MASTER

*Phase III
Final Review*

NUCLEAR SHUTTLE SYSTEM DEFINITION STUDY

PHASE III

FINAL PERFORMANCE REVIEW

APRIL 14, 1971

Prepared for

GEORGE C. MARSHALL SPACE FLIGHT CENTER

HUNTSVILLE, ALABAMA

UNDER CONTRACT NAS 8-24715

MSFC DRL-197

LINE ITEM 2

NOTICE

PORIONS OF THIS REPORT ARE ILLEGIBLE. It has been reproduced from the best available copy to permit the broadest possible availability.

LOCKHEED MISSILES & SPACE COMPANY
Sunnyvale, California

DISTRIBUTION OF THIS DOCUMENT UNLIMITED

DISCLAIMER

This report was prepared as an account of work sponsored by an agency of the United States Government. Neither the United States Government nor any agency Thereof, nor any of their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness, or usefulness of any information, apparatus, product, or process disclosed, or represents that its use would not infringe privately owned rights. Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise does not necessarily constitute or imply its endorsement, recommendation, or favoring by the United States Government or any agency thereof. The views and opinions of authors expressed herein do not necessarily state or reflect those of the United States Government or any agency thereof.

DISCLAIMER

Portions of this document may be illegible in electronic image products. Images are produced from the best available original document.

MASTER

DISTRIBUTION OF THIS DOCUMENT UNLIMITED

FOREWORD

This report contains copies of the material presented by Lockheed Missiles & Space Company at the Final 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, April 14, 1971.

NOTICE

PORTIONS OF THIS REPORT ARE ILLEGIBLE. It has been reproduced from the best available copy to permit the broadest possible availability.

NOTICE

This report was prepared as an account of work sponsored by the United States Government. Neither the United States nor the United States Energy Research and Development Administration, nor any of their employees, nor any of their contractors, subcontractors, or their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness or usefulness of any information, apparatus, product or process disclosed, or represents that its use would not infringe privately owned rights.

PRESENTATION OUTLINE

INTRODUCTION

- C. D. McKEREGHAN

RNS SYSTEMS DEFINITION

- F. FRIEDLAENDER

RNS DEVELOPMENT AND DEPLOYMENT

- C. VIVIAN

RNS MISSION OPERATIONS

- P. HUTCHINSON

RNS PROGRAM REQUIREMENTS

- C. HOPKINS

CONCLUSIONS AND RECOMMENDATIONS

- C. D. McKEREGHAN

INTRODUCTION

C. D. McKEREGHAN

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

**RNS
SYSTEMS
DEFINITION
F. FRIEDLAENDER**

WORK BREAKDOWN STRUCTURE

The Reuseable Nuclear Shuttle (RNS) definition, as performed by Lockheed, was concentrated solely on a modular stage design. The basis for this design and the general operating approach had been developed in Phase II of this study and formed the baseline system for Phase III.

In the modular concept, all elements of the RNS are assembled in orbit from hardware delivered by a sequence of Space Shuttle launches. In this concept the Space Shuttle also provides all support and logistics services to the RNS. The concept has the additional advantage of extending the level of maintainability of the RNS, since any module can be replaced and returned to Earth for maintenance and refurbishment.

The Work Breakdown Structure has been developed in accordance with a procedure agreed upon by NASA and the contractors. The Reuseable Nuclear Shuttle 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. The Work Breakdown Structure also provides the basis for collecting all technical and cost data.

WORK BREAKDOWN STRUCTURE

SUBSYSTEM SELECTION

The subsystems consist of all structural, propulsion, and astrionic elements. The structure is basically made of aluminum. The pressurized tanks utilize 2021-T81 aluminum designed with fracture mechanics based allowables. Load-carrying elements are shell-stiffened elements of 7075-T6 aluminum, and tubular structural elements such as the spaceframe and docking structure are made from 6061-T6 aluminum. All propellant and propulsion modules are insulated with double aluminized mylar multilayer insulation with tissuglass spacers. All tanks are also individually shielded with dual-wall aluminum meteoroid bumpers.

The propulsion system consists of the NERVA engine for primary propulsion, a feed system, pressurization system, and a reaction control system for auxiliary propulsion requirements. The feed system includes all of the propellant lines, manifolding, propellant conditioning system, in-flight and ground vent system, and on-orbit and ground fill lines. An autogenous pressurization system is utilized to pressurize the propellant tanks during each burn. The RCS system is based on earth-storable N_2O_4/MMH propellants which provide the lightest weight and most mature system for this application.

The astrionic system is based on a completely integrated astrionics approach. The data management system utilizes a central processor, data bus, and local processors on each module. Conventional S-band primary and secondary links and a VHF voice link form the basis of the communications system. The navigation, guidance, and control system is based on a strapdown inertial guidance system supported by mission peculiar navigation aids. A laser rendezvous and docking radar is utilized for vehicle assembly, payload docking, and interfacing with other space elements. The power system utilizes solar panels for the primary source and rechargeable secondary batteries for carrying peak loads and during sun shading periods. The environmental control system is based on coated surfaces for radiating energy and heaters on specific components where energy is required.

SUBSYSTEM SELECTION

STRUCTURE

TANKS - 2021-T81 ALUMINUM

UNPRESSURIZED - 7075-T6 ALUMINUM

SPACE FRAME - 6061-T6 ALUMINUM

INSULATION - DOUBLE ALUMINIZED MYLAR MULTILAYER

METEOROID SHIELDING - DUAL WALL ALUMINUM BUMPER

PROPULSION

FEED SYSTEM - CENTRALIZED DISTRIBUTION

PRESSURIZATION - AUTOGENOUS

RCS - EARTH STORABLES - (N_2O_4 / MMH)

ASTRIONICS

DATA MGMT - INTEGRATED / CENTRALIZED CONTROL / DATA BUS / LOCAL PROCESSORS

COMMUNICATIONS - S-BAND PRIMARY & SECONDARY LINKS, VHF VOICE LINK

NAVIGATION, GUIDANCE & CONTROL - INERTIAL GUIDANCE - STRAPDOWN / LASER DOCK-
ING RADAR

POWER - SOLAR PANELS / SECONDARY BATTERIES

ECS - PASSIVE / HEATER

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 total impulse requirements 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 storable systems such as those using $\text{OF}_2\text{-B}_2\text{H}_6$ ($I_{\text{sp}} = 380$ to 410 sec) offer attractive performance, but were eliminated on the basis of possible safety problems and lack of developmental effort currently planned.

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) • HYPERGOLIC IGNITION <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) • IGNITER REQUIRED <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} • LOW INERT WEIGHT <p>SYSTEM WEIGHT 2400 LB</p>	MAJOR DEVELOPMENT OF A NEW TECHNOLOGY	NO LARGE-SCALE TECHNOLOGY DEVEL- OPMENT PLANNED

POWER SYSTEM COMPARISON

The power system sources were re-examined 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 + \$ 6.0M 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

MODIFICATIONS DUE TO FMEA

Failure mode and effects analyses (FMEA) were conducted on the RNS systems to verify that the following basic guideline requirements were met:

- All credible single failure modes or credible combinations of failures and errors which result in loss of crew and passengers or unacceptable risk to general population groups will be eliminated by design change and/or mission modification.
- No single failure or credible combination of failures and errors will prevent or preclude operation of the NERVA engine in the Emergency Mode.

From these guidelines, specific criteria were established for each system. For the Propellant Feed System pictured in the schematic, the following criteria were to be satisfied:

- No single failure shall result in loss of crew or passengers or in abandonment of the RNS in an orbit or situation from which ultimate recovery of the RNS cannot be effected.
- No single failure shall preclude NERVA emergency mode operation.
- Flow to either turbopump shall be independently controlled.
- No double malfunction shall prevent flow of coolant to the engine.
- Liquid level in the propulsion tank shall never be allowed to fall below that required for engine shut-down and complete cooldown.

Modifications to the feed system resulting from the FMEA consisted primarily of added redundancy in the individual areas. For other systems, such as the RCS, the FMEA indicated the design criteria could best be satisfied with a net reduction in components. The FMEA identified design modifications in each of the systems evaluated to satisfy the established design criteria.

MODIFICATIONS DUE TO FMEA

RCS CRITICALITY APPRAISAL

Based upon the generic failure rate data, duty cycle requirements, and single malfunction analysis data, a preliminary criticality appraisal was performed for the RCS system. For components which had more than one significant failure mode, all significant failure modes were accounted for in the analysis. The criticality values presented do not account for redundancies in the system. The analysis does indicate the significance of specific components and the impact that duty cycle has on the criticality value. The duty cycles assumed for the thrusters are as follows:

<u>No. of Cycles</u>	<u>Thruster Function</u>
24,634	Pitch
14,780	Yaw
1,340	Axial
536	Axial (Spaceframe Only)

These numbered cycles were based on using a typical impulse bit of 1.5 lb-sec per operation.

RCS CRITICALITY APPRAISAL

SUBSYSTEM	CRITICALITY	COMPONENT	CRITICALITY
FILL-FEED	618.25	FUEL SPACE FRAME-PROP. MODULE CONNECT	222.0
		OXIDIZER SPACE FRAME-PROP. MODULE CONNECT	222.0
PROPELLANT MODULE NO.1	4643.35	He PRESSURE REGULATOR VALVE #1	1892.42
		He PRESSURE REGULATOR VALVE #2	1892.42
PROPULSION MODULE NO.2	4643.35	He PRESSURE REGULATOR VALVE #1	1892.42
		He PRESSURE REGULATOR VALVE #2	1892.42
THRUSTER CLUSTER NO.101	157.24	PITCH ENGINE ASSEMBLIES	60.48
" " " 102	133.05	YAW ENGINE ASSEMBLIES	36.28
" " " 103	157.24	AXIAL ENGINE ASSEMBLIES	3.29
" " " 104	133.05	AXIAL ENGINE ASSEMBLIES	1.32
" " " 105	75.86	(SPACE FRAME ONLY)	
" " " 106	124.24		
" " " 107	9.21		
RCS SYS CRITICALITY	10,694.83		

RNS INTEGRATED SCHEMATIC

Although the RNS is a modular system, it is functionally completely integrated. The core of the system is the spaceframe which contains the data management central processor, guidance system, communications system, power system, and RCS system. The spaceframe is consequently a completely autonomous self-contained propulsive vehicle. The addition of modules merely increases its propulsive capability. The propulsion module adds a propellant tank and the NERVA engine. Each of the six propellant modules adds only increased total impulse to the vehicle.

The data management system completely integrates all of the propulsion and astrionic functions of the modules. Each module is linked to the basic system by a data bus which interfaces with a local processor on the module. The processor provides capability for checkout and commands which can be accomplished without communicating back to the central processor, thereby minimizing the data flow on the main data bus. Power is carried to all modules by an unregulated 28 volt dc bus with a junction box connection for each module. The pressurization system takes engine bleed gas and distributes it through a manifold to all propellant tanks. Each tank is controlled independently so that only the active tanks are actually pressurized. Each of the propellant modules is also connected to a manifolded feed line that connects to the propulsion module tank. Propellant can be drained from all, one, or any combination of tanks with this system. The propulsion module feeds the engine with two parallel feed lines. The propulsion module tank also acts as an accumulator and can handle short burns and aftercooling, thereby minimizing activation of the propellant tanks. The RCS system is enhanced by the addition of the propulsion module to the spaceframe. Several more clusters of engines become operative and control is more efficient with the longer moment arms.

RNS INTEGRATED SCHEMATIC

PROPELLANT MODULE -- INBOARD PROFILE

The propellant module consists of a propellant tank, propellant, insulation system, meteoroid protection system, and forward 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 spaceframe 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 spaceframe to allow for tank thermal effects at cryogenic temperatures.

PROPELLANT MODULE - INBOARD PROFILE

PROPULSION MODULE - INBOARD PROFILE

The propulsion module consists of the NERVA engine, engine flyaway control module, a propellant tank and supports, propellant, and meteoroid and thermal protection systems. It also incorporates ground and emergency venting, flight venting, 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 spaceframe 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.

PROPULSION MODULE-INBOARD PROFILE

NERVA ENGINE INTERFACE

The NERVA/RNS interface has been defined. 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. Both physical and functional requirements are accommodated at the interface. The engine flyaway control module, which is mounted on the ground to the engine, also provides the functional transition from the NERVA subsystems to the RNS subsystems concepts.

NERVA ENGINE INTERFACE

NERVA INTERFACE REQUIREMENTS

The NERVA interface requirements encompass both physical and functional characteristics. Proper mechanical attachment conditions must be satisfied so that the engine thrust loads can be transferred and all plumbing and wiring connections are aligned and seated properly. The functional requirements are listed on the adjoining chart.

NERVA INTERFACE REQUIREMENTS

FEED SYSTEM

- PRESSURE - 18 → 26 PSIA SATURATED LH₂
- FLOW RATE - 91.7 LB/SEC

PRESSURIZATION SYSTEM

- PRESSURE - 200 → 750 PSIA
- TEMPERATURE - 175° → 250° R

COOLDOWN LINE

- PRESSURE - 18 → 26 PSIA SATURATED LH₂
- FLOW RATE - 1.7 LB/SEC

THRUST VECTOR CONTROL

- GIMBAL DEFLECTION 5° / RATE 0.25°/SEC / ACCELERATION 0.5°/SEC²

POWER SYSTEM

- PEAK 3.5 KW, AVERAGE 2.25 KW (OPERATION)

TELEMETRY

- DIGITAL SAMPLE RATES 10 TO 100 SAMPLES/SEC
- ANALOG-FREQUENCY RANGE 0-3000 Hz

SAFETY

- SYSTEM TO PREVENT ACCIDENTAL CRITICALITY

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 thrust 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

STRUCTURAL FRAME -- INBOARD PROFILE

The spaceframe 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 spaceframe module contains the propulsive and intelligence capability and has its own station-keeping capability. The module is 15 ft. by 60 ft. in the stowed position, and extended to 15 ft. by 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 spaceframe. The six umbilical panels provide connections for the data management, propellant feed, pressurization, and power from the tank modules to the spaceframe. The propellant feed manifold interconnects to the propulsion module through an aft umbilical plate.

STRUCTURAL FRAME - INBOARD PROFILE

RNS EQUIPMENT LAYOUT

The spaceframe contains most of the subsystems for the RNS. The primary location for equipment is in the equipment module at the forward end of the spaceframe. The equipment module is divided into bays with specific equipment allocations. The RCS propellants are stored in redundant supply modules in bays 1 and 4. Data management and communications is stored in bay 3; guidance, navigation, and control and power are stored in bay 6. Bay 2 is configured like bay 3 and bay 5 is configured like bay 6. Although these bays are empty, their design is such that replacement modules can be installed in them and continuity of operation provided during maintenance or replacement operations. RCS thrusters are clustered in groups of three wherever possible. The solar panels and communications antenna are deployed from the lower section of the equipment module.

RNS EQUIPMENT LAYOUT

RNS MODULE INTEGRATION

The three basic modules form the building blocks of the integrated RNS. The spaceframe is the first module deployed and is a completely autonomous space vehicle. In addition to its propulsive and astrionic capabilities, it also contains equipment for assembly of the RNS. The propulsion module tank is the next element deployed. It is attached to the spaceframe with a docking ring and six connecting pins that are aligned with the longitudinal members of the hexagonal spaceframe. All of the functional systems are connected through an umbilical plate which is mated subsequent to the structural mating. The NERVA engine is the next element deployed. The NERVA engine has been mated to an engine flyaway control module on the ground which allows the engine system to fly itself into position with the propulsion module tank. The propellant modules are delivered next. They are attached to the spaceframe by large conical pins. The aft attachment, which is rigid, is the primary attachment point and the forward attachment can slide axially in the spaceframe to allow for thermal expansion. The functional attachment is by means of an umbilical which is similar to the propulsion module. All of the six propellant modules are attached in the same manner. The total system can then be checked out completely.

RNS MODULE INTEGRATION

REUSABLE NUCLEAR SHUTTLE

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

The RNS consists of a central spaceframe, 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 spaceframe, which contains the necessary fluid manifolding to feed the propulsion module.

The propulsion module is mounted on the aft end of the spaceframe. 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. by 60 ft. envelope for compatibility with the Space Shuttle cargo bay. The NERVA engine and the orbital assembly adapter are delivered to orbit separately 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 83,675 lb., including a 5 percent contingency. The RNS performance for the nominal lunar mission is 78,451 lb. delivered to the moon and 20,000 lb. returned from the moon. These weights and performance data include a 8100-lb. supplemental shield for crew protection.

REUSABLE NUCLEAR SHUTTLE

THERMODYNAMIC ENVIRONMENT

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 LH_2 tanks.

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.

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

RNS THERMODYNAMIC DEFINITION

A thermodynamic optimization analysis was conducted utilizing a 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.

RNS THERMODYNAMIC DEFINITION

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

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 Station 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

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	$> 10^8$	$\gg 10^8$	—	
VALVE SEALS	5×10^5	3×10^5	$> 10^7$	$\gg 10^8$	—	ASSUMING URETHANE OR ESTHANE TYPE SEAL
WIRING INSULATION	1×10^6	6×10^5	$> 5 \times 10^7$	$> 10^8$	—	ASSUMING ORGANIC INSULATION
TRANSDUCERS	1.5×10^6 $2 \times 10^{11} *$	1×10^6 $5 \times 10^{14} **$	$> 2 \times 10^7$ $> 10^{15} **$	$> 2 \times 10^7$ $> 10^{16} *$	—	
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 FLUX N/CM²/SEC
 ** NEUTRON INTEGRATED FLUX NVT

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			
		MODULES			VEHICLE*
		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	(1,141)	(2,617)		(9,463)
	INSULATION	374	729		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)	(37,413)	(667)	(39,535)
	NERVA ENGINE		27,728		27,728
	EXTERNAL DISC SHIELD FOR NERVA	—	8,100		8,100
	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		1,134		1,134
6.00	AUXILIARY PROPULSION		(113)	(548)	(661)
	REACTION CONTROL SYSTEM		113	548	661

RNS WEIGHT STATEMENT- III

CODE	DESCRIPTION	WEIGHT, LB			
		MODULES			VEHICLE
		PROPELLANT	PROPULSION	CONTROL & ASSY STRUCTURE	
7.00	ASTRIONIC SYSTEM	(40)	(45)	(1,533)	(1,818)
	GUIDANCE, NAVIGATION & CONTROL	—	—	296	296
	INSTRUMENTATION & SENSORS	10	10	10	80
	COMMAND & CONTROL	—	—	254	254
	ELECTRICAL POWER	—	—	560	560
	ELECTRICAL NETWORKS	5	10	140	180
	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)	(411)	(403)	(2,278)
	TOTAL INERT	5,125	44,464	8,461	83,675
10.00	RCS PROPELLANT			(2,200)	(2,200)
14.00	HYDROGEN PROPELLANT	38,676	37,371	—	269,427
	TOTAL WEIGHT	43,801	81,835	10,661	355,302

RNS ALTERNATE CONFIGURATION

The alternate configuration of the Lockheed Reuseable Nuclear Shuttle is a variation on the baseline concept, with propellant load expanded to 301,437 pounds.

The alternate configuration consists of the baseline spaceframe and six propellant modules. The spaceframe has been modified by the incorporation of a 17,300-pound propellant tank in its core. The propulsion module has also been expanded. A 13,405-pound hydrogen tank has been inserted between the baseline propulsion module tank and the NERVA engine. The small tank next to the engine has been sized so that this tank and the engine will fit within the 15 by 60-foot cargo bay of the Space Shuttle as a unit. All of the other systems are essentially the same as for the baseline RNS.

The overall length of the alternate configuration is 2230 inches, including the length of the equipment module which is deployed in orbit. The current estimate for the inert weight of the assembled vehicle is 84,803 pounds, including a 5 per cent contingency. The performance for the nominal lunar mission has been significantly increased over the baseline vehicle and is 118,125 pounds delivered to the moon and 20,000 pounds returned from the moon. These weights and performance data include a 3900-pound supplemental shield for crew protection.

RNS ALTERNATE CONFIGURATION

RNS ALTERNATE CONFIGURATION WEIGHT SUMMARY

The alternate configuration weight summary provides a weight breakdown by subsystem of the vehicle. The 5 percent contingency is carried for all dry inert weight except the NERVA engine and the external disc shield. The total launch weight includes RCS propellant weight and all of the hydrogen in the tanks.

RNS ALTERNATE CONFIGURATION

WEIGHT SUMMARY

CODE	DESCRIPTION	WEIGHT LB
2.00	STRUCTURE	31,206
3.00	METEOROID / THERMAL PROTECTION	10,635
4.00	DOCKING / CLUSTERING	2,759
5.00	MAIN PROPULSION	35,092
6.00	AUXILIARY PROPULSION	661
7.00	ASTRIONIC SYSTEM	1,918
		82,271
	CONTINGENCY - 5% (EXCLUDES NERVA ENGINE & DISC SHIELD)	2,532
	TOTAL INERT (DRY)	84,803
	RCS PROPELLANT	2,200
	HYDROGEN PROPELLANT	301,437
	LAUNCH WEIGHT	388,440

**RNS
DEVELOPMENT
AND
DEPLOYMENT
C. VIVIAN**

FUNCTIONAL FLOW DIAGRAMS - TOP LEVEL

The functional flow diagrams on the next page represent the top level functions for the RNS Development Program. In the interest of clarity the next discussion will be presented as depicted functionally on these diagrams. In this manner the hardware can be described and the major pieces of hardware can be followed as they flow through the various functional phases of development, manufacturing, acceptance testing, shipment, launch, and orbital assembly. Subsequent discussions will describe the flight operations.

REUSEABLE NUCLEAR SHUTTLE

FUNCTIONAL FLOW DIAGRAM - TOP LEVEL

INTEGRATED TEST PLAN

The primary purpose of the Integrated Test Plan is to ensure the close coordination of the various engineering organizations, manufacturing, product assurance, and the test organization necessary to obtain an efficient, orderly development of the RNS components, subsystems, and systems. To provide this effective means of management control, the Integrated Test Plan delineates the following:

- All test programs to be implemented
- Time phasing of individual test programs
- Interrelationship among individual test programs
- Relationship of each major test and test program to the overall RNS program

The plan defines and integrates the various test programs from the initial R&D tests through the final flight tests. Such definition and integration is essential to the successful achievement of the overall objectives of an extensive program.

The chart on the opposite page pictorially presents the basic development of the major subsystems during the early phases of the development program, which permits an orderly evolution to the total integrated systems test. The culmination of the Integrated Test Plan is the flight test program.

INTEGRATED TEST PLAN

RNS TEST SCHEDULE

The test schedule, shown on the opposite page, reflects the test spans for the major tests required for the RNS development. The major test activities during Phases B and C concern component screening, subsystem development, and bread board testing. Design verification and qualification occur primarily during the Phase D activities. Environmental qualification and reliability testing overlap into the earlier Phase C test activities to provide preliminary data on candidate components and subsystem suitability to the design environments and operating requirements.

SPACEFRAME MODULE ASSEMBLY

The proposed sequence of spaceframe fabrication is shown on the following three pages. These diagrams show the major steps in the course of the spaceframe buildup supporting subassembly activities plus a tooling concept as envisioned at this time.

Structural welding of the spaceframe can be accomplished using a lay-down fixture consisting of a base and a roller-supported two-piece external cage. The cage halves join longitudinally along the centerline with removable longeron locators spaced adjacent to the juncture of the struts and braces.

The cage halves may be used for support and handling of the spaceframe through subsequent steps of structural buildup, cleaning, final assembly, and system test.

The assembly fixture will be designed to maintain alignment relationships between the spaceframe deployment mechanisms and the equipment section, to control avionics module interfaces, and for positive location of the bi-stem turntable, solar array, and RCS thruster package mountings. Avionics will be built up separately and bench-tested prior to installation in the equipment section. Separate buildup and test of subassemblies include aft bi-stem, propellant transfer and pressurization manifolds, piping and valves, docking and propellant tank transfer booms, and umbilical drive mechanisms.

SPACE FRAME MODULE ASSEMBLY 1

SPACE FRAME MODULE ASSEMBLY 2

SPACE FRAME MODULE ASSEMBLY 3

PROPULSION AND PROPELLANT MODULE ASSEMBLY

The initial concept developed for build-up of the propulsion and propellant modules is shown on the following pages. These diagrams reflect an interpretation of hardware design and shows the most logical breakdown of the propulsion module from a manufacturing point of view. This breakdown and flow is, for the most part, equally applicable to propellant module fabrication.

An effort has been made to diagram the most significant steps in the course of assembly buildup. Where appropriate to help define a given process, tooling concepts have been included. However, it should be recognized that the major tooling concepts suggested here are preliminary and will be subjected to in-depth scrutiny during future program phases of hardware definition.

PROPULSION & PROPELLANT MODULE ASSEMBLY 1

PROPULSION & PROPELLANT MODULE ASSEMBLY 2

PROPULSION & PROPELLANT MODULE ASSEMBLY 3

RNS MANUFACTURING SCHEDULE

The schedule on the opposite page identifies the time spans required to fabricate, assemble, refurbish, and test the major pieces of development hardware. In addition, the relationships of tooling procurement and program phasing to the total manufacturing activities are presented.

The refurbishment activities shown reflect changes to test hardware to be used for different test programs. Terms used in this schedule are identified as follows:

FTV	Flight Test Vehicle
OV	Operational Vehicle
REFURB	Refurbish
PLM	Propellant Module
PPM	Propulsion Module
SFM	Spaceframe Module
GSE	Ground Support Equipment

REUSABLE NUCLEAR SHUTTLE MANUFACTURING SCHEDULE

*INCLUDES ASTRIONICS

PRELAUNCH OPERATIONS

Upon completion of fabrication and assembly of the RNS individual module, the flow of hardware is basically as shown on the opposite page. That is, the RNS module will go from the manufacturing facility to the appropriate test site and then to Kennedy Space Center via Super Guppy or ocean-river barge. It is recommended that the final acceptance testing be accomplished at the manufacturing site to minimize module movement and cost. Transportation modes to be employed would include both ocean and air. Ocean transportation is recommended for the initial shipment of the RNS modules. Subsequent shipment of replacement modules would be by air.

PRELAUNCH OPERATIONS

ACCEPTANCE TEST OPTIONS

Replacement of modules and components on orbit requires acceptance of those modules and components to rigorous ground acceptance specifications. This acceptance testing will require use of interface simulators. Development of acceptance specifications and interface simulators to make on-orbit module and component replacement feasible also provides the opportunity for ground checkout by major modules without resorting to assembly of all major modules to the complete vehicle on the ground. Establishment of component and module interface tolerances could be accomplished by complete assembly and integrated test of a development test vehicle. This vehicle would also be used to determine systems interactions and to verify satisfactory propellant flow.

The dimensions of the propellant and propulsion modules will be affected considerably by the propellant cryogenic temperatures. This will affect the fit of the propellant interface connectors and various module alignments. Therefore, the interface fit checks of the modules using master tooling and also certain alignments will be conducted while the modules are tanked with liquid hydrogen.

The environment for conducting the required acceptance tests has been determined by a consideration of test results which may be achieved in various environments, as shown on the opposite page. The tabular data indicate generally that manufacturing tests will require ambient and cryogenic conditions, leak tests can be conducted best in a liquid hydrogen environment, and electronics systems can be tested best in the space environment of a temperature-altitude simulation chamber. From the systems involved in the various modules it follows that acceptance tests of the propellant and propulsion modules can be accomplished in a cold flow facility. The tanking with liquid hydrogen will subject the tank to operational cryogenic conditions, and much of the instrumentation experiences the liquid hydrogen temperatures. However, testing of the spaceframe can be accomplished best in the space environment of a temperature-altitude simulation chamber. The chamber, programmed to provide space cold and solar heat, will present the electronics modules and attitude control system with the environment conditions that these systems would experience in orbit. With these exceptions, other tests to be conducted on the propellant, propulsion, and spaceframe modules will be similar.

ACCEPTANCE TEST OPTIONS

ORDER OF INCREASING FACILITY COMPLEXITY

TASKS	AMBIENT	LH ₂	TASC	TASC & LH ₂
●ALIGNMENT	GOOD	EXCELLENT	—	—
●FIT CHECKS				
●BATTERIES, ACCESS DOORS	EXCELLENT	EXCELLENT	—	—
●MODULE INTERFACES	GOOD	EXCELLENT	—	—
●ELECTRICAL CONTINUITY	GOOD	EXCELLENT	GOOD	EXCELLENT
●ELECTRONICS SYSTEMS TEST	GOOD	GOOD	EXCELLENT	EXCELLENT
●PROGRAMMED INTEGRATED ACCEPTANCE TESTS	GOOD	GOOD	EXCELLENT	EXCELLENT
●MODULE INTERFACE LEAK TESTS	GOOD	EXCELLENT	GOOD	EXCELLENT
●PROPELLANT SYSTEM LEAK TESTS	GOOD	EXCELLENT	GOOD	EXCELLENT
●INSULATION SYSTEMS PERFORMANCE	—	GOOD	—	EXCELLENT
●PROPELLANT MEASURING SYSTEM CALIBRATION	FAIR	EXCELLENT	FAIR	EXCELLENT

FACTORY-TO-LAUNCH OPERATIONS

The flow of the first RNS modules upon completion of fabrication and assembly is shown on the next page. Basically, the various RNS modules will go to the appropriate test site for acceptance testing and then they will be transported individually (via air) or in groups (via barge) to the launch site at KSC. When necessary, the various modules will be placed in storage, either at the manufacturing site or at the launch site, to await the next step. At the proper time the various modules will be removed from storage and each piece of hardware will flow through check-out, assembly, servicing, and launch operations. The primary activity prior to assembly into the Space Shuttle will be to determine status of all equipment mounted on the module and to ascertain that there have been no charges since the final acceptance test at the manufacturing facility or since the last test. The sequence of launch of the flight test vehicle modules is dependent upon assembly requirements and priorities. However, in all cases the spaceframe modules must be deployed first to establish the stable platform for final assembly in orbit.

FACTORY TO LAUNCH OPERATIONS

LAUNCH AND ORBITAL OPERATIONS

All of the RNS modules are compatible with the 15 by 60-ft cargo bay of the Space Shuttle. As such, all modules maintenance parts and supplies necessary to the RNS operations can be delivered on orbit by the Space Shuttle. Of all the major elements, i.e., Space Station/Base, Space Tug, etc., the Space Shuttle is the only element essential to RNS earth orbital operations.

The Space Shuttle plays a major role, not only in the delivery of the modules to earth orbit, but in the determination that the modules are in flight readiness and ready for assembly. In addition, the Space Shuttle provides the necessary command and control functions during all orbital operations.

The initial assembly of the RNS system in orbit is accomplished in four basic phases. Launch of the spaceframe module is the first phase, as it contains the propulsive and intelligence capability necessary to maintain station-keeping and establishes readiness for tank assembly and all subsequent autonomous operations. Deployment of the power system, astrionic equipment tank bays that are located in the spaceframe module, prepares for the next phase of assembly operation, which is to assemble the propellant and propulsion module tanks into the spaceframe. The engine is deployed by the Space Shuttle, the autonomous docking system is activated, and the rendezvous and docking of this RNS subsystem is accomplished. Payload is assembled as required by its configuration and nature. Large payloads would be assembled autonomously through rendezvous and docking or by systems similar to the tank handling devices. Small packages compatible with EVA would be handled in that manner, although it is not anticipated that the payloads would normally be of that nature.

LAUNCH AND ORBITAL OPERATIONS

CHECKOUT PHILOSOPHY

The acceptance testing or checkout philosophy specifies the testing to be accomplished during manufacturing test and systems test at the manufacturing site, during launch preparation at the launch site, and during validation tests on orbit following final nuclear stage assembly or module replacement. The nuclear shuttle testing will follow existing test concepts during manufacturing and systems test, but will differ in the extent of testing required at the launch base, and will introduce the opportunity for checkout and test on orbit. One reason for the change in testing concept is that at the time of launch and during ascent the nuclear stage modules are passive in a transported capacity, rather than active as a rocket stage. A second reason is that the nuclear stage will be assembled on orbit. To accommodate this on-orbit module assembly the capability has been provided for checkout and failure diagnosis following assembly. These circumstances and capabilities reduce the checkout requirements at the launch base, which will be an interim point rather than the final point for assembly, checkout, and corrective action.

CHECKOUT PHILOSOPHY

	MFG & ASSY	TRANS- PORT TO KSC	KSC		SPACE SHUTTLE LAUNCH & BOOST	EARTH ASSEMBLY ORBIT		IN- FLIGHT
			PRE- LAUNCH	LAUNCH PAD		SPACE SHUTTLE PAYLOAD C/O	RNS C/O	
PROPELLANT MODULE	▲●	○	○	△○	○	△○	■	○■
PROPULSION MODULE	▲●	○	○	△○	○	△○	■	○■
SPACE FRAME MODULE	▲●	○	○	△○	○	△○	■	○■
NERVA ENGINE	TBD	TBD	TBD	TBD	TBD	TBD	■	○■
REPLACEMENT SUBSYSTEMS (SPACE REPLACEABLE UNITS)	▲●	○	○	△○	○	△○	■	○■

LEGEND

- ▲ FUNCTIONAL AND/OR SIMULATED TEST (IN DEPTH)
- △ PARTIAL FUNCTIONAL TEST (MINIMAL)
- ACCEPTANCE TEST
- STATUS MONITORING
- INTEGRATED RNS CHECK
- TBD TO BE DETERMINED BY ENGINE CONTRACTOR

RNS MODULE DATA INTERFACE

The Space Shuttle will deliver to earth orbit four distinct units (propellant module, propulsion module, engine, and spaceframe) which must be ready to go and in good operating condition when assembled.

To ensure prior assembly credibility, certain key checks (performance, environment and tie-down) must be made during delivery which requires Space Shuttle and RNS delivery unit breakdown interfaces. Applicable questions arising from the above are what support functions are required to be checked during delivery of particular RNS units and whether the capability to perform self-check is inherent in the design or must it be supported or totally provided by the space shuttle.

The chart shows that the spaceframe during delivery can generally support itself, while the other three units (propellant, propulsion, and engine) will require some support from the Space Shuttle.

RNS MODULE DATA INTERFACE

RNS SUPPORT FUNCTIONS	RNS DELIVERY BREAKDOWN							
	PROPELLANT		PROPULSION		SPACE FRAME		ENGINE	
	EXIST'G	REQ'D	EXIST'G	REQ'D	EXIST'G	REQ'D	EXIST'G	REQ'D
PROVIDE DATA ON ALL FAILURES		△		△		△		△
MONITOR CARGO COMPARTMENT		△		△		△		△
PROVIDE STIMULI	△		△		△		△	
MONITOR PARAMETER LIMITS	△		△		△		△	
PROVIDE ALARM & COMMANDS		△		△	△			△
DETECTION OF SAFETY HAZARDS	△		△		△		△	
RECORD SYSTEM STATUS		△		△	△			△
PROVIDE TREND ANALYSIS		△		△		△		△
PROVIDE PREDICTION		△		△		△		△
PROVIDE TEST & C/O	△		△		△		△	
PROVIDE SYSTEM CONFIGURATION MGMT		△		△	△			△
PROVIDE ELECTRICAL POWER		△		△		△		△
PROVIDE ENVIRONMENTAL CONTROL						△		

NOTE: EXIST'G - FUNCTIONS ARE INHERENT IN MODULE DESIGN
 REQ'D - FUNCTIONS MUST BE SUPPORTED TOTALLY OR IN
 PART BY SPACE SHUTTLE

RNS/SPACE SHUTTLE INTERFACE NO. 1

The Space Shuttle interfaces with each one of the RNS modules. The central and assembly spaceframe, the propellant modules, the propulsion modules and the integrated fly-away module (FAM), and NERVA engine are all compatible with and are launched in the 15 by 60-ft cargo bay of the Space Shuttle. The basic interfaces for the spaceframe and the propulsion module are shown on the opposite page.

Both of these modules are mounted on a forward-hinged, swing-out mechanism provided in the current Space Shuttle design considerations. Clearance for a direct swing-out is provided in the 60-ft length limitation.

The functional interface for the spaceframe includes data management and power supply, RCS fuel (MMH) oxidizer (N_2O_4), and pressurization gas (H_e). The propulsion module functional interface includes connectors for data management and power supply, vent system, insulation purge (H_e), and tank purge.

The spaceframe is mounted in the Space Shuttle with the equipment bay forward. The propulsion module is installed with the FAM/RNS interface forward.

RNS/SPACE SHUTTLE INTERFACE 1

RNS/SPACE SHUTTLE INTERFACE NO. 2

The Space Shuttle/Propellant Module and Space Shuttle/FAM-NERVA interfaces are identified on the next page. The FAM-NERVA assembly is deployed in the same manner as the spaceframe and the propulsion module. However, the propellant module requires a slightly different deployment mechanism.

The spaceframe is shown docked to the Space Shuttle and the propellant module is rotated out of the Space Shuttle bay for assembly. However, the propellant module must be aligned parallel to the spaceframe on a centerline aft of the other module deployment centerline. An additional hinged point, then, must be incorporated in the Space Shuttle deployment mechanism to properly align the propellant module for boom attainment.

All modules have built-in test equipment and are capable of checking themselves upon interrogation. As such, these modules, like the previously discussed modules have a simply data bus and power interface. The FAM requires RCS fuel oxidizer and pressurant gas connections in addition to engine purge lines.

All propellant-carrying modules are fueled within the Space Shuttle through a LH_2 line at a common location. The Space Shuttle controls all servicing and vent functions on the ground and in flight until it is deployed.

RNS SPACE / SHUTTLE INTERFACE 2

RNS MODULE/SPACE SHUTTLE DEPLOYMENT

RNS - ENGINE SYSTEM/SPACE SHUTTLE INTERFACE

OPERATIONAL MODE FLEXIBILITY

The modular RNS design has many favorable features, and the spaceframe provides a flexibility that enhances the various recycle operations (i.e., refueling, earth-based major maintenance activity, etc.). This flexibility becomes apparent when considering the various operating modes possible for refueling.

Basically, the refueling operations can be accomplished in two modes: (1) direct fueling from a Space Shuttle Integral tanker and (2) tank exchange with topping from a Space Shuttle/integral tanker. A variation on the tank exchange mode is a complete exchange of all RNS modules by bringing up a spaceframe to start a new RNS assembly and recycle the old RNS modules back to earth on the return trip. These capabilities in different operational modes will permit mission planning to proceed without undue constraints from recycle or maintenance operations.

OPERATIONAL MODE FLEXIBILITY

COST IMPLICATIONS OF DEWAR ALTERNATIVES

Two general classes of propellant resupply dewars were considered -- dedicated systems designed exclusively for propellant storage in orbit, and "spare-RNS" systems in which the principal elements of a Reusable Nuclear Stage (less engine) would function as the dewar. Two configurations were considered for each class of dewar -- single-tank concepts launched by the INT-21, and multiple-tank configurations launched by the space shuttle and assembled in orbit.

The program alternatives, cost elements, and estimated costs associated with fulfilling the maximum propellant-delivery objective with an orbital propellant depot are presented on the next page. Note that the development (nonrecurring) costs of the dedicated dewars make these concepts higher in cost than the extra-RNS concepts. In addition, the INT-21 launch vehicle costs tend to drive the total cost of single-tank dewar systems higher than comparable multi-tank configurations.

COST IMPLICATIONS OF DEWAR ALTERNATIVES

OBJECTIVE	ALTERNATIVES	COST ELEMENTS	COSTS	
			SINGLE TANK	MULTIPLE TANK
ACCUMULATE PROPELLANT IN ORBIT TO TAKE MAXIMUM ADVANTAGE OF SPACE SHUTTLE CAPACITY	DEVELOP A DEDICATED DEWAR	DEWAR NONRECURRING COSTS	\$ 300 M	\$ 300 M
		DEWAR RECURRING COSTS	61*	58*
		LAUNCH VEHICLE & LAUNCH OPNS COSTS	216	45
			<u>\$ 577 M</u>	<u>\$ 403 M</u>
	ADD ONE EXTRA RNS TO THE FLEET	EXTRA RNS RECURRING COST (LESS ENGINE)	\$ 65 M*	\$ 62 M*
		RNS LAUNCH VEH & LAUNCH OPNS COSTS	216	45
			<u>\$ 281 M</u>	<u>\$ 107 M</u>

* FOR ONE DEWAR ONLY
MULTIPLE ORBIT MODES REQUIRE MORE DEWARS

REFUELING CONSIDERATIONS

As can be seen on the opposite page, the losses from an orbit propellant transfer from one tank to another is relatively small. Far more significant are the boil-off losses that can be incurred whenever the assembly time is stretched out due to infrequent Space Shuttle flights. If the saturated condition of the LH_2 propellant is at or near the vent pressure, the first tank delivered will be venting during the complete recycle operations. Vent time will decrease with successive tanks until the last tank is delivered. This holds true for the direct fill operations also.

These losses can be eliminated by subcooling the propellant on the ground for installation in a subcooled condition spaceframe. The propellant in each tank can be delivered at a different saturated condition, depending on the desired pressure at time of use. It is conceivable that each tank could be conditioned to fit the mission. This would minimize boiloff during the mission as well as during orbital assembly or recycle operations.

REFUELING CONSIDERATIONS

PROPELLANT TRANSFER LOSSES

BOIL-OFF LOSSES

Psat REQUIREMENTS

INDUCED RADIATION ENVIRONMENT

Gamma dose rates arising from the neutron activation of materials in the propulsion-module/engine-adapter 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, OC) reaction with Al, while Co-58 and Mn-54 isotopes result from (n, p) reactions with Fe-54 and Ni-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

RNS MODULE INTEGRATION

The approach to RNS assembly is based upon the technique of a hinged, rotating deployment system for cargo handling within the Space Shuttle. 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 spaceframe either with or without use of the assembly booms. The propellant modules would be installed in a like manner shown on the facing chart.

The propulsion module is installed on the RNS axis at the aft end of the spaceframe. The Space Shuttle will move to the position shown on the chart.

The propulsion module is swung 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 spaceframe. Final alignment and structural joining are achieved by motor-driven bolts mounted in the spaceframe longerons.

For installation of the propellant modules, the RNS spaceframe hard-docks to the Space Shuttle (assuming the Space Shuttle is equipped with a docking adapter) and then the propellant module is rotated out to a position parallel with the spaceframe. At this point the booms are attached and the transfer is completed in a close system.

RNS MODULE INTEGRATION

MODULE ASSEMBLY MECHANISMS

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.

MODULE ASSEMBLY MECHANISMS

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 seconds, only the propulsion module transfer associated with linear cargo deployment, presents a failure mode condition in the booms.

TRANSFER LOADS ANALYSIS

PROPULSION MODULE (ALTERNATE)

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

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

PROPELLANT MODULE (ALTERNATE)

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

PROPELLANT MODULE

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

RNS INTERFACE MECHANISMS

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 spaceframe by the booms and positioned for structural attachment.

The structural tiedown of the propellant module is accomplished by male conical pins located on the spaceframe 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 spaceframe 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.

Assembly of the propulsion module into the frame is shown in View E on the opposite page. Final assembly is accomplished by a docking system consisting of six pins that align and engage six longeron member of the spaceframe. Final alignment and structural joining are achieved by motor-driven bolts mounted in the spaceframe longerons.

Following structural attachment, the umbilical connection between the spaceframe 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. Connection of the propellant module is accomplished in an identical fashion.

RNS INTERFACE MECHANISMS

EARTH RECYCLE CHECKOUT OPTIONS

It is envisioned that with the data management system collecting component trend data, and with built-in RNS system and component redundancy, it will be possible to establish a schedule maintenance cycle for all RNS systems. Scheduled maintenance implies that this operation is performed not only where and when, but also in consideration of the most opportune economic time when the most efficient resources of the Space Shuttle, orbiting station, space tug, and astronauts are available. This operating mode implies that preventive maintenance on particular components or systems may not be performed at the first opportunity in an earth recycle. When the RNS maintenance controller evaluates all program aspects, he may elect to delay the scheduled maintenance to the next earth recycle period. This sophisticated maintenance scheduling is possible because of the on-board data management system with its capability to monitor and collect real-time component and system data.

Earth recycle operations, therefore, consist of post-flight assessment of vehicle status and the program decision as to what requires maintenance. In this process the depth or level of component, subsystem, or system to be replaced will be determined based on the relative ease of replacement of the component versus its larger subassembly module, and this decision will most likely be influenced by the planned operational time left for that unit.

EARTH RECYCLE CHECKOUT OPTIONS

EQUIPMENT LAYOUT

The RNS is composed of three types of modules: propulsion, propellant, and spaceframe. At the top level maintenance is accomplished by replacing these modules. The propellant module and tank assembly of the propulsion module may be replaced after each mission if the tank exchange approach is used for refueling. Otherwise, these modules are exchanged on a scheduled or as-required basis. The spaceframe may also be replaced in this manner, although in general, maintenance of the spaceframe will be by replacement of the various functional subassemblies. These subassemblies are shown on the opposing page. The primary location of these elements is in the equipment module. At the upper level are six bays with the modularized functional equipment. At the next level are the thrusters, and below that are the solar panels and antenna. Within the spaceframe itself are also the bi-stem boom assemblies forward and aft. The boom assemblies, solar panel assemblies, antenna, and RCS thruster assemblies can all be replaced in orbit. The functional equipment in the bays have been assigned specific locations. Bays 1 and 4 contain the redundant RCS propellant modules. Bay 3 contains communication and data management, and Bay 6 contains the power and guidance, navigation, and control subsystems. Bays 2 and 5 have not been allocated. Bay 2, however, has the same functional interface with the spaceframe module as Bay 3. This allows for the installation of a completely redundant communication and data management subsystem. This greatly enhances the exchange capability, since the new system can be installed and checked out while the old system is still operational. The old system can then be removed without loss of continuity. In a like manner, Bay 5 provides the same redundant capability for the power and guidance, navigation, and central subsystems.

TYPICAL MODULE INSTALLATION/INTERFACE

There are a variety of methods for mounting the various assemblies in the equipment bay. They can slide in on a track with functional interfaces and latching mechanism, as shown on the next page. Several options are available for individual components. They can be either external plug-in or they can be attached to a slide-out or swing-out tray, shown as alternate No. 1; or a vertical panel with components mounted on both sides, as shown in alternate No. 2. In all cases, however, there is a common power bus and data bus so that the interfaces between the functional equipment in the bay and the spaceframe are very simple.

TYPICAL MODULE INSTALLATION /INTERFACE

EXTRAVEHICULAR WORK SYSTEMS

The RNS concept development requires a detailed consideration of the role of man in maintenance, assembly, and refurbishment operations. Man is seen as an available resource which may significantly impact system reliability, cost, flexibility, and complexity. Accordingly, an analysis was made of the extravehicular work systems available to support RNS operations, the applicability of these extravehicular modes to RNS specific functions and the nature of the crew system interface between RNS, the space shuttle, and the space station. In addition, preliminary design concepts to facilitate EVA maintenance were formulated.

Five categories of extravehicular work systems were considered as shown on the opposite page:

- a. Unaided Manual EVA - Suited astronaut activities in the free space environment utilizing handholds, tethers, and rails at a fixed work-site.
- b. Aided Manual EVA - The suited astronaut is provided supplementary means for translating to an EVA worksite, cargo or astronaut transport, auxiliary illumination, and docking/stabilization aids.
- c. Primary Vehicle/Remote Manipulator - The operator conducts operations from the interior of a primary vehicle by means of a remote manipulator system comparable to advanced underwater submersible systems.
- d. Manned Auxiliary Vehicle/Remote Manipulator - Small vehicle which encapsulates the astronaut with a remote manipulator system.
- e. Unmanned Auxiliary Vehicle/Remote Manipulator - Vehicle and manipulator are remotely controlled by a teleoperator located within a primary vehicle, the space station or a ground base.

EXTRAVEHICULAR WORK SYSTEMS

UNAIDED MANUAL
EVA

AIDED MANUAL EVA

PRIMARY VEHICLE-
REMOTE MANIPULATOR

MANNED AUX.
VEHICLE-REMOTE
MANIPULATOR

UNMANNED AUX. VEH
REMOTE MANIPULATOR

RNS PREPARED WORK SITE

At the worksite a lightweight work platform will be erected, as shown on the opposite page. This work platform incorporates "Dutch" shoes which have been proven effective for anchoring an astronaut so that he can exert maximum work forces. The platform also contains side and front rails allowing the astronaut to tether himself to the platform. The platform is designed to ride along a channel track located three feet below the edge of the equipment packages shown. The channel permits the astronaut to position himself with relative ease anywhere along the periphery of the equipment section. A control along the platform guard rail allows the astronaut to lock the platform into position. The worksite platform is stored within the space frame adjacent to the equipment bay modules.

RNS PREPARED EVA WORK SITE

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 spaceframe 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 spaceframe 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 by 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 spaceframe 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

**RNS
MISSIONS
OPERATIONS
P. HUTCHINSON**

PHASE III REFERENCE MISSIONS

The basic objective of the Mission Analysis Task was to establish representative performance and mission sizing data to support design, operations, manufacturing, test, and cost effectiveness trade studies based on candidate missions within the proposed space program recommended to the President in September 1969 by the Space Task Study Group. The range of missions examined included earth orbital and lunar shuttle applications as well as unmanned and manned planetary missions in the reusable and partially reusable modes. The latter missions were included primarily to provide insight into the impact of potential advanced mission applications on stage design and sizing while the program is still in the concept definition phase.

PHASE III REFERENCE MISSIONS

**CLASS I—LUNAR / GEOSYNCHRONOUS ORBIT SHUTTLE
MISSIONS**

CLASS II—UNMANNED PLANETARY PROBE MISSIONS

CLASS III—MANNED PLANETARY MISSIONS

LUNAR SHUTTLE MISSION

The primary application of the RNS will be to transport manned and unmanned payloads between low circular earth orbit and either lunar or geosynchronous orbit, returning in each case to the initial low orbit. The reference low earth orbit was specified as 260 nm circular having an inclination of 31.5 deg. The reference lunar orbit was 60 nm circular at 90 deg inclination and the geosynchronous a circular equatorial orbit at synchronous altitude, approximately 19,323 nm.

For performance comparison a reference lunar mission with transfer times of 108 hr outbound and 72 hr return was specified by MSFC with a 30 deg plane change maneuver at both lunar orbit arrival and departure. Mid-course correction maneuvers of 50 fps were to be included for each transfer. A flight performance reserve (FPR) equal to 0.75 percent of the mission delta-V was also to be included.

A nominal lunar mission was defined for the purpose of RNS design and mission operation definition, based on transfer opportunity schedules and assuming zero plane change at LOI. Two such transfers occur during each 54.64-day cycle. In one case a return opportunity after approximately 15 days occurs, varying from a 2-deg departure plane change at 108 hr transfer to 49 deg at 48 hr. For the alternate case a 0-deg departure occurs after about 17 days for transfer times of about 80 hours with modest plane changes (14° to 16°) required for earlier or later departures. A 30-deg departure plane change with a 72-hr transfer time was selected for the nominal mission, since this capability would provide ample performance margin to allow either opportunity to be selected.

The velocity budgets used in the LMSC performance analyses are shown in the figure. To these must be added the gravity losses, which vary as a function of thrust-to-weight ratio, and the FPR allowances. Three-burn lunar arrival and departure maneuvers were utilized assuming a 36-hr intermediate elliptical orbit.

LUNAR SHUTTLE MISSION

TRANSEARTH INJECTION

72 HR TRANSFER

$\alpha = 30^\circ$

XFR TO 36 HR	1917 FPS
30° PLANE CHG	318 FPS
TRANSEARTH INJ.	1340 FPS
MIDCOURSE	50 FPS
TOTAL TEI ΔV	3625 FPS

EARTH ORBIT INSERTION

$\alpha = 0^\circ$

EOI ΔV = 10,158 FPS

LUNAR ORBIT INSERTION

NOMINAL: $\alpha = 0^\circ$

LOI ΔV = 2874 FPS

ORBITING LUNAR STA
60 NM CIRCULAR
POLAR ORBIT

REFERENCE: $\alpha = 30^\circ$

LOI (36 HR) 1025 FPS

30° PLANE CHG 333 FPS

XFR TO CIRC 1917 FPS

TOTAL ΔV 3275 FPS

260 NM 31½° INCLINED
CIRCULAR ORBIT

EARTH

TRANSLUNAR INJECTION

108 HR TRANSFER $\alpha = 0^\circ$

TLI ΔV = 10,051 FPS

MIDCOURSE 50 FPS

TOTAL ΔV = 10,101 FPS

TRANSFER OPPORTUNITIES

Transfer opportunities for coplanar earth arrivals and departures were established during Phase II for synchronized earth orbits (258.9 nm x 31.7 deg inclination), which are close enough to the reference earth orbit specified for the Phase III analysis to be considered valid. These opportunities are presented for two earth orbits, designated "A" and "B", whose nodal lines are oriented 90 deg from each other and represent the two orbits the RNS would operate from if no support systems remain in earth orbit after RNS departure. If orbital facilities such as propellant depots, engine shields, or maintenance facilities are left in earth orbit all transfers would be made to and from a single earth orbit.

In determining the schedule of transfer opportunities, an initial in-phase condition for orbit "A" and the lunar orbit, corresponding to a 96-hour flight time was assumed. Lunar declination at arrival was taken as 0-deg at the time of the ascending node. Opportunities were determined only for those times when the earth orbit was aligned for in-plane arrivals and departures. Orbit "B" in-phase conditions are the same except that lunar declination at arrival was 0-deg at the descending node.

TRANSFER OPPORTUNITIES

LUNAR SHUTTLE MISSION

ORBIT 'A'

ORBIT 'B'

NERVA THRUST-TIME PROFILE

The NERVA thrust-time profile, particularly the post-operational cooldown phase, has a significant effect on the RNS mission operations. The startup phase includes engine temperature conditioning, nuclear startup, bootstrap, and thrust buildup covering a time period of about 160 seconds, of which some 130 seconds is needed for the chardown. Steady-state (SS) operation is a variable and must be regulated by the NG&C system based on predicted thrust-time profiles for the pump tailoff (PTO) and cooldown phases which themselves are functions of time at steady-state power and the throttle hold. PTO times can vary from 35 seconds for short SS runs of 20 seconds up to 20 minutes for SS times of 3000 seconds. The cooldown phase for the same SS times would vary from 9 hours up to 8 days and involve total impulse in the 2×10^5 to 5×10^6 lb-sec range.

To minimize the performance degradation effects of this extended cooldown time, as much as possible of the impulse must be used to satisfy mission requirements.

NERVA THRUST-TIME PROFILE

TLI MIDCOURSE CORRECTION

For most of the RNS propulsive maneuvers the recovered cooldown impulse will be used to provide a portion of the velocity increment required for the maneuver. For the translunar injection burn the long steady-state operating times (1500 to 2000 sec) and the resulting long cooldown periods (4.0 to 5.5 days) complicate the cooldown impulse recovery, since the RNS will pass beyond the trajectory region in which effective use of the available ΔV can be made. One alternative is to use a portion of the cooldown pulses to perform any midcourse correction required to eliminate TLI injection errors.

An evaluation confirmed that the cooldown impulse could be effectively used in this manner based on TLI injection errors characteristic of lunar transfer. For the case evaluated it was found that about 31 pulses would be needed, starting 15 hours after engine shutdown and continuing for about 23 hours.

TLI MIDCOURSE CORRECTION

THRUST NULLING MODES

Thrust nulling of the cooldown pulses can be accomplished in a variety of ways, four of which were considered; direct opposition by the RCS thrusters, discharge through opposing thrust nozzles instead of the main nozzle (engine alternate flow path), stage rotation with an interrupted or split pulse, and a trajectory compensated mode in which the RNS orientation is periodically changed by 180 deg.

The engine alternate flow path is the most attractive in that it permits complete flexibility in RNS orientation, but has the disadvantage of increasing the complexity of the engine flow system. Direct RCS opposition permits freedom in selecting orientation, but imposes severe penalties on the RCS in terms of propellant weight and total operating time.

The remaining two methods are not true nulling modes, but rather seek to eliminate the effect of the thrust on the trajectory. In the rotational mode the stage is rotated at a slow rate about an axis normal to the thrust vector and the pulse split so that the second half removes any perturbation caused by the first half. The main objection to this mode is the loss of freedom to select vehicle orientation during the cooldown phase.

In the trajectory compensated mode the vehicle flight path is initially altered to compensate for subsequent cooling pulse perturbations with the RNS holding a preselected orientation. If the RNS orientation is periodically reversed, the trajectory would appear to undulate about the nominal. Assuming reasonable accuracy in pulse train prediction, this mode appears well suited to the mission demands.

THRUST NULLING MODES

DIRECT RCS OPPOSITION

ENGINE ALTERNATE FLOW PATH

ROTATIONAL MODE
SPLIT PULSE

TRAJECTORY
COMPENSATED

VEHICLE ORIENTATION

Vehicle orientation throughout the mission is significant 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 and engine alternately face the sun (trajectory compensated mode), unless NERVA cooldown thrust nullifiers are incorporated, and therefore the RNS flies a trajectory that undulates about an ideal path. During the lunar orbit phase and during coast, the payload faces the sun.

VEHICLE ORIENTATION

LUNAR MISSION TIMELINE

A timeline of propulsive events is shown for a typical round-trip lunar shuttle mission. The example follows the profile of the "nominal" mission with earth departure from orbit "A" at an opportunity for in-phase lunar orbit insertion. Trans-earth injection occurs 15 days later when an opportunity exists for return to orbit "A" with 30-deg plane change at lunar departure. An opportunity for a 72-hr return to orbit "A" also exists about 4.5 days after LOI with a plane change of 17 deg; however, when LOI cooling time (about 24 hours) and 3-burn departure maneuver time (24 to 36 hours) are considered the time in lunar orbit may be somewhat brief for payload transfer maneuvers.

The propulsive maneuvers assume a 75-percent recovery of pulse flow cooling impulse. Unused pulses would be "nulled" in the trajectory compensated mode. The 30-deg plane change at TEI (LM2 burn) assumed NERVA partial power operation with the TPA operating. The return midcourse maneuver assumed NERVA idle mode operation without use of the TPA.

LUNAR MISSION TIMELINE

(EVENT TIMES IN SEC)

SU - STARTUP
SS - STEADY STATE
SD - SHUTDOWN
PTO - PUMP TAILOFF
PFC - PULSE FLOW COOLING

LOI USING COOLDOWN IMPULSE

Recovery of the pulse flow cooling impulse for orbit insertion at Earth and Moon will produce trajectories which approximate spirals. The number of orbit revolutions required to complete the maneuver will depend on the steady-state operating time for the main burn and the percent recovery planned. These factors will also be of concern in determining the radiation dose which would be received at an orbiting station, assuming the RNS completes the insertion maneuver in the near vicinity of the orbiting station.

For the nominal mission the RNS will complete about 1.5 revolutions during the 75-percent impulse recovery at lunar orbit insertion. At EOI the RNS will complete about 2.5 revolutions for a similar insertion. If more than 75-percent recovery is planned, the insertion time and number of orbit revolutions will increase significantly because of the ever increasing interval between subsequent pulses. Pulse spacing for the LOI and EOI burns is about 15 to 20 minutes at the 75 percent cutoff time.

LOI USING COOLDOWN IMPULSE

108 HOUR TRANSFER TRAJECTORY

THREE-BURN LUNAR DEPARTURE

Of the four propulsive maneuvers to be performed on the lunar shuttle mission (TLI, LOI, TEI and EOI) the lunar departure is the one most apt to require out-of-plane maneuvering, since the penalties in terms of propellant expended are smaller for this maneuver than for the other three. The impact of finite thrusting and extended cooldown periods on a three-burn departure maneuver, including a 90-deg plane change, are shown. Lesser plane change trajectories would be the same except for the middle burn, which would require less impulse.

Of the three burns only the first burn would be of concern from the standpoint of radiation dose to the OLS, and then only if it is performed in the near vicinity of the station.

The second and third burns take place at such high altitudes and are so short in duration that no measurable dose is received at the OLS.

THREE-BURN LUNAR DEPARTURE

INCIDENT DOSE TO SPACE ELEMENTS

The radiation dose from neutron and gamma radiation which would be measured at the outer surface of orbiting stations in earth or lunar orbit was calculated for each of the four propulsive maneuvers of the lunar shuttle mission. The radiation environments included both the neutron and gamma emissions during NERVA full-power operation and the post-shutdown fission product gamma radiation. Separation distance and view angles were used for typical lunar mission maneuvers.

Recovery of the aftercooling impulse, while it delays orbit insertion, has the favorable safety effect of placing the RNS at a considerable distance from the orbiting stations during the main burn. Insertion ahead of the station, with the RNS in a retro attitude, provides the most favorable view angle during the extended insertion maneuver, with the result that the allowable incident dose of 0.100 Rem stated in the guidelines is never approached.

Departure from positions ahead of the space stations results in a view angle for maximum dose rate, but avoids any close flyby during or shortly after the burn. For these maneuvers the guideline dose criteria can be satisfied with mode initial separation distances.

RNS orbit parking separation distances of 10 km would be sufficient to preclude undesirable doses unless extended times at the worst view angle were required. Closer positioning would be permitted, but may require continuous RNS attitude control to take advantage of the reduced radiation field in the forward direction provided by the engine shield.

INCIDENT DOSE TO SPACE ELEMENTS

LUNAR SHUTTLE MISSION

SPACE ELEMENT	MANEUVER	RNS POSITION AT START OR END	INCIDENT DOSE (REM)
EOSS	TLI	56 NM AHEAD	0.100
OLS	LOI	10 KM AHEAD	0.0002
OLS	30 DAY ORBIT STAY	10 KM DISTANCE 90° VIEW ANGLE	0.130
OLS	TEI	36 NM AHEAD	0.100
EOSS	EOI	15 NM AHEAD	0.006

EMERGENCY MODE MISSION ABORT

The NERVA engine is designed to operate in an emergency mode for a single operating cycle within the engine operating envelope at or above the minimum conditions indicated. A study was conducted to determine whether sufficient capability exists in the emergency mode to place the RNS in an orbit from which rescue and/or recovery could be accomplished.

An emergency was assumed to exist during each of the four indicated mission points, just prior to shutdown or just after startup, requiring immediate application of the emergency mode operation. Vehicle weights and available propellant were taken from the nominal mission with sufficient propellant for final cooldown held in reserve. Pericenter altitudes of 260 nm at earth and 60 nm at the moon were used as target conditions following a simple guidance program to achieve the minimum ellipticity orbit.

All available propellant was consumed for each case except LOI for which minimum conditions were satisfied earlier under the simplified guidance constraints used. A more optimum program would result in a somewhat improved orbit. The earth orbits achieved might also be improved, but must be held at or above 260 nm x 3140 nm required for safe orbit lifetime.

Maximum operating time in the emergency mode would be about 2000 seconds (6×10^7 lb-sec impulse) for the TLI abort. Assuming a suitable rescue vehicle were provided in earth or lunar orbit it appears that the NERVA emergency mode operation as specified is sufficient to meet crew safety requirements throughout the mission.

EMERGENCY MODE MISSION ABORT

NERVA EMERGENCY MODE

THRUST 30,000 LB
ISP 500 SEC
TOTAL IMPULSE 10^8 LB-SEC

ABORT POSITION	ABORT MODE	FINAL ORBIT
TLI SHUTDOWN	RETRO TO EARTH ORBIT	260 NM X 7350 NM
LOI STARTUP	BRAKE INTO LUNAR ORBIT	60 NM X 530 NM
TEI SHUTDOWN	RETRO TO LUNAR ORBIT	60 NM X 1100 NM
EOI STARTUP	BRAKE INTO EARTH ORBIT	260 NM X 6050 NM

SHIELDING VALUE OF ABORT RESERVES

If no failures were encountered during the mission, the emergency propellant reserve carried would remain in the tank and while it was not used in the mission it would provide radiation protection to the payload in much the same manner as added shielding. The shielding effects of such residual liquid were determined for three RNS configurations: the baseline 7-tank vehicle and two alternate cluster versions. All cases were evaluated for an incident dose at the payload of 10 rad. The higher shield weights of the baseline vehicle are related to scatter from the outer tanks, and while some reduction is possible, the improvement is limited. For the alternate cases in which the scatter from the outer tanks is minimized, the need for added shielding would be eliminated if 6000 to 7000 lb of residuals remained in the tank at final shutdown.

For return to 260 nm orbit in the malfunction mode an emergency reserve of about 2000 lb would be required. Engine shield weight reductions would vary from 700 to 1600 lb for the three cases, but shield savings could not offset the performance loss of such a reserve.

(NOTE: In all cases, it is assumed that the reserves are in addition to the propellant normally remaining in the tank for final cooldown - in this evaluation 3,000 lb)

SHIELDING VALUE OF ABORT RESERVES

MALF MODE MISSION ABORT

The NERVA engine is also capable of malfunction-mode (MALF) operation (operation with a single TPA) in the event of a malfunction which prevents dual TPA operation. In the MALF mode the engine may be operated at rated specific impulse and 80 percent rated thrust and is capable of meeting normal startup, shutdown, and throttling requirements. Indication of impending failure might also require operation at reduced I_{sp} and thrust, but permit dual TPA flow. This condition was also considered. It is assumed that the mission is aborted when the malfunction is detected and the RNS returned to 260 nm circular earth orbit. The amount of reserve propellant which must be carried to provide this earth return capability was determined so that its effectiveness in terms of added shielding could be evaluated. The reserve propellant requirements shown reflect a retreat to the MALF mode at EOI startup since this maneuver has a higher demand than do the other maneuvers. A thrust level of 60,000 lb was assumed for all cases.

The propellant increase, including velocity losses associated with reduced thrust, are shown by the solid line and represent the upper bound of the reserves required. The dashed curve considers I_{sp} effects only and represents the lower bound. Thrust levels between 60 and 75,000 lb would lie between these bounds. Single TPA operation (malfunction mode) would require operation at near rated I_{sp} . Lower I_{sp} operation requires two TPAs to satisfy the indicated flow rates. Degraded I_{sp} operation would be consistent with engine failures in which indication of impending failure would require reduced reactor power, but would not limit the flow rate.

For return to 260 nm orbit in the malfunction mode a reserve of about 2000 lb would be required, while reduced I_{sp} reserves would be even higher. Performance losses associated with such reserves would be severe since not only must the additional weight be carried round trip, but the available impulse propellant for the mission would be reduced by the same amount.

MALF MODE MISSION ABORT

RETURN TO 260 NM EARTH ORBIT

RESCUE MISSION PROFILE

During all of the geosynchronous orbit shuttle mission and most of the lunar shuttle mission, a NERVA thrust failure would leave the RNS stranded in earth or lunar orbit. Assuming the crew module was undamaged and had adequate life support for many days survival, no immediate hazard would exist as long as a suitable vehicle were available in earth or lunar orbit with which a rendezvous and rescue operation could be performed. At two points in the lunar mission, however, loss of RNS propulsion would leave the vehicle on a trajectory requiring prompt rescue initiation in order to avoid crew loss. The most serious condition would arise if the lunar orbit insertion burn were missed, which could result in an RNS lunar flyby and escape into heliocentric orbit.

The other critical point would be the earth-orbit insertion maneuver, and while the RNS would remain in earth orbit following the unpowered flyby, the geometry of the orbit and long period would require immediate rescue vehicle deployment unless survival times of two weeks or so could be assured.

For the analyses it was assumed that the rescue vehicles were in 260 nm circular earth orbit and 60 nm polar lunar orbit and that the RNS arrival was coplanar with these orbits. The rescue vehicle would be required to execute two propulsive maneuvers to rendezvous with the RNS.

For long times to rendezvous the first burn would occur approximately under the perigee position of the RNS or orbit, but would move toward the RNS as faster times are selected. A shift of about 30 degrees was noted for the fastest trajectories presented. The second burn of the rescue vehicle will occur at RNS intercept and will be used to match the velocity vectors for the two vehicles. After transfer of the crew to the rescue vehicles two more burns would be required to complete the transfer back to the circular orbit.

The rescue mission profile for the missed LOI burn would be similar to the EOI case except that the RNS would be on an escape trajectory after the lunar flyby.

RESCUE MISSION PROFILE

FOR MISSED EOI MANEUVER

RESCUE VEHICLE REQUIREMENTS

The velocity requirements which would be imposed on a rescue vehicle for missed EOI and LOI burns was computed to determine if such a rescue capability represented a reasonable approach to satisfying crew safety requirements. For the LOI rescue the time from rescue vehicle departure to RNS rendezvous was assumed to be 12 hours. Longer rendezvous times would reduce the outbound velocity requirement, but the reduction would be too small to compensate for the increased total rescue mission time. Velocity to return to lunar orbit was fixed at 5500 fps, which is near minimum unless excessively long return times are considered. Total velocity requirements and total elapsed time are presented for various delays in rescue vehicle departure. All cases allow one hour for crew transfer after rendezvous.

The EOI rescue vehicle requirements were determined in a similar manner except the time from rescue vehicle departure to rendezvous was increased to 18 hours and return velocity fixed at 16000 fps.

Although total rescue mission times are comparable for both cases, the high velocity requirements for EOI rescue would demand a very large vehicle and may represent a greater price than is practical to enhance crew safety. Assuming conventional chemical vehicles with I_{sp} of 450 seconds propellant capacities of 10,000 lb and 100,000 lb would be representative of lunar and earth orbit requirements.

From these simple evaluations it appears that a standby rescue vehicle in lunar orbit, perhaps a fueled and ready lunar tug, would be a practical method of assuring maximum crew safety. Since this maneuver is the most critical in terms of potential crew loss, it would probably be a worthwhile mission requirement.

RESCUE VEHICLE REQUIREMENTS

DELAY TIME (HR)	EOI RESCUE		LOI RESCUE	
	ΔV (FPS)	TIME (HR)	ΔV (FPS)	TIME (HR)
0	26,000	44	8100	46
2	27,100	47	8950	53
4	28,200	50	9800	59
6	29,200	53	10500	66

POPULATION SAFETY

During the early portion of the EOI burn the RNS will be in an extremely long life orbit, but as retro thrust is applied this lifetime is rapidly reduced until at 260 nm circular orbit the average lifetime for an intact empty RNS would be about 1.7 years. For single fuel elements tumbling in the same orbit a lifetime of about 0.42 years would be expected.

To evaluate the relative degree of hazard for various conditions the exclusion radius is used, defined as that distance from a reference source within which a dose of D or greater will be received over a specified continuous exposure interval. In the evaluation a single fuel element was selected as the reference source, on the assumption that an element would be the largest core fragment which might be expected to survive reentry intact.

If the RNS and engine remain intact it could be tracked until orbit decay eventually causes re-entry. In such a case the debris would all be located in one area and prompt countermeasure action could be employed to retrieve the debris or exclude people from the area. Four-day exposure data would be characteristic of such a case. On the other hand, if the reactor broke up in orbit the debris would be difficult if not impossible to track. The random re-entry of the debris would not result in a single discrete area to be decontaminated, but would be distributed throughout the latitude band contained within the orbital trace. Since considerable time might elapse before such fragments were discovered, one year exposure data would be a better measure of the relative hazard of the orbital debris type abort.

If a catastrophic failure occurred in 260 nm circular orbit after the 10th lunar mission which resulted in engine breakup, the exclusion radius for an MLD50 dose of 450 Rem would be 7.2 meters, the maximum for any case examined. However, since reactor disassembly generally assumes a loss-of-coolant type accident when decay heat rates are high, such failures would be very remote after pulsed flow cooling is initiated. Perhaps a better estimate of the exclusion radius for the MLD50 dose would be 2 meters, corresponding to the end of pump tailoff for the same case.

If the RNS and engine remain intact, whole body gamma exposures to the population in the 12 to 50 Rem range, considered to be the threshold of detectability, would be characterized by exclusion radii less than two meters and would be considered very unlikely events.

POPULATION SAFETY

EO I AFTER 10TH LUNAR MISSION

REFERENCE MISSION PERFORMANCE

Performance data are presented for the Phase III lunar and geosynchronous missions. Two RNS configurations were used in the evaluation: the baseline 7-tank RNS and the 9-tank alternate design.

Gravity losses and FPR allowances were included in each burn. Propellant requirements for full-power engine burns simulate engine startup, steady-state, shutdown, pump tail-off, and pulse flow cooling on an equivalent impulse basis using Phase III engine data as provided by Aerojet Nuclear Systems Company. For the evaluation, a 75-percent recovery of the impulse from the pulsed flow cooling was assumed. Boiloff and vapor residuals were computed based on nose-to-the-sun orientation. Residual vapor was dropped at the completion of individual tank drains.

Midcourse maneuvers assumed idle-mode operation at an average I_{sp} of 480 sec, except for the TLI midcourse which was accomplished using cooling pulses.

The middle burn of the three burn LOI and TEI maneuvers are too short for normal full-power engine operation and were simulated as partial power burns, with an average I_{sp} of 480 sec. Tank pressurization and TPA operation were assumed for the partial power burns.

REFERENCE MISSION PERFORMANCE

LUNAR SHUTTLE

GEO SYNCHRONOUS SHUTTLE

PLANETARY PROBE PERFORMANCE

The combination of a reusable nuclear shuttle and an expendable chemical upper stage provides an attractive vehicle for boosting large payloads to escape on missions with medium to high energy requirements. The performance capability for this combination was determined using the interorbital data for the baseline 7-tank RNS and a hypothetical chemical stage.

For missions below 30,000 fps excess velocity, substantial increases in payload weight can be realized by staging in lower elliptical orbit using larger chemical stages.

Although the performance capability of the RNS-chemical combination is generally greater if low orbit staging is selected, the ratio of payload to chemical stage propellant, having a direct relationship to mission cost, is significantly improved for the higher orbit staging. Thus, staging should be accomplished in the highest orbit consistent with payload demands.

For higher energy missions the payload becomes insensitive to intermediate orbit period, and higher staging would be preferred to minimize chemical stage size.

For comparison purposes the payload which can be delivered by the RNS alone is presented for the reusable and expendable modes, the latter being of interest only as a last mission for the RNS.

In the reusable mode the RNS performs three propulsive maneuvers, the first of which boosts the payload to escape. The second assumed to occur 1 hour after first-burn shutdown, retros the RNS into a 48-hr elliptical orbit with a perigee altitude of 260 nm. The third burn occurs at perigee, retroing the RNS into circular orbit.

Gravity losses and effects of NERVA operating characteristics were included in the analyses.

PLANETARY PROBE PERFORMANCE

BASELINE RNS - $W_p = 269.4 \text{ K LB}$

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 $\odot$	60
LM	2446560	7422	12 HR ϵ $\odot$	150
AE	2446710	3273	24 HR ϵ $\oplus$	

27100

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 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 150-fps for 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 ϵ $\oplus$	340
AM	2448460	3722	12HR ϵ $\odot$	360
LM	2448820	4339	12HR ϵ $\odot$	300
AE	2449120	2806	24HR ϵ $\oplus$	

14986

24 HR ELLIPTICAL
ARRIVE/DEPART
ORBIT

EARTH

260 NM CIRCULAR
ASSY ORBIT

EXPENDABLE
TIER (3 TANKS)

INTERPLANETARY RNS
(7 TANKS)

RETURN
TO EARTH

MARS

12 HR ELLIPTICAL
RNS ORBIT

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

OFF-LOADED INTERORBITAL PAYLOAD

If an RNS is used to transfer the vehicle and payload segments for the Mars mission from the 260 nm assembly orbit to the elliptical earth departure orbit, it will rarely be possible to size these segments to the optimum RNS interorbital performance capability. To assist in evaluating the total propellant requirements for the Mars missions, performance data for the off-loaded baseline RNS were generated for one-way payload transfers to 24 hour and 48 hour elliptical orbits.

As in previous analyses the performance data account for gravity losses and simulate NERVA startup, steady-state, shutdown, pump tailoff, and pulsed flow cooling. A 75 percent recovery of pulsed cooling impulse was assumed.

OFF-LOADED INTERORBITAL PAYLOAD

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 inter-orbital (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 (NUTOP) 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 the three I/O trips can be accomplished in 56 days with only one non-coplanar transfer. The I/O RNS for 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 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 interorbital (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 325K 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

*~13 #/day
TRIP Term
1060000
pg 4-43*

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

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

**RNS
PROGRAM
REQUIREMENTS
C. HOPKINS**

RNS MASTER PHASING SCHEDULE

The RNS Master Program Phasing Schedule reflects LMSC's current philosophy for implementation of Phases B, C, and D for the Reusable Nuclear Shuttle through IOC and are responsive to NASA long-range planning. This schedule depicts the phasing of resources and program objectives. They are time-phased from right to left, backward from IOC and the first operational launch which is 51 months after start of Phase D (or eighty-four months from Phase B go-ahead). Built into each schedule is a 3-month time span between Phases B and C and C and D to permit NASA sufficient time to compute contractor performance evaluation on the preceding phase and for proposal evaluation and selection of contractor(s) for the succeeding phase.

The ground rules used were that Phase B would have a duration of 12 months followed by 3-month NASA evaluation period, at the completion of which Phase C would start. Phase C has a 15-month period of performance, also followed by a 3-month period for contractor and proposal evaluation, followed by Phase D go-ahead.

Some of the assumptions utilized in schedule preparation were: PDR would occur at two months into the Phase C schedule and one month after the preliminary requirements review (PRR), thus allowing time to update the Part I CEI specification prior to starting detail design; a preliminary CDR to be computed for long lead items at seven months into Phase C to allow for long lead procurement; that final CDR would occur at the end of Phase C; that a series of FACI's would be held due to different completion dates of flight test hardware; that a series of flight readiness reviews would be accomplished in accordance with the start of deployment and completion of pre-deployment operations; and that the first operation vehicle (OV-1) would be built and ready for deployment operations coincident with IOC.

REUSABLE NUCLEAR SHUTTLE MASTER PHASING SCHEDULE

ASSUMPTIONS AND GUIDELINES

The cost analysis was performed within the constraints of the assumptions and guidelines listed here. The costs so derived represent only the estimated investment in the vehicle aspects of RNS evolution, since NERVA facilities and R&D costs are omitted. The costs also represent only the investment made after commitment to a hardware program. Phase A and Phase B development costs were omitted because these expenditures depend on programmatic factors such as the number of contractors selected and the time available to perform the studies.

ASSUMPTIONS AND GUIDELINES

NASA SUPPLIED GUIDELINES

- BASELINE RNS MISSION RATE 6 LUNAR ROUND TRIPS/YR
- ALTERNATE RATES OF 2,4, AND 8 LUNAR MISSION/YR
- BASELINE SPACE SHUTTLE PAYLOAD 25K TO REF. ORBIT
- ALTERNATE SPACE SHUTTLE CAPACITY 40K
- HARDWARE LIFETIME 10 MISSIONS
- NERVA R&D COSTS EXCLUDED
- NERVA UNIT COST \$13M
- SPACE SHUTTLE COST \$5M PER FLIGHT
- FLIGHT TEST ARTICLE IS EXPENDED AFTER TEST PROG.
- ALL COSTS IN 1971 DOLLARS

LOCKHEED ASSUMPTIONS

- RNS COSTS QUOTED FOR PHASE C AND BEYOND
- HARDWARE QUANTITIES TOO LOW FOR LEARNING EFFECTS
- MANUFACTURING FACILITIES ASSUMED AVAILABLE

APPROACH

A systematic approach was used in costing the Reusable Nuclear Shuttle system. The basis for this analysis was the refined RNS vehicle design evolved in the Phase III study. Updated and cross-checked cost estimating relationships were applied against the technical characteristics of the RNS concept and these costs served as the basis for all subsequent calculations. The Work Breakdown Structure served as the control mechanism for all configuration- and cost-related data.

APPROACH

TECHNICAL CHARACTERISTICS CORRELATED TO COSTS

- WORK BREAKDOWN STRUCTURE AS MECHANISM

COST ESTIMATING RELATIONSHIPS APPLIED TO TECHNICAL CHARACTERISTICS

- UPDATED FOR RECENT COST ESCALATION
- CROSS-CHECKED AGAINST AEROSPACE CORP. CER'S

HISTORICAL COST FACTORS USED TO CALCULATE FUNCTION-ORIENTED COSTS

- RATIOS OF FUNCTION COSTS TO HARDWARE COSTS

ALL COSTS COMPUTER SUMMED BY WBS LEVELS

NONRECURRING COSTS

The costs for Phase C/D development of the RNS through flight test are listed in the figure opposite. These costs are stage-related expenditures only; in particular, it is assumed that the facilities costs are limited to that funding needed to provide brick-and-mortar installations sufficient for RNS module processing.

Major cost drivers in the nonrecurring costs include test hardware, and flight-test operations and services.

NONRECURRING (DEVELOPMENT) COSTS

ITEM	COST(\$ MILLIONS)
RNS HARDWARE	
● PROPULSION MODULE	92.7 *
● PROPELLANT MODULE	102.9
● CONTROL & ASSY STRUCTURE MODULE(SPACEFRAME)	166.1
GSE	46.8
SYSTEMS ENGINEERING/INTEGRATION	50.9
PROGRAM MANAGEMENT	16.0
FACILITIES	15.0
TEST HARDWARE	175.0
TEST OPERATIONS	42.1
LAUNCH OPERATIONS(FLIGHT TEST)	77.0
MISSION OPERATIONS(FLIGHT TEST)	77.6
TOTAL	862.1

* LESS NERVA ENGINE R&D

RECURRING PRODUCTION COSTS

The figure opposite details the recurring-production (or investment phase) costs required to purchase a fleet of RNS vehicles sufficient to support a 10-year operational program. The costs quoted are for an inventory that would sustain a mission rate of 6 lunar roundtrip flights per year for this 10-year program.

Major cost drivers in this expenditure include the NERVA engine, the astrionics modules, and vehicle spares. The spares costs include one equivalent RNS to be available at the launch base as a backup in case of equipment failures or schedule problems.

The investment costs can be reduced by approximately \$44 million (just under ten percent) by refurbishing the flight test article to serve as part of the operational RNS inventory.

RECURRING PRODUCTION COSTS

ITEM	COST (\$ MILLIONS)	
	UNIT	TOTAL AT 6/YR FOR 10 YRS
RNS HARDWARE		
● PROPULSION MODULE	19.7	124.4 *
● PROPELLANT MODULE	4.0	149.6 *
● CONTROL & ASSY STRUCTURE MODULE(SPACEFRAME)	13.4	92.4 *
GSE MAINTENANCE		14.0
SE & I (SUSTAINING)		11.2
PROGRAM MANAGEMENT		3.5
SPARES		66.4
TOTAL		461.5

* INCLUDES TOOLING MAINTENANCE

RECURRING OPERATIONS COSTS

The recurring operations costs encompass all expenditures associated with the 10-year operation of an RNS lunar transportation system. The costs in the figure opposite are quoted for a 6-flight-per-year mission rate.

These costs are dominated by the Space Shuttle launch and resupply requirements, and most particularly by the propellant-delivery costs.

RECURRING OPERATIONS COSTS

ITEMS	COST (\$ MILLIONS)
RNS	
● PROPULSION MODULE	8.3 *
● PROPELLANT MODULE	4.9 *
● CONTROL & ASSEMBLY STRUCTURE MODULE	15.6 *
GSE MAINTENANCE	30.4
SE & I (SUSTAINING)	92.6
PROGRAM MANAGEMENT	29.2
FACILITIES MAINTENANCE	5.3
LAUNCH OPERATIONS	440.0
MISSION OPERATIONS	2,896.3
TOTAL	3,522.6

* TOOLING MAINTENANCE

SPACE SHUTTLE PAYLOAD CAPACITY EFFECT ON RNS TOTAL COST

The most promising area for cost avoidance in the RNS system is the Space Shuttle propellant-delivery expenditure. To measure the extent of the savings that would be made possible by matching the payload capability of the Space Shuttle to the weight of a fully-loaded propellant or propulsion module, a comparative evaluation was made of RNS total program costs for systems supported by Space Shuttles of 25 and 40K payload capacity (to the Space Station reference orbit).

The results, shown in the figure opposite, indicate savings of from 16 to 22 percent for an RNS system supported by a 40K shuttle. More important, the dollar value of these savings ranges up to \$1.3 billion for the 10-year program.

These savings result from the fact that, with a 25K shuttle, the propellant and propulsion modules must be delivered to orbit offloaded and it then requires three additional Space Shuttle flights per RNS mission to top off the propellants. Thus, the 40K Space Shuttle, which can deliver fully loaded modules, saves 3 resupply flights per RNS launch.

SPACE SHUTTLE PAYLOAD CAPACITY

EFFECT ON RNS TOTAL COST

TOTAL PROGRAM COST (\$BILLIONS)

RNS COST EFFECTIVENESS

The cost effectiveness of an RNS system in delivering payload to lunar orbit is shown in the figure opposite. The cost effectiveness parameter, dollars per pound of payload delivered to lunar orbit, is plotted as a function of the number of missions per year over the 10-year program. Two sets of curves are shown; the lower set includes only costs of the operational program (i. e., recurring production and recurring operations), whereas the upper set includes also the amortized nonrecurring costs of RNS development. Two curves are shown in each set, one for an RNS system supported by a 25K Space Shuttle, and one for a system using the 40K Space Shuttle.

The results of this analysis show that an RNS cost effectiveness of \$623/lb can be achieved with the 40K shuttle, and that this figure increases to only \$760/lb when RNS development costs are amortized against the program.

RNS COST EFFECTIVENESS

CONCLUSIONS AND RECOMMENDATIONS

C. D. McKEREGHAN

CONCLUSIONS

- MODULAR RNS DESIGN IS FULLY COMPATIBLE WITH OVERALL OPERATIONAL REQUIREMENTS
- SPACEFRAME CONCEPT PROVIDES INDEPENDENT ASSEMBLY PLATFORM, ALIGNMENT AND CONTROL, AND CONFIGURATION FLEXIBILITY
- OPERATION IN EARTH ORBIT REQUIRES ONLY SPACE SHUTTLE FOR SUPPORT
- RNS SYSTEMS AND MISSION DEFINITION SATISFY THE SAFETY CRITERIA ESTABLISHED FOR THE STUDY
- OVERALL RNS DESIGN IS CONSISTENT WITH SPACE SHUTTLE STATE OF TECHNOLOGY
- RNS OPERATIONALLY AVAILABLE APPROXIMATELY FOUR YEARS AFTER PHASE D GO-AHEAD
- DEVELOPMENT FUNDING REQUIREMENTS THROUGH FLIGHT TEST ARE \$860 MILLION, EXCLUDING NERVA DEVELOPMENT

RECOMMENDATIONS

- CONTINUE NERVA DESIGN INTEGRATION
- EXTEND ANALYSIS OF GUIDANCE AND CONTROL CONTROL SYSTEM REQUIREMENTS, REFLECTING NERVA CHARACTERISTICS
- EXPAND ASSESSMENT OF ORBITAL OPERATIONS, WITH EMPHASIS ON MAINTENANCE AND PAYLOAD INTEGRATION
- REFINE SPACE SHUTTLE INTEGRATION AND INTERFACE DEFINITION, INCLUDING TANKER VERSION OF SPACE SHUTTLE
- INVESTIGATE ALTERNATE SOLUTIONS TO MULTIPLE-TANK PROPELLANT MANAGEMENT SYSTEM
- INITIATE DEVELOPMENT ACTIVITIES PROGRAM IN FIELD OF ASSEMBLY AND DEVELOPMENT MECHANISMS (INTEGRATE WITH SPACE SHUTTLE PROGRAM)
- INITIATE PLANNING FOR ORBITAL PROPELLANT TRANSFER EXPERIMENT IN CONJUNCTION WITH SPACE SHUTTLE PROGRAM