Midcourse correction ΔV allowances of 500 fps for the trans-Mars and trans-Earth legs and for 300-fps Mars orbit maneuvering were also included in the evaluation.

MISSION CHARACTERISTICS

MARS CONJUNCTION CLASS MISSION 10 TANK I/P RNS

BURN	JULIAN DATE	ΔV (FPS)	REF ORBIT ARR/DEP	LEG TIME (DAYS)
LE	2448120	4119	24HR $\oplus$	340
AM	2448460	3722	12HR $\oplus$	360
LM	2448820	4339	12HR $\oplus$	300
AE	2449120	2806	24HR $\oplus$

PAYLOAD ITEM	WEIGHT (LB)
PMM (LE WEIGHT)	168,000
MEM (2)	200,000
PROBES	30,000
MEM/PROBE HANGAR	11,500
MARS SAMPLES	1,500

MISSION WEIGHT SUMMARY -- MARS CONJUNCTION-CLASS MISSION

For the 1990 conjunction-class mission, approximately 1.0×10^6 lb of propellant must be delivered to the 260-nm assembly orbit; approximately 64 percent of this propellant is used by the I/O RNS in performing the three transfers to the elliptical departure orbit. The I/O trips are considerably off-loaded; however, since the total weight to be transferred exceeds the full capability of two RNSs, the third trip cannot be eliminated unless the departure orbit is changed or the I/O shuttle performance increased. If the I/O RNS propellant capacity is increased to about 325 K lb, the transfer could be accomplished in two trips, although the total mission propellant required in low orbit would not be greatly changed.

MISSION WEIGHT SUMMARY

MARS CONJUNCTION CLASS MISSION

**TOTAL PROPELLANT REQUIRED AT 260 NM
CIRCULAR ORBIT = 996,001 LB**

INTER-ORBITAL SHUTTLE TRANSFERS			
TRIP	PAYLOAD	P/L WT. (LB)	I/O Wp (LB)
1	PMM + EXP 3 TK TIER	287,402	212,000
2	MEM'S + HGR + PROBES	241,500	186,000
3	I/P RNS (7 TANK)	337,722	238,000

MARS MISSION SUMMARY			
PROPELLANT EXPENDED		NON - PROPELLANT ITEMS	
ITEM	WEIGHT (LB)	ITEM	WEIGHT (LB)
IMPULSE PROPELLANT	336,305	LE INERT WEIGHT	97,123
COOLDOWN LOSSES	3,766	LE PAYLOAD WEIGHT	409,500
VAPOR BLOWDOWN	5,147	3 TANK EXP. TIER	18,828
BOILOFF	13,225	LIFE SUPPORT EXP.	13,000
RESIDUALS	1,558	AE PAYLOAD WEIGHT	155,000
TOTAL PROPELLANT	360,001	MARS SAMPLES	1,500

TANK CHARACTERISTICS -- MARS CONJUNCTION-CLASS MISSION

The 1990 conjunction-class mission requires a 10-tank I/P RNS configuration; three of the tanks are located in an expendable tier and are jettisoned after the leave-Earth burn. Thermal analysis for this mission suggests that tank design pressure and insulation thickness could be "tuned" to meet mission requirements; however, since the I/P vehicle is off-loaded by approximately 20,000 lb, the additional weight to provide common tank design could be tolerated with only a modest increase in propellant delivered to the assembly orbit.

TANK CHARACTERISTICS

MARS CONJUNCTION CLASS MISSION

**BASE LINE
7-TANK TIER**

**PROPULSION
MODULE**

**EXPENDABLE
3-TANK TIER**

TANK NO.	INSULATION THICKNESS (IN)	OPERATING PRESSURE (PSI)
1	3 1/4	26.5
2	2 1/2	27.0
3	1 3/4	27.0
4 - 7	1	25.7
8 - 10	1/2	24.0

ORBITAL ASSEMBLY

INITIAL ASSEMBLY SEQUENCE

The multiple-tank concept of the RNS requires that final assembly of the stage take place in earth orbit. The general sequence of events followed in conducting this assembly is shown on the facing chart. In the Lockheed concept, the first module to be deployed is the space frame, which is a self-contained, autonomous system. The tanks are then individually delivered by the Space Shuttle and transferred to the RNS assembly in sequence until the completed system is achieved.

This general process has been established in previous phases of the study. The information contained in this presentation covers the recent analysis associated with the feasibility and design considerations of this assembly.

INITIAL ASSEMBLY SEQUENCE

SPACE FRAME

DEPLOYMENT

TANK INSTALLATION

ENGINE INSTALLATION

SPACE FRAME MODULE DEPLOYMENT

The first module delivered by the Space Shuttle for RNS assembly is the space frame. This module is delivered to the desired assembly orbital location, fully checked out by the Space Shuttle, and then released.

The primary cargo delivery concept used in this study is one in which cargo is moved laterally out of the Space Shuttle cargo bay. The forward-hinged, swing-out delivery was also evaluated and the RNS modules are compatible with either technique.

The procedure followed in deploying the space frame is as follows:

- The space frame is carried in the Space Shuttle with all equipment in the stowed position. Following arrival at the desired orbital location, the space frame will be moved out of the cargo bay but remain attached to the Space Shuttle.
- After checkout update, the equipment module on the space frame will be moved to its operational position, latched, and functionally checked out again.
- When final checkout has been completed, the RNS space frame will be released and begin its operational functions.
- The solar arrays will be deployed and the power system fully activated, followed by deployment of the antenna arrays.

The space frame is a self-contained, autonomous spacecraft, and is an active member in all remaining steps of the assembly process.

SPACE FRAME MODULE DEPLOYMENT

RNS / EOS ACQUISITION

Following deployment of the space frame, the Space Shuttle will begin delivery of the remaining propellant and propulsion modules for completion of the RNS assembly. On each return flight to the RNS, the Space Shuttle will establish communication with the space frame and the acquisition process will follow the sequence shown on this chart.

The docking function shown on the chart may be either a soft or hard dock, depending on the final design procedure selected. During the final approach procedure, however, a common stabilization of both vehicles will be performed. This is of particular significance in the soft dock approach.

RNS / EOS ACQUISITION

MORE THAN 75 MILES

- INERTIAL GUIDANCE SYSTEM
- TRACKING

75 MILES TO 2 MILES

- LASER RADAR SYSTEM
 - LONG RANGE MODE
 - RANGE ACCURACY - 1/2 PERCENT

2 MILES TO DOCKING

- LASER RADAR SYSTEM
 - SHORT RANGE MODE
 - RANGE ACCURACY $\pm .02\%$ OR ± 10 CM
(WHICHEVER IS GREATER)

STABILIZATION

- ALIGN THE INERTIAL REFERENCE OF BOTH VEHICLES SIMULTANEOUSLY TO A COMMON REFERENCE
- RNS CONTROL ACCURACIES
 - POSITION $\pm .5$ FT
 - VELOCITY $< .05$ FT/SEC
 - ANGLE $\pm .2^\circ$
 - ANGULAR RATE $< .02^\circ$ /SEC

RNS / EOS ASTRIONICS

The interaction of the astrionics systems of the RNS and the Space Shuttle during the rendezvous and terminal maneuvers are shown on this chart. Primary reliance is placed on the laser system for precision data related to the relative motion and position of the two vehicles.

The Space Shuttle will act as the master system during these operations. The communications link will transfer RNS guidance sensor data to the Space Shuttle and command and control data from the Space Shuttle to the RNS. Following position reference and guidance updating, the Space Shuttle will utilize the radar link established by its laser system and the RNS corner reflectors to maintain positional input information. Total operational control will be performed by the two-vehicle command program within the Space Shuttle guidance software. This approach will result in the positional accuracies shown in the preceding figure for the stabilization phase.

RNS / EOS ASTRIONICS

PROPULSION MODULE DEPLOYMENT

The propulsion module is installed on the RNS axis at the aft end of the space frame in the manner indicated on this chart. The Space Shuttle will move to the position shown on the figure, using the stabilization mode previously described.

The propellant module is moved out of the Space Shuttle cargo bay and positioned for attachment of the RNS mounted assembly booms. Following attachment of the booms and after final checkout from the Space Shuttle, the module is released from the Shuttle support and drawn into the RNS frame by the assembly booms. Assembly into the frame is accomplished by a docking system consisting of six pins that engage the six longeron members of the space frame. Final alignment and structural joining are achieved by motor-driven bolts mounted in the space frame longerons.

Following structural attachment, the umbilical connection between the space frame and the propulsion module is completed. This umbilical carries the main feed line, pressurization line, RCS fluid lines and the astrionics connections. All connections except the main LH₂ feed line are made in the process of connecting the umbilical plates. This operation also aligns and positions the feed line. Final sealing of this line is then performed by a motor-driven Marmon clamp which completes the line connection and sets the seal. At this point, the propulsion module is in place and under the control of the RNS system.

PROPULSION MODULE DEPLOYMENT

ENGINE DEPLOYMENT

The NERVA engine is installed on the RNS by an approach similar to that used in the propulsion module installation. The Space Shuttle delivering the engine unit is positioned aft of the propulsion module, the NERVA engine and the orbital assembly adapter are readied for delivery to the RNS, and the final checkout of the system is completed by the Space Shuttle.

The orbital assembly adapter contains a "flyaway" unit within its system. This unit is used to perform the translation from the Space Shuttle cargo bay to the aft docking system on the propulsion module. This flyaway unit is incorporated into the engine assembly in order to provide the capability to remove the engine from the RNS at later times as a means of providing radiation environment control.

The engine assembly is docked to the thrust structure on the propulsion module through a conical docking system. Following structural tiedown, the umbilical connections are completed in a manner similar to that for the propulsion module assembly, namely, the plate and all connections other than final sealing of the propellant feed lines are completed; the last step is the sealing of the feed lines with Marmon clamps.

ENGINE DEPLOYMENT

PROPELLANT MODULE DEPLOYMENT

The reference, or baseline, approach to assembling the propellant modules is shown on this chart. This approach uses the technique of linear translation of the module from the Space Shuttle cargo bay to the final installed position in the space frame. The Space Shuttle can position the tank in the range of 5 ft. to 40 ft. from the space frame. This range in positioning is dependent upon the final configuration of the orbiter. A delta body orbiter would permit closer approach, because of absence of empennage interference.

The transfer approach follows the basic procedure previously described in that the module is positioned by the Space Shuttle, the RNS assembly booms are attached (two forward and two aft), the module undergoes final checkout by the Space Shuttle and then is released to the RNS. The module is then drawn into the space frame by the booms and positioned for structural attachment.

The structural tiedown is accomplished by male conical pins located on the space frame mating with female adapters located on the module tank skirts. There are two attachments, one on each end of the module. The aft connection is rigid and represents the primary tiedown point. The forward connection is installed in the space frame to provide axial motion for thermal expansion compensation. The structural joint is completed by a motor-driven Marmon clamp on each of the two connectors.

Following structural tiedown, the umbilical connection is made in a manner similar to that described previously. The umbilical is located on the aft end of the module, adjacent to the fixed structural joint, to minimize the effects of any relative motion.

PROPELLANT MODULE DEPLOYMENT

PROPELLANT MODULE DEPLOYMENT (ALTERNATE)

An alternate approach to RNS assembly based upon the technique of a hinged, rotating deployment system for cargo handling within the Space Shuttle has been evaluated. In this approach, the modules are pivoted out of the cargo bay and are delivered to the RNS in a vertical position. This handling technique would permit direct emplacement of the propulsion module onto the space frame either with or without use of the assembly booms. The propellant modules would be installed in the manner shown on the facing chart.

The Space Shuttle positions the tank as shown and then moves into the transfer position with the RNS. The geometry of this position is such that if the Space Shuttle were to be equipped with a docking adapter, the RNS/Shuttle could hard dock and the propellant module transfer could take place in a closed system. In either event, the overall assembly can be accomplished with either method of Space Shuttle cargo deployment.

PROPELLANT MODULE DEPLOYMENT (ALTERNATE)

SPACE FRAME / TANK ORBITAL ASSEMBLY

This chart presents the basic tolerances established in the design of the RNS assembly equipment. The broadest tolerances exist for the boom attachment operation and are compatible with the stabilization accuracy of the RNS. The boom attachment mechanisms for the forward end of the propellant module are provided with additional axial tolerance to compensate for thermal growth of the tank.

As the tank is drawn into the spaceframe, the positional accuracy is increased as the boom length decreases. As the tank is positioned on the support cones and readied for structural attachment, the installation accuracy is improved by the conical surface on the fittings. Final structural locking is accomplished by Marmon clamps.

The umbilical plate is driven by motorized screw jacks. Initial engagement is by four centering pins which move from 0.5 in. tolerance to 0.03 in. as the pins seat. The electrical and pressurization connections are made at the end of the plate movement. These connections in turn have centering keys and are mounted with one side fixed and the other floating, to ensure proper seating with no damage.

The main hydrogen feed line is sealed after all other motion has been completed. As the umbilical plates join and are latched, the feed line centers and mates by means of a standard male/female flange fitting. Following latching of the umbilical, the feed line is sealed by a Marmon clamp, motor-driven, and a seal such as the standard Cono seal. This ensures a high degree of system integrity and completes the assembly.

SPACE FRAME / TANK ORBITAL ASSEMBLY

OPERATIONAL STEP	FUNCTION	TRAVERSE MOTION	POSITIONAL TOLERANCE
1	ATTACH BOOM TO TANK	5'-40'	$\pm 3.0''$ (Y,Z) $\pm 5''$ (X)
2	POSITION TANK IN SPACE FRAME		$\pm 1.0''$ (INITIAL) $\pm 0.3''$ (FINAL)
3	ATTACH TANK TO SPACE FRAME		$\pm 0.01''$ (MARMON CLAMP)
4	ENGAGE UMBILICAL PLATE	6.0"	0.5" (INITIAL) 0.03" (FINAL)
5	CONNECT ELECTRICAL & PRESSURIZATION LINE		0.015"
6	SEAL LH ₂ LINE		$\pm 0.01''$ MARMON CLAMP/ CONOSEAL

TRANSFER LOADS ANALYSIS

A critical element in the RNS assembly operation is the loading of the assembly or transfer booms under adverse conditions. The chart on the facing page presents a series of cases involving inadvertent loading of the boom by Space Shuttle thruster malfunction. These conditions are the worse case situations for each of the basic transfer operations. The Space Shuttle malfunction effect is more severe than RNS thruster failures, because of the greater thrust rating of the Space Shuttle motors.

Those analyses show that for the severe cases considered in which thruster failure exists for periods on the order of 10 secs, only the propulsion module transfer associated with linear cargo deployment, presents a failure mode condition in the booms.

TRANSFER LOADS ANALYSIS

PROPULSION MODULE

THRUSTER A OPERATING @1500 LB FOR >10 SEC = F_A
 F_A CAUSES $M_R = 130 \text{ K IN-LB}$
 F_B CAUSES $M_R = 29 \text{ K IN-LB}$

PROPULSION MODULE (ALTERNATE)

F_A CAUSES $M_R = 52 \text{ K IN-LB}$

PROPELLANT MODULE

F_A CAUSES $M_R = 72 \text{ K IN-LB}$

PROPELLANT MODULE (ALTERNATE)

F_B CAUSES $M_R = 42 \text{ K IN-LB}$

TRANSFER BOOM CHARACTERISTICS

This chart presents the general strength characteristics of the SPAR bi-stem boom used in evaluation of the assembly process. The critical bending moment is the controlling factor in evaluating the assembly procedure. It is of importance to note that a critical bending failure is not destructive, but rather will reduce the life cycle expectancy. As a result, the assembly process as developed to date in this study is feasible.

TRANSFER BOOM CHARACTERISTICS

- SIZE- 5 IN. DIAMETER, 40 FT LONG
- PERFORMANCE- TENSILE STRENGTH 25,000 LB
CRITICAL BENDING MOMENT 70,000 IN.-LB
COMPRESSIVE STRENGTH @ 40 FT 650 LB
@ 20 FT 2500 LB
- FAILURE MODE
 - CRITICAL BENDING FAILURE FOR 40 FT OCCURS AT $\delta=9.5^\circ$
 - SINGLE UNBALANCED 1500 LB THRUSTER TAKES 3 SEC TO DEFLECT 9.5° (UNCOUPLED)
 - FULL MOMENT DEVELOPED IN BOOM TAKES 5-10 SEC
 - CRITICAL BENDING FAILURE CAN REDUCE BOOM LIFE BUT IS NOT DESTRUCTIVE

SUPPORT STRUCTURE DEPLOYMENT -- MODULAR PAYLOAD INTEGRATION

The approach for payload integration into the modular RNS that has been developed by Lockheed is the utilization of the space frame concept. This approach is based on the consideration that the majority of all cargo handled by the RNS will have been delivered on modular packages by the Space Shuttle or be compatible with modular packaging.

The use of the space frame concept for cargo handling will also add an additional degree of freedom to the operation of the RNS. Since this frame would be set to handle 15 x 60 ft. modules, it could be used to accommodate supplemental propellant modules for missions in which payload delivery requirements might exceed the nominal capability. In addition, RNS propellant modules could be temporarily stored in the cargo bay during a tank exchange procedure, in order to minimize procedural problems. The cargo bay space frame could contain its own subsystems or share functions with the RNS equipment. This concept should receive additional study to validate its utility.

SUPPORT STRUCTURE DEPLOYMENT, MODULAR P/L INTEG

PRESSURIZATION ANALYSIS

TANK PRESSURIZATION -- TASK 9.3

The primary objective of this task was to determine optimum RNS tank operating pressure levels and corresponding engine bleed gas flowrates. Maximum pressurant flowrates were determined as a function of pressurization Δp requirements and initial tank ullage volume for each burn. Optimum operating pressure levels were determined using the NUTOP (Nuclear Thermodynamic Optimization Program) for four RNS configurations for the reference lunar mission, abort mode. Engine NPSP was assumed to be that required for operation in the MALF (turbopump malfunction) mode. Optimization runs were made in each case for two pressurization modes, the tracking and the step modes. In the tracking mode, the active tank is pressurized to a level determined by the NPSP corresponding to the current propellant vapor pressure in that tank. In the step mode, each tank is pressurized to a fixed operating pressure level for all burns (except midcourse burns). A single NUTOP run was made for the LMSC baseline configuration. This run was made for the lunar abort mission (inbound payload: 20,000 lb), operating with NPSP valves for the normal pressurization mode. In order to accurately estimate residual vapor weights at the end of the earth-arrival burn, this burn was simulated in two segments. The first segment simulated the startup, steady-state and shutdown operation for which active pressurization of the tank is required. The second segment simulated the unpressurized drain of the tank during which useful impulse was obtained. At the end of the second segment it was required that a minimum of 660 lb of propellant be available for non-impulsive cooldown of the engine.

TANK PRESSURIZATION – TASK 9.3

- DETERMINE MAXIMUM PRESSURANT FLOW RATE REQUIREMENTS
- COMPUTE REQUIRED OPERATING PRESSURE LEVELS FOR REPRESENTATIVE RNS CONFIGURATIONS
- DETERMINE SENSITIVITY OF OPTIMUM OPERATING PRESSURE LEVEL TO VARIATIONS IN ENGINE OPERATING MODE AND TANK PRESSURIZATION MODE

ENGINE OPERATING MODE PRESSURIZATION CONFIGURATION	MAF	MAF	NORMAL
	TRACK	STEP	TRACK
LMSC SINGLE TANK	X	X	
LMSC MULTIPLE TANK	X	X	X
NAR SINGLE TANK	X	X	
MDAC CLASS I HYBRID	X	X	

MAXIMUM PRESSURANT FLOW RATES DURING STARTUP

A simplified energy balance of the ullage vapor in the tank yields an approximate relation for the mass required to effect initial pressurization. Assuming that the maximum pressurant flow rate, $\dot{m}_{p_{\max}}$, is proportional to the mass of pressurant required for initial pressurization, this relation can be written

$$\dot{m}_{p_{\max}} = K_1 \Delta p V_{u_i} + K_2$$

where Δp is the difference between the initial propellant saturation pressure and the operating pressure, V_{u_i} is the initial ullage volume, and K_1 and K_2 are constants. Using the maximum pressurant flow rates predicted by the propellant heating computer program, and fitting the computed points with a straight line, the relation

$$\dot{m}_{p_{\max}} = \frac{\Delta p V_{u_i}}{68667} + 0.35$$

is obtained for both the single tank and the baseline propellant module.

MAXIMUM PRESSURANT FLOWRATES

DURING STARTUP

NERVA TURBOPUMP OPERATING REQUIREMENTS

The normal-mode NSP requirements were developed as a function of saturation pressure of the outflow propellant at full thrust. For malfunction mode operation, the minimum thrust level for negligible performance degradation is 65 percent of rated thrust. Using the malfunction-mode NSP requirements for operation at the 65-percent level produces malfunction-mode NSP requirements nearly identical with the normal-mode requirements. This allows the engine user considerable flexibility in vehicle design and operation.

NERVA TURBOPUMP OPERATING RQMTS

LMSC SINGLE-TANK RNS

The LMSC single tank configuration is designed for a nominal propellant load of 300,000 lb with 5-percent ullage volume. Multilayer insulation conductivity is a variable that depends on insulation thickness and hot and cold boundary temperatures. The fixed inert weights used as input to the NUTOP were broken down as follows:

• Structure (Base structure plus main propulsion structure)	5490 lb
• Meteoroid Protection	6560 lb
• NERVA Engine	27798 lb
• Auxilliary Propulsion	420 lb
• Astrionics	1527 lb

The weight of RCS propellant used during the mission was 2200 lb. The total inert weight for this configuration included a 10-percent contingency.

LMSC SINGLE-TANK RNS

- PROPELLANT LOAD = 300,000 LB LH₂
- TANK MATERIAL IS 2021 ALUMINUM
- FIXED INERT WEIGHT = 48,778 LB
- VARIABLE INERT WEIGHTS ARE TANK WEIGHT AND INSULATION WEIGHT
- OPTIMIZATION VARIABLES ARE INSULATION THICKNESS, TANK OPERATING PRESSURE, AND VENT PRESSURE
- MULTILAYER INSULATION CHARACTERISTICS

$$K \doteq 3.6 \times 10^{-5} \text{ BTU/HR-FT}^2\text{-R}$$
$$\rho = 2.3 \text{ LB/FT}^3$$

LMSC SINGLE TANK -- OPTIMIZED DESIGN DATA

The results for the single tank illustrate the differences due to operation in the tracking and the step mode of pressurization. In the track mode, the tank operating pressure increases with each successive burn until the propellant vapor pressure reaches the vent pressure level. This point is reached at the end of the moon-arrival burn. Consequently, less hot pressurant gas is added to the tank with operation in the track mode and the boiloff (vent gas) losses are reduced. In order to best utilize this advantage, the optimum point for the track mode is at a slightly higher vent pressure than for the step mode, resulting in greater residual vapor weight than in the step mode but with considerably reduced boiloff. However, further increases in vent pressure result in increases in inert weight that outweigh the advantages caused by reduced boiloff.

As shown earlier, the maximum pressurant flowrates are proportional to the product of the ullage volume and the difference between the operating pressure and the propellant vapor pressure at the start of each burn. For the mission considered, this product is a maximum for the moon-arrival burn. The propellant vapor pressure at the start of this burn is still below the vent pressure level. Thus the Δp supplied in the track mode is less than that supplied in the step mode, and the maximum pressurant flowrates for the track mode are correspondingly less.

LMSC SINGLE TANK-OPTIMIZED DESIGN DATA

	<u>MALE TRACKING MODE</u>	<u>MALE STEP MODE</u>
INSULATION THICKNESS, INCHES	1.25	1.25
DESIGN PRESSURE, PSIA	26.25	26.00
OPERATING PRESSURE, PSIA	23.38 TO 26.25	26.00
VENT PRESSURE, PSIA	26.25	26.00
TOTAL INERT WEIGHT*, LB	72,419	72,218
BOIL-OFF LOSSES, LB	9,445	10,459
RESIDUAL VAPOR WEIGHT, LB	9,975	9,886
MAXIMUM PRESSURANT FLOW RATE, LB/SEC	2.75	3.75

*INCLUDES 10% CONTINGENCY

LMSC MULTIPLE-TANK RNS

The LMSC multiple-tank configuration is designed for a nominal propellant load of 269,427 lb, which is based on 38,676 lb in each propellant module and 37,371 lb in the propulsion module. The propellant level in the propulsion module is maintained at 90 percent of the tank volume until the last burn, during which the tank is depleted. The fixed inert weights used as input to the NUTOP were broken down as follows:

• Structure (Base structure plus clustering & propulsion structure	6722 lb
• Meteoroid Protection	5360 lb
• NERVA Engine	27798 lb
• Auxilliary Propulsion	415 lb
• Astrionics	1567 lb

The weight of RCS propellant used during the mission was 2200 lb. The total inert weight included a 10-percent contingency.

LMSC MULTIPLE-TANK RNS

- PROPELLANT LOAD = 269,427 LB LH₂
- TANK MATERIAL IS 2021 ALUMINUM
- FIXED INERT WEIGHT = 48,449 LB
- VARIABLE INERT WEIGHTS ARE TANK WEIGHT AND INSULATION WEIGHT
- OPTIMIZATION VARIABLES ARE VENT PRESSURE ON TANKS 5, 6, AND 7, INSULATION THICKNESS ON TANKS 5, 6, AND 7, AND MINIMUM OPERATING PRESSURE
- INSULATION THICKNESS ON TANKS 1-3 IS 0.5 INCHES; VENT PRESSURE IS 18 PSI
- PROPULSION MODULE ULLAGE FRACTION MAINTAINED AT 10%

LMSC MULTIPLE TANK -- OPTIMIZED DESIGN DATA

The optimized design points for engine operation in the malfunction mode are virtually identical for the track and the step pressurization modes. The optimum values of vent pressure for this configuration are near the minimum allowable value (18 psia). This is a result of the low boiloff values inherent in the use of small, multiple propellant tanks rather than a single, large tank. In addition, the vapor in each propellant module is completely vented (dropped) when the liquid supply is depleted, so that the vapor is not carried throughout the mission. A comparison between significant vapor weights for the LMSC single- and multiple-tank configurations is shown below:

Propellant Vapor Weights for MALF Mode, Track Pressurization

Configuration Item	LMSC Single Tank	LMSC Base- line 7-Tank
Boiloff Weight (lb)	9,445	4,224
Residual Vapor Weight (lb)	9,975	918
Dropped Vapor Weight (lb)	0	3,620

For both pressurization modes, the maximum pressurant flowrate is required for the moon-arrival burn during pressurization of Tank No. 4. This tank is also pressurized during the first (leave-earth) burn. As a result, the propellant in this tank is saturated at the vent pressure level by the beginning of the moon-arrival burn. Thus, no difference is observed in maximum pressurant flowrates required in the step or track modes.

Tank operating pressure levels are slightly reduced for engine operation in the normal mode, because of the lower NPSP required in this mode of operation. This allows a corresponding reduction in tank design pressure with resulting smaller tank weights. The reduced boiloff (because of less pressurant addition to the tanks) results in an optimized vent pressure value for Tank 7 that is 0.5 psi lower than that for operation in the MALF mode. A very small (0.01 lb/sec) reduction in maximum pressurant flowrate is observed because of the lower Δp required at the start of the moon-arrival burn.

LMSC MULTIPLE TANK-OPTIMIZED DESIGN DATA

	MALF TRACKING MODE	MALF STEP MODE	NORMAL TRACKING MODE
INSULATION THICKNESS, TANKS 4-7, INCHES	1.00	1.00	1.00
VENT PRESSURE, TANKS 4-6, PSIA	20.00	20.00	20.00
VENT PRESSURE, TANK 7, PSIA	20.00	20.00	19.50
OPERATING PRESSURE, TANKS 1-4, PSIA	23.89 to 24.35	24.35	23.05 to 23.59
OPERATING PRESSURE, TANKS 5 & 6, PSIA	24.35	24.35	23.46 to 23.59
OPERATING PRESSURE, TANK 7, PSIA	23.39 to 23.85	23.85	22.55 to 23.09
DESIGN PRESSURE, ALL TANKS, PSIA	24.35	24.35	23.59
TOTAL INERT WEIGHT,* LB	72,231	72,231	71,943
BOILOFF LOSSES, LB	4,224	4,246	4,173
RESIDUAL VAPOR IN TANK 6, LB	841	841	839
RESIDUAL VAPOR IN TANK 7, LB	918	919	898
MAXIMUM PRESSURANT FLOWRATE, LB/SEC	0.82	0.82	0.81

* INCLUDES 10 % CONTINGENCY

NAR SINGLE-TANK RNS

The NAR single-tank RNS configuration is designed for a nominal propellant load of 300,000 lb. The insulation system includes fixed foam insulation whose thickness was assumed to be constant. The fixed inert weights used as input to the NUTOP were broken down as follows:

● Structure (base structure plus docking and propulsion structure, plus safety system)	9,447 lb
● Meteoroid protection	5,900 lb
● Fixed insulation	3,100 lb
● NERVA engine	27,798 lb
● Auxiliary propulsion	1,320 lb
● Astrionics	6,275 lb

The weight of RCS propellant and fuel cell expendables used during the mission was 5,152 lb. The total inert weight used in the NUTOP included no additional contingency.

NAR SINGLE-TANK RNS

- PROPELLANT LOAD = 300,000 LB LH₂
- TANK MATERIAL IS 2014 ALUMINUM
- FIXED INERT WEIGHT = 53,840 LB
- FIXED WEIGHT INCLUDES EXTERNAL FOAM INSULATION
- VARIABLE INERT WEIGHTS ARE MULTILAYER INSULATION WEIGHT AND TANK WEIGHT
- OPTIMIZATION VARIABLES ARE MULTILAYER INSULATION THICKNESS, TANK OPERATING PRESSURE AND VENT PRESSURE
- MULTILAYER INSULATION PROPERTIES ASSUMED TO BE THE SAME AS THOSE USED ON THE LMSC CONFIGURATIONS

NAR SINGLE TANK -- OPTIMIZED DESIGN DATA

The optimized design data for this configuration are similar to those for the LMSC single tank, but show somewhat greater sensitivity to environmental heating. This is indicated by the higher value of optimum insulation thickness for the step mode. The saving in boiloff in going from the step to the track mode was apparently not sufficient to produce a better optimum at a higher vent pressure. Consequently, the results for the two configurations are very similar.

NAR SINGLE TANK-OPTIMIZED DESIGN DATA

	<u>MALE TRACKING MODE</u>	<u>MALE STEP MODE</u>
INSULATION THICKNESS, INCHES	1.25	1.5
DESIGN PRESSURE, PSIA	26.00	26.00
OPERATING PRESSURE, PSIA	23.39 to 26.00	26.00
VENT PRESSURE, PSIA	26.00	26.00
TOTAL INERT WEIGHT, LB	72,677	73,210
BOIL-OFF LOSSES, LB	9,350	10,028
RESIDUAL VAPOR WEIGHT, LB	9,875	9,872
MAX. PRESSURANT FLOW RATE, LB/SEC	2.78	3.87

McDONNELL-DOUGLAS CLASS I HYBRID RNS

The MDAC Class I hybrid RNS is designed for a nominal propellant load of 290,500 lb in the propellant module and 9,500 lb in the propulsion module. Although the MDAC design calls for prepressurization of the propulsion module from a stored gas supply, in the study it was assumed that all pressurant gas comes from the engine. It was also assumed that the propellant level in the propulsion module was kept constant so that the ullage volume was maintained at 10 percent of the tank volume. The fixed inert weights used as input to the NUTOP were broken down as follows:

• Structure (base structure plus propellant handling plus shield)	16,640 lb
• Meteoroid protection	2,135 lb
• Fixed insulation (foam)	2,755 lb
• Auxiliary propulsion	1,050 lb
• NERVA engine	27,798 lb
• Astrionics	3,160 lb

The weight of RCS propellant and fuel cell expendables used during the mission was 3,002 lb, with 925 lb of this amount being LH_2 removed from the main propellant module. The total inert weight used in the NUTOP included no contingency amount.

MCDONNELL-DOUGLAS CLASS I HYBRID RNS

- PROPELLANT LOAD = 300,000 LB LH₂
- TANK MATERIAL IS 2014 ALUMINUM
- FIXED INERT WEIGHT = 53,538 LB
- FIXED WEIGHT INCLUDES EXTERNAL FOAM INSULATION
- VARIABLE INERT WEIGHTS ARE MULTILAYER INSULATION WEIGHT AND TANK WEIGHTS
- OPTIMIZATION VARIABLES ARE MULTILAYER INSULATION THICKNESSES, TANK OPERATING PRESSURES & TANK VENT PRESSURES ON BOTH TANKS
- MULTILAYER INSULATION PROPERTIES ASSUMED TO BE THE SAME AS THOSE USED ON LMSC CONFIGURATIONS

McDONNELL-DOUGLAS RNS -- OPTIMIZED DESIGN DATA

Propellant module vent and design pressures for the configuration optimized at relatively high levels for both track and step pressurization modes, as in the LMSC and NAR single-tank configurations. The high vent pressure level minimizes boiloff losses for the large tank. However, in the track mode, the optimum vent pressure level for the small propulsion module is relatively low. This is because operating the large tank in the track mode yields such a large reduction in boiloff from this tank ($\Delta M_{BO} = 1,276$ lb) that the vent pressure and insulation thickness on the small tank can be reduced to the indicated values while incurring only an additional 416 lb of boiloff from the small tank. Thus, pressurization in the track mode is also advantageous when a large tank and a small tank are coupled.

In the step mode, however, no advantage is gained by operating the small tank at a low vent pressure, because both tanks must be pressurized to 26 psia for each burn. Thus, the small tank vent pressure optimum is at a relatively high value in order to minimize the (small) boiloff losses from this tank. Computed maximum pressurant flowrates for this configuration are slightly higher than those for the single-tank configurations, because active pressurization of both tanks from the engine was assumed.

MCDONNELL-DOUGLAS RNS-OPTIMIZED DESIGN DATA

	MALF TRACKING MODE	MALF STEP MODE
PROPELLANT MODULE INSULATION THICKNESS, INCHES	1.50	1.50
PROPULSION MODULE INSULATION THICKNESS, INCHES	0.75	1.00
PROPELLANT MODULE VENT PRESSURE, PSIA	26.00	26.00
PROPULSION MODULE VENT PRESSURE, PSIA	19.75	26.00
PROPELLANT MODULE OPERATING PRESSURE, PSIA	23.89 TO 26.50	26.50
PROPULSION MODULE OPERATING PRESSURE, PSIA	23.39 TO 26.00	26.00
PROPELLANT MODULE DESIGN PRESSURE, PSIA	26.50	26.50
PROPULSION MODULE DESIGN PRESSURE, PSIA	26.00	26.00
TOTAL INERT WEIGHT, LB	74,888	74,931
BOILOFF LOSSES, LB	11,455	12,315
RESIDUAL VAPOR IN PROPELLANT MODULE, LB	9,188	9,184
RESIDUAL VAPOR IN PROPULSION MODULE, LB	195	250
MAXIMUM PRESSURANT FLOWRATE, LB / SEC	3.00	3.92

PRESSURIZATION STUDY CONCLUSIONS

The analyses performed in the first part of the pressurization study show that the maximum pressurant flowrate required of the engine is proportional to the tank ullage volume and the difference between the propellant saturation pressure and the operating pressure at the start of a burn. Results of mission performance calculations using optimized data show that the multiple, small-tank configuration requires the smallest values of maximum pressurant flowrates. These calculations also show that the tracking mode of pressurization results in lowest values of maximum pressurant flowrates for any given configuration.

The optimization calculations show that the range of tank operating pressures is very small for a broad range of configurations. The multiple, small-tank systems operate best at low operating pressures, and with the lowest vent pressures. Furthermore, virtually no difference in performance is observed for the system when operating in either the track or the step pressurization mode. Because only the active propellant modules are pressurized, total propellant heating due to addition of hot pressurant gas is minimized for the small tank system, resulting in lower boiloff losses for this system than with other configurations. Operation of the multiple-tank system with the engine in the malfunction mode requires an increase in tank pressure of only 0.76 psi more than that used with normal mode operation.

PRESSURIZATION STUDY CONCLUSIONS

- SMALL PROPELLANT TANKS REQUIRE LOWEST MAX. PRESSURANT FLOW RATES
- TRACKING MODE OF PRESSURIZATION RESULTS IN LOWEST MAXIMUM PRESSURANT FLOW RATE REQUIREMENTS
- MAXIMUM PRESSURANT FLOW RATE REQUIREMENTS ARE 4 LB/SEC
- OPTIMUM OPERATING PRESSURES ARE BETWEEN 23.39 AND 26.25 PSIA
- SMALL PROPELLANT TANK SYSTEMS OPERATE AT LOWEST PRESSURES
- SMALL-TANK SYSTEM DESIGN IS INDEPENDENT OF PRESSURIZATION MODE
- BOIL-OFF LOSSES ARE SIGNIFICANTLY LOWER FOR SMALL-TANK SYSTEMS
- RESIDUAL VAPOR WEIGHTS ARE LOWEST FOR SMALL-TANK SYSTEMS
- PENALTY FOR MALF MODE ENGINE OPERATION IS SMALL FOR SMALL-TANK SYSTEM

E/STS-2 DESIGN SUPPORT—TASK 9.4

TASK OBJECTIVES

- (1) ESTABLISH PRIMARY TEST OBJECTIVES FOR NRDS TEST PROGRAM
- (2) ESTABLISH TEST ARTICLE CONFIGURATION
- (3) DEVELOP NRDS TEST PROGRAM FOR EACH CANDIDATE RNS CONCEPT
- (4) DEVELOP TEST PROGRAM SUPPORT REQUIREMENTS

STATUS

- PRIMARY TEST OBJECTIVES HAVE BEEN DEVELOPED, REVIEWED, & ESTABLISHED
- NRDS TEST ARTICLES WILL BE BASED ON SINGLE-TANK, HYBRID, & MODULAR RNS DESIGN CONCEPTS
- TEST SERIES, SEQUENCES, PROFILES ETC., ARE BEING DEVELOPED
- FLUID REQUIREMENTS ARE 90% COMPLETE

NRDS TEST OBJECTIVES ESTABLISHED

- VERIFICATION OF DESIGN ASSUMPTIONS AND OF RNS COLD FLOW AND NERVA DEVELOPMENT TEST RESULTS BY PERFORMING INTEGRATED SYSTEMS TEST
- DEMONSTRATION THAT THE INTEGRATED RNS SYSTEMS ARE READY FOR FLIGHT TEST