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# OPERATIONALLY EFFICIENT PROPULSION SYSTEM STUDY(OEPSS) DATA BOOK

## Volume VI Space Transfer Propulsion Operational Efficiency Study Task of OEPSS

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## FOREWORD

This document is the final report for the Space Transfer Propulsion Operational Efficiency Study Task of the Operationally Efficient Propulsion System Study (OEPSS) conducted by the Rocketdyne Division of Rockwell International. The study was conducted under NASA contract NAS-10-11568, and the NASA Study Manager is Mr. R. E. Rhodes. Technical assistance was received from R. Zurawski and P. Richter, NASA LeRC and N. Munoz, from NASA JSC. The Rocketdyne Program Manager was R. P. Pauckert, the Deputy Program Manager was G. Waldrop, and the Project Engineer was T. J. Harmon. Assistance was rendered by J. Ziese, Space Systems Division of Rockwell International, and R. Beach of General Dynamics. The period of study was from January through October 1992.

## SUMMARY

The Space Transfer Propulsion Operational Efficiency Study task studied, evaluated and identified design concepts and technologies which minimized launch and in-space operations and optimized in-space vehicle propulsion system operability. NASA defined a Lunar Lander mission/vehicle as the propulsion system to apply operability methodology and conceptualize an operable in-space propulsion system. The four design concepts that were developed were driven by operational considerations, and each iteration provided a more operable concept. The final design iteration is highly operable, and the supporting technologies are doable and would support an early year 2000 Lunar mission schedule. These operationally efficient designs revealed the necessary technologies to allow development of an operable Lunar lander concept.

Study task elements included acquiring operations databases from four current and past flight systems, initiating and defining a process to produce an in-space operations index, conceptualizing four operations-driven Lunar lander propulsion system designs, and recommending technologies which require development in order to bring these operational designs to fruition.

A database of operations experience from four current and past flight systems was assembled to provide a documented source of applicable in-space propulsion system operations experience. These systems, the Centaur, Saturn S IV-B, Shuttle OMS, and Lunar Module Ascent propulsion systems, were space environment operable that are on par with present conceptual designs for a Lunar Lander vehicle system. Separate databook volumes were produced for each in-space propulsion system. The database volumes, though expansive, are limited because of task resources and schedule constraints However, these volumes do represent a valuable source of pertinent data. The databook material sources included NASA centers (LeRC, KSC, MSFC, and JSC), vehicle contractors (General Dynamics and Rockwell Space Systems), archives and libraries (University of Alabama, Huntsville), published reports and personal interviews and files. Impediments to developing complete databooks included: 1) No single area where data is filed under the system category; 2) Data located in personal files, libraries, history files, repositories, on micro-fiche, or missing; 3) Apollo era data archived in regional storage facilities; and 4) regional storage facilities which require extensive travel to suspected data sources. A recommended subsequent task would be to add

additional materials to the databooks and complete the operations information analysis using a focused approach.

A methodology process was formulated to allow comparative analysis of in-space propulsion system operability. An operations indicator or index is needed to provide a quantifiable measurement to permit assessment of operability along with other system characteristics such as performance, reliability, and unit cost. The approach taken was to work towards a top-level, strategic index which would serve as a tool for operability evaluations at early stages of design. The index would not require operations experience on the part of the tool user, but would embody that experience within it. This index would provide evaluators and designers the means to assess relative operabilities of alternate concepts. First, the index would provide a measure of goodness of a propulsion system's operability. Ideally, the index would serve as an indicator of how close a propulsion subsystem's operability is to an optimal design for operability. With this insight and added flexibility, a designer would be able to improve the operability of a propulsion design.

The intention of creating an operations index is to provide operator experience to the designer/program manager, etc. to promote better system operability through this communication tool. A clear, common definition of the term operability is important if it is to have meaning and usefulness in designing better spacecraft. The study used the following as the definition of in-space operations: "In-space operations includes preparing and placing a propulsion system (that is already in space) into operation (but not including the operation itself) and keeping it ready for its next use."

The methodology described is a first draft for an In-Space Operations Index (ISOI) approach. It is intended to stimulate thought by those experienced with in-space operations. This In-Space Operations Index is also intended to be improved over the long term, through workshops, seminars, and in-space operations database additions. A similar approach was used to develop the Launch Operations Index.

Using NASA requirements for a Lunar Lander propulsion system, conceptual designs were devised which minimized operability concerns and issues. These propulsion systems designs included propellant tanks, propellant distribution and the rocket engines. Major operability enhancing features include a two-fluid (LOX/$LH_2$) system, integrated designs including RCS, differential throttling for thrust vector control, zero NPSH pumps (no tank pressurization), turbopumps interfaced directly to propellant tanks, and no hydraulics, pneumatics, helium, hypergolics, monopropellants, gimbal systems or flex lines. Several propellant tank arrangements were studied around the basic four-thrust-chamber/two-turbopump set modular engine arrangement. These propellant tank arrangements led to the development of four concept designs. Figure 1 presents a sketch of one of the propulsion system designs. Table 1 describes the operability features incorporated into these systems.

Design comparisons between the four Lunar Lander propulsion system designs and the Centaur and S IV-B systems were completed. The immaturity of the In-Space Operations Index precluded complete propulsion system comparison against in-space concerns, however, a Launch Operations Index comparison was made. As all systems must be earth launched, use of the LOI has initial validity. The LOI percentages were all in the low 80's for the Lunar Lander conceptual designs. This compares with LOI percentages in the mid 30's for the Centaur and S IV-B. Existing and previously designed propulsion systems were not designed with operability as the primary objective. The large difference in LOI values reflects this difference in design objectives.

A list of technologies was prepared identifying operational efficient technologies considering STPOES task results, space transfer propulsion system concept designs, the mission and related factors (i.e., OEPSS and upper stage programs). Operational technology lists were compared and reviewed for propulsion system applicability. Four technology development areas were identified; the Oxidizer-rich preburner, SLIC$_{TM}$ turbopump, Jet boost pump/SLIC$_{TM}$ turbopump module, and a test bed for the integrated propulsion module. Technology development plans were formulated for these technologies.

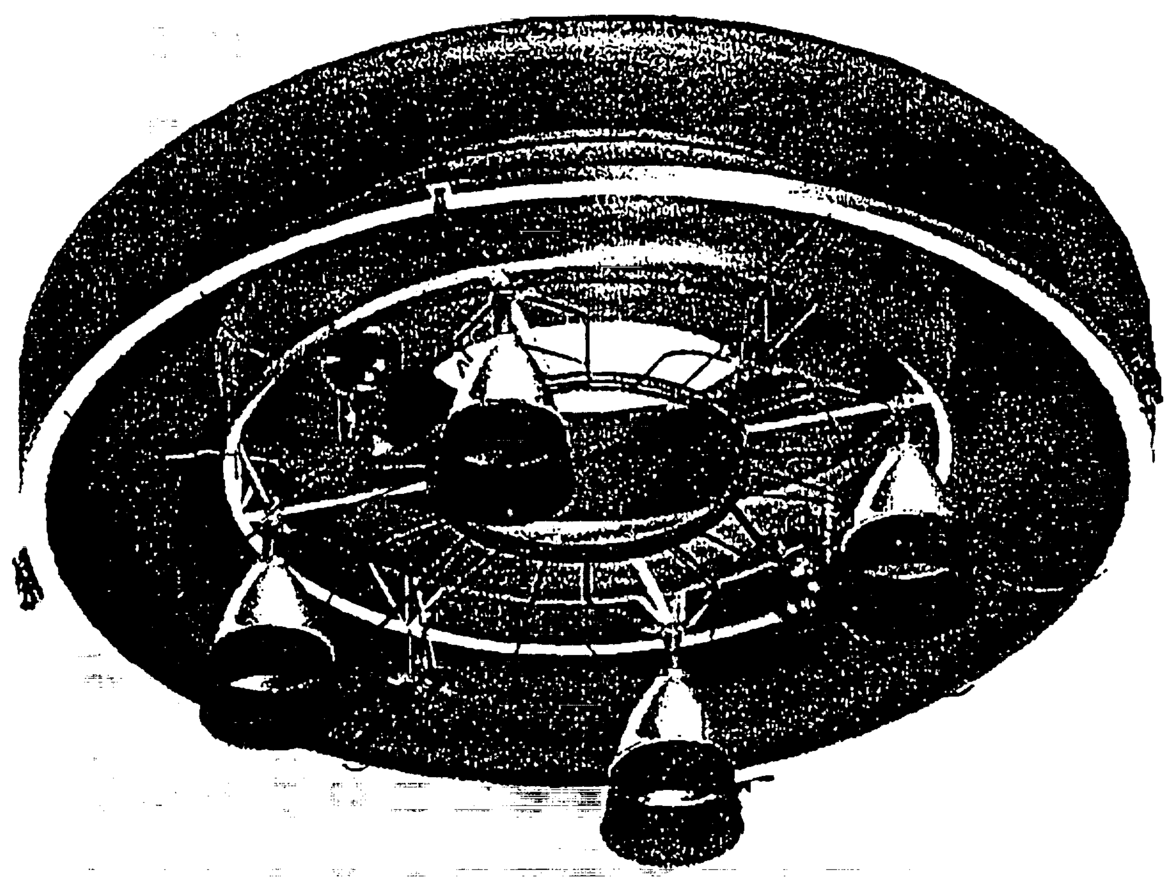

**Figure 1. Operationally Efficient Lunar Lander Conceptual Design**

# Table 1. Lunar Lander Design Features

- Open Propulsion Compartment
- Automated Checkout
- Non-intrusive propellant gaging system
- Two Fluid System -- LOX/$LH_2$
- $O_2/H_2$ RCS
- Laser Ignition (Engines & Ordnance)
- EMA Actuators
- Differential Throttling TVC (no Gimbal)
- Zero NPSH Pumps (no Tank Pressurization)
- Integrated Systems
- Accessible Components
- Turbopumps Interfaced Directly to Propellant Tanks (no preconditioning)
- No Hydraulics, Pneumatics, Helium, Hypergolics, Monopropellants APU's, Gimbal Systems, Flex Lines

# INTRODUCTION

Historically, propulsion system design has focused on performance and in-flight reliability attributes. Operationally efficient features were not a prime consideration during the design phase. Operational features were implemented during development and flight phases of programs only if they did not impact performance, cost to implement, and schedule. The results were systems which were costly and time consuming to operate. Modifications to improve operability are difficult to implement in systems where designs are fixed and flight requalification of new retrofit hardware is expensive and may require some risk.

In 1986 NASA KSC contracted Boeing to conduct a study of launch operations. The study showed the importance of incorporating operational features into the vehicle early in the design phase and identified the propulsion system (engine, propellant distribution systems, propellant tankage and supporting systems) as one of the major systems responsible for high operational costs. This result led KSC to initiate the OEPSS study contract to Rocketdyne to focus on operational issues of current and past propulsion systems.

The OEPSS program has generated ground operations data to provide information for designers, development engineers, program managers, etc., to assess future designs. The study identified: (a) major operational problems and their impact on operational requirements; (b) operations technology that will enhance operability and simplify launch site operations and support requirements; and (c) illustrative design approaches that achieve operability and operational efficiency in future propulsion systems. The results of the OEPSS study have been widely disseminated in briefings, workshops, symposiums , and Propulsion System Interface Working Group (PSWIG) meetings.

Recognizing that operations in space are even more difficult to perform than ground based operations, the Space Transfer Propulsion Operational Efficiency Study task was initiated to focus on space propulsion systems. A space propulsion system was defined as any system that is started in-space or in-flight (i.e., a second stage).The purpose of this effort was to identify operations issues and related concepts and technologies which would enhance the operability of space propulsion systems.

The scope of the effort includes gathering data for current and past cryogenic and storable propulsion systems; determining a methodology of comparing propulsion system concepts with respect to operability; identifying operable space propulsion concepts and their enabling technologies; and comparing these concepts to conventional approaches.

Resources limited the extent of the effort. However, sufficient accomplishments have been achieved to provide initial results and to indicate methodologies for expanding the work to assure that designs for future space propulsion systems include features which lower operational costs.

## DISCUSSION

Operations may be defined broadly as the activity or special systems (like ullage rockets or thermal conditioning, etc.) required to get a propulsion system ready to operate. These activities or resources may be manpower and/or materials and/or equipment and/or conditions and/or controls and/or maintenance etc. having to do with preparing a propulsion system for operation. Also, an important objective was to identify and document operations issues. The STPOES task's overall objective was to define technologies and design approaches for in-space cryogenically fueled ($LOX/LH_2$) propulsion systems which reduce ground, flight and in-space operations. For this study, the propulsion system includes the propellant tanks, auxiliary propulsion, feed system, and integrated engine systems. Five subtasks were implemented to meet the overall objective. The initial subtasks were: 1) Select four space propulsion systems as reference, and 2) Assemble a data base of operations experience on the selected propulsion systems. Tasks 1 and 2 did not prove to be efficient, i.e., attempts to collect operations data on these systems revealed the data was not as readily available as assumed. Instead, the more efficient process was to get experienced flight and ground operations personnel to define the major operations concerns. Information was gathered on ground-based operations and then in-space operations concerns. From these concerns, a data base was assembled that addressed areas from past program performance. By concentrating on operations concerns, the function drives the form and defines space propulsion systems conceptual designs which simplify operations. The change in approach with subtasks 1 & 2 required using "an experience base" to define concerns and a more operationally efficient propulsion system. It is recognized that this method did not adequately anchor experience data to support new functionally driven form as well as designers would like.

The remaining tasks were: 3) Formulate general methodologies for comparing space propulsion operability; 4) Define space propulsion system conceptual designs which simplify ground, mission and space-based operations; and 5) Identify technologies and formulate development plans for critical operational efficiency areas. The STPOES subtasks provided the direction and initial steps in a process of assuring that future space propulsion system designs will include features which lower operational costs.

## 2.0 PROPULSION SYSTEM BASELINES

Four systems were identified to be included in the propulsion system operational efficiency study baseline reference; the Centaur (RL 10), the Saturn Stage S IV-B (J-2), The LM Ascent, and the Shuttle OMS. The Centaur and Saturn Stage S IV-B use cryogenic propulsion systems (LOX/$LH_2$), and the LM Ascent and the Shuttle OMS use storable propellants (NTO/MMH). A short description of each baseline propulsion system is presented below. Additional propulsion system information is presented in the individual propulsion system data books, together with operational data for each propulsion system.

### 2.1 CENTAUR PROPULSION SYSTEM

The Centaur upper stage vehicle is 10 feet in diameter and 30 feet long. The Centaur employs high energy liquid hydrogen ($LH_2$) and liquid oxygen ($LO_2$) propellants separated by a double-wall, vacuum insulated intermediate common bulkhead. The propellant tanks are constructed of thin-wall, fully monocoque, pressure stabilized, corrosion-resistant stainless steel. Tank stabilization and integrity is achieved by internal helium pressurization. Tank stabilization is maintained at all times by either internal pressurization or the application of mechanical stretch. Figure 2.1-1 presents a sketch of the Centaur showing its major components.

The Centaur propulsion system uses two RL10A-3-3A engines manufactured by Pratt & Whitney. These engines are regeneratively cooled and turbo-pump fed, with a rated thrust of 16,500 pounds each. Attitude control during coast phases of flight is achieved with a Hydrazine Reaction Control System. The Reaction Control System (RCS) is mounted on the Centaur $LO_2$ tank aft bulkhead.

A Liquid Helium Chilldown System provides prelaunch thermal conditioning for the main engines. The main engine turbopumps are cooled by cold helium gas obtained by vaporization of liquid helium. Prechilling of the turbopumps allows inflight chilldown time to be minimized for first burn. This system is activated 45 minutes prior to the opening of the first launch window. Liquid helium flow is terminated at T-8 seconds. This system is not active during flight.

Two identical and separate hydraulic power supply systems gimbal the Centaur main engines. Each power package contains two pumps that supply pressure to the actuators. One pump, coupled to the engine turbine drive, operates while the engines are firing. During the coast phase, another electrically powered pump is computer-controlled to circulate hydraulic fluid through the system and to null engines before engine start.

### 2.2 SATURN STAGE S IV-B PROPULSION

The Saturn S-IV B stage and propulsion system functions were to inject the Command, Service and Lunar Module (LM) assembly (Apollo spacecraft) into low earth orbit, coast in earth orbit for a period of approximately four and one-half hours, or three orbits of earth, and then restart the main (J-2) engine and put the S-IV B and the Apollo spacecraft into translunar trajectory. While in earth orbit, the S-IV B relied on its auxiliary propulsion system to ensure proper attitude control and propellant tank

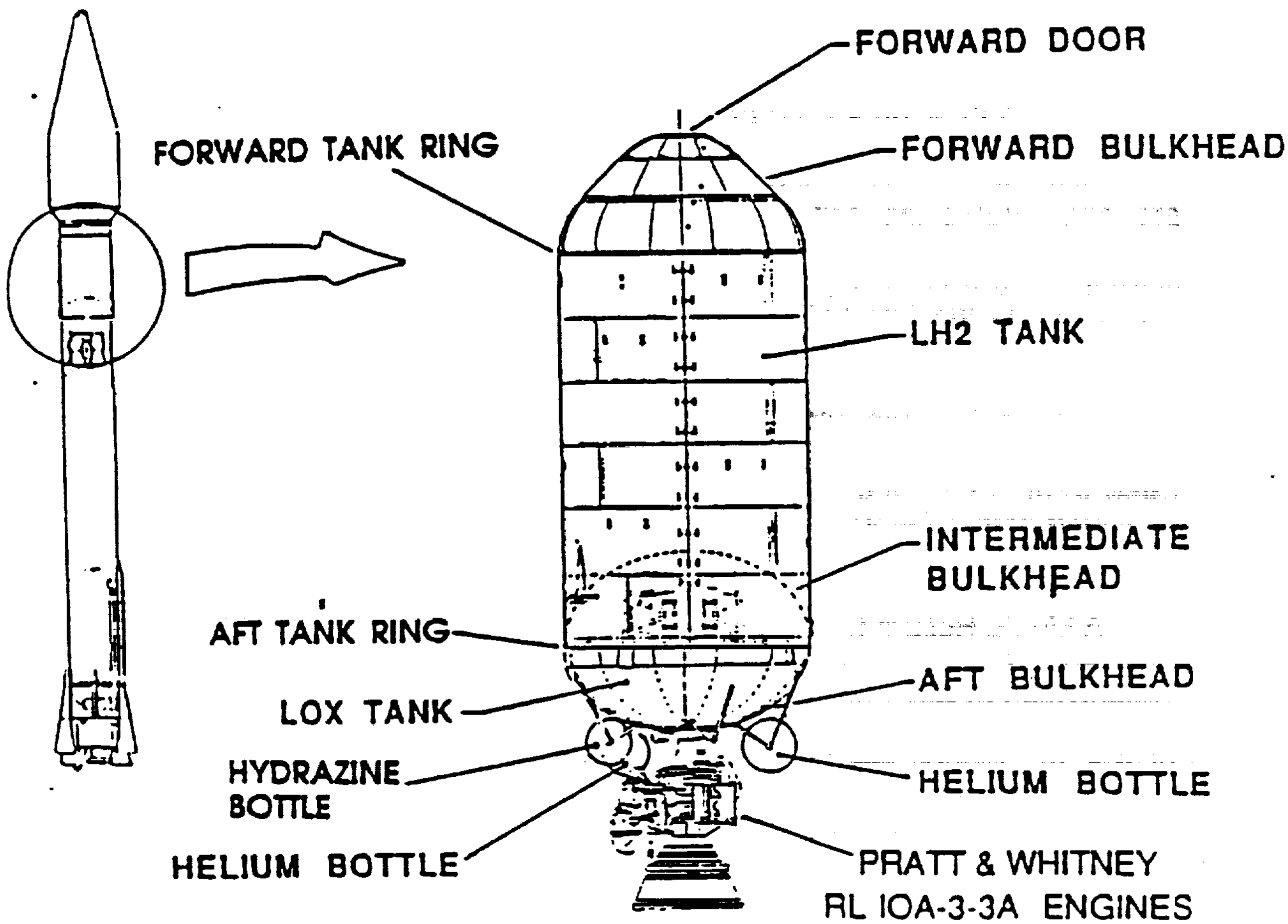


**Figure 2.1-1. Centaur Major Components**

orientation for engine restart. When the secured burn was completed, the transposition maneuver was carried out, resulting in nose-to-nose rendezvous of the LM and CSM. After completion of the transposition maneuver, the S-IV B was separated from the LM-CSM assembly by retro rockets.

The S-IV B propulsion system consisted of a fuel and oxidizer propellant tank assembly, main rocket engine system (J-2), flight control systems, auxiliary propulsion system and ullage control solid rockets. Figure 2.2-1 presents a sketch of the S IV-B showing its major components.

The propellant tank assembly includes a cylindrical tank section, common bulkhead, aft dome and forward dome. The propellant feed system consists of separate oxidizer and fuel turbopumps, main fuel valve, main oxidizer valve, propellant utilization valve, oxidizer and fuel flowmeters, fuel and oxidizer bleed valves, and interconnecting lines. There were also propellant mass measuring systems, a fuel thermal conditioning feed system, a non-propulsive vent system, tank pressurization systems and a pneumatic purge and valve control system.

The main propulsion system consists of the J-2 engine, including its thrust chamber and gimbal system, propellant feed system, gas generator and exhaust system, control system, start tank assembly, a pneumatic control package and system and flight instrumentation system. The J-2 engine is a 225,000 pound thrust, high performance, upper stage, propulsion system, utilizing liquid hydrogen and liquid oxygen propellants, and incorporating a built-in capability for restart in flight. An oxidizer turbopump is mounted on the thrust chamber diametrically opposite the fuel turbopump. It is a single-stage centrifugal pump with direct turbine drive. The fuel turbopump, also mounted on the thrust chamber, is a turbine-driven, axial flow pumping unit consisting of an inducer, a seven-stage rotor, and a stator assembly. A gas generator produces hot gases to drive the oxidizer and fuel turbines and consists of a combustor containing two spark plugs, a control valve containing oxidizer and fuel ports, and an injector assembly.

The flight control system provides stage thrust vector steering and attitude control. Steering is achieved by gimbaling the J-2 engine during powered flight. Hydraulic actuator assemblies provide J-2 engine deflection rates proportional to steering signal corrections. Stage roll attitude during powered flight is controlled by firing the auxiliary propulsion system (APS) attitude control engines.

The pneumatic system consists of a high pressure helium controlled gas storage tank, a regulator to reduce the pressure to a usable level, and electrical solenoid control valves to direct the central gas to the various pneumatically controlled valves and purge systems. The hydraulic system performs engine positioning upon command.. Major components are a J-2 engine-driven hydraulic pump, an electrically driven auxiliary hydraulic pump, two hydraulic accumulator assemblies, and an accumulator-reservoir assembly.

The Auxiliary Propulsion System (APS) includes modules that provide three-axis attitude control. Two APS modules are mounted 180 degrees apart on the aft skirt assembly. Each APS module contains three, 150-pound thrust engines. Each APS module contains an individual oxidizer system, fuel system, and pressurization system.

Propellants are NTO and MMH. The modules are self-contained and easily detached for separate checkout and environmental testing.

## SATURN V/S-IVB EXPLODED VIEW

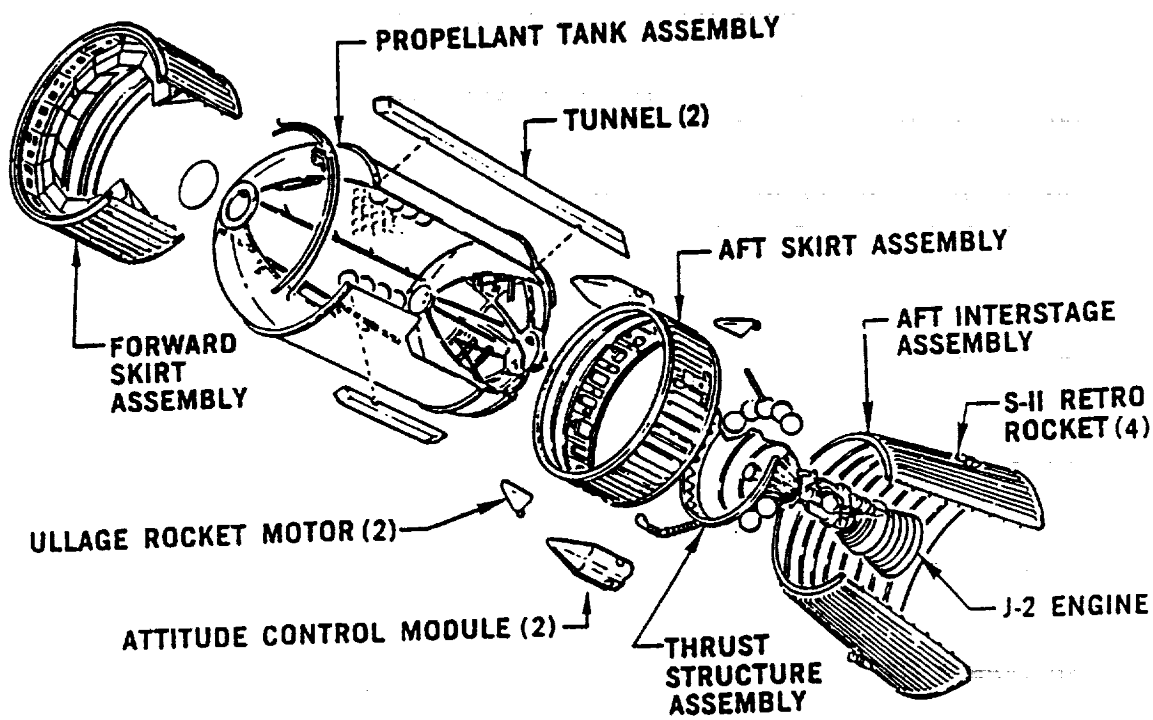


Figure 2.2-1.

Propellants are NTO and MMH. The modules are self-contained and easily detached for separate checkout and environmental testing.

Two solid-propellant ullage rocket motors are mounted 120 degrees apart on the aft skirt assembly. The two solid-propellant Thiokol TX-280 rocket motors, each rated at 3,390 pounds of thrust, are ignited following SII and SIVB separation for ullage control. This thrust produces additional positive stage acceleration during separation, and positions LOX and $LH_2$ propellants toward the aft end of their tanks to cover outlets to allow conditioning and engine start.

## 2.3 LUNAR MODULE PROPULSION SYSTEM

The Ascent Propulsion System (APS) provides the velocity ($\Delta V$) necessary to take the ascent stage from the lunar surface into lunar orbit. The APS consists of a pressure-fed, liquid-bipropellant, ablatively-cooled rocket engine and its propellant feed, storage, and pressurization systems. The ascent engine consists of an ablative-lined thrust chamber, and injector assembly, two propellant ducts and trim orifices, and a bipropellant valve assembly. The ascent engine is a constant-thrust, restartable engine, which develops a nominal 3500 pounds of thrust in a vacuum. The engine is rigidly mounted to the ascent stage and oriented so that the thrust vector passes approximately through the center of gravity of the stage. Figure 2.3-1 presents a sketch of the Lunar Module showing its major components.

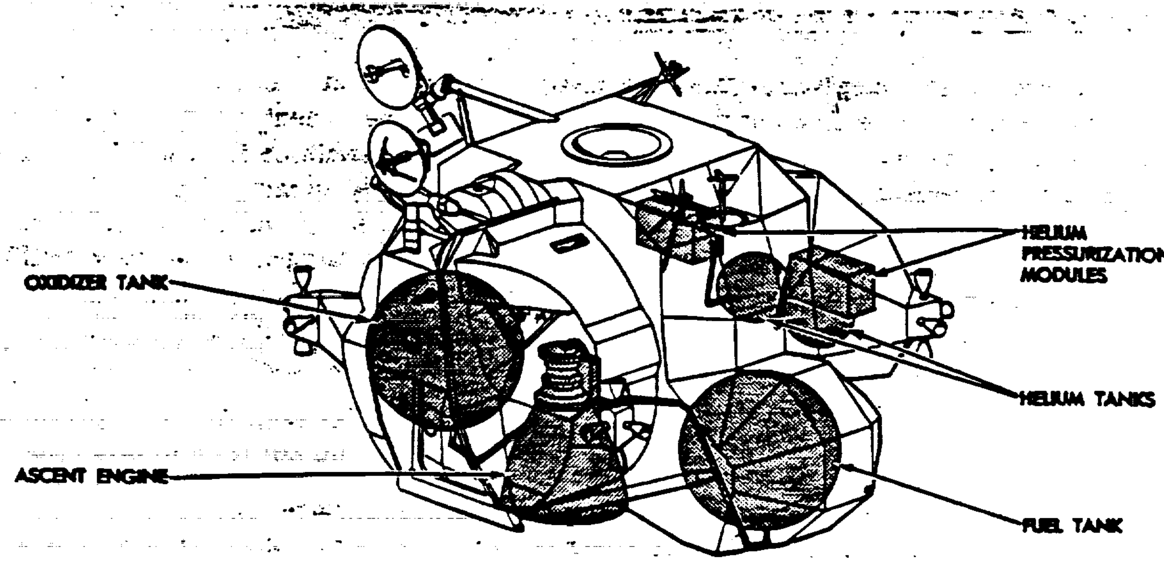


Figure 2.3-1. Ascent Propulsion System--Major Equipment Location

The ascent stage also contains the LM reaction control system (RCS). The RCS serves both the LM descent and ascent stages to provide attitude control. To ensure reliable system performance, the design of the LM RCS was based on system and component redundancy -- similar to the Mercury and Gemini spacecraft and the Apollo CSM. Two independent and operationally identical LM RCS systems, each capable of providing attitude control and positive and negative longitudinal translation, were provided. The propellant tanks were of the positive-expulsion configuration. Each propellant was contained inside a Teflon bladder that was in turn placed inside a titanium shell.

The APS contains one oxidizer tank and one fuel tank. The outflow from each tank divides into two paths. The main path leads through a trim orifice and a filter to the engine shutoff valves; the other path leads to normally closed solenoid valves interconnecting the APS and the reaction control system (RCS) propellant systems. Opening these valves permits the use of APS propellants by the RCS.

The gaseous helium pressurant for the APS is stored in two spherical pressure vessels at a pressure of 3025 psia. A relief valve and its isolation burst disk are located in the helium pressurization line to each propellant tank to prevent catastrophic tank overpressurization. The APS propellant tanks do not have a quantity-gaging system, but do contain low-level sensors that are used to provide an approximate 10-second warning of propellant depletion.

## 2.4 SHUTTLE ORBITAL MANEUVERING SYSTEM (OMS)

The OMS provides the thrust for orbit insertion, orbit circularization, orbit transfer, rendezvous, deorbit, abort to orbit, and abort once around and can provide up to 1,000 pounds of propellant to the aft reaction control system. The OMS is housed in two independent pods located on each side of the orbiter's aft fuselage. The pods also house the aft RCS and are referred to as the OMS/RCS pods. Each pod contains one OMS engine and the hardware needed to pressurize, store and distribute the propellants to perform the velocity maneuvers. Figure 2.4-1 presents a sketch of the Shuttle OMS showing its major components.

The OMS in each pod consists of a high-pressure gaseous helium storage tank, helium isolation valves, dual pressure regulation systems, vapor isolation valves for on the oxidizer regulated tank, and a propellant distribution system consisting of tank isolation valves, crossfeed valves, and an OMS engine. The propellant storage and distribution system has one fuel tank and one oxidizer tank in each pod. In addition, the distribution system includes a mass measuring system and an elaborate propellant acquisition system to capture liquid propellant at zero G and prevent any gas from entering the engine feed system. Each OMS engine also has a gaseous nitrogen storage tank, a gaseous nitrogen pressure isolation valve, a gaseous nitrogen accumulator, bipropellant solenoid control valves, and actuators that control bipropellant ball valves, and purge valves.

Each OMS engine produces 6,000 pounds of thrust. An OMS engine can be reused for 100 missions and is capable of 1,000 starts and 15 hours of cumulative firing. Each engine has two electromechanical gimbal actuators, to control thrust pitch and

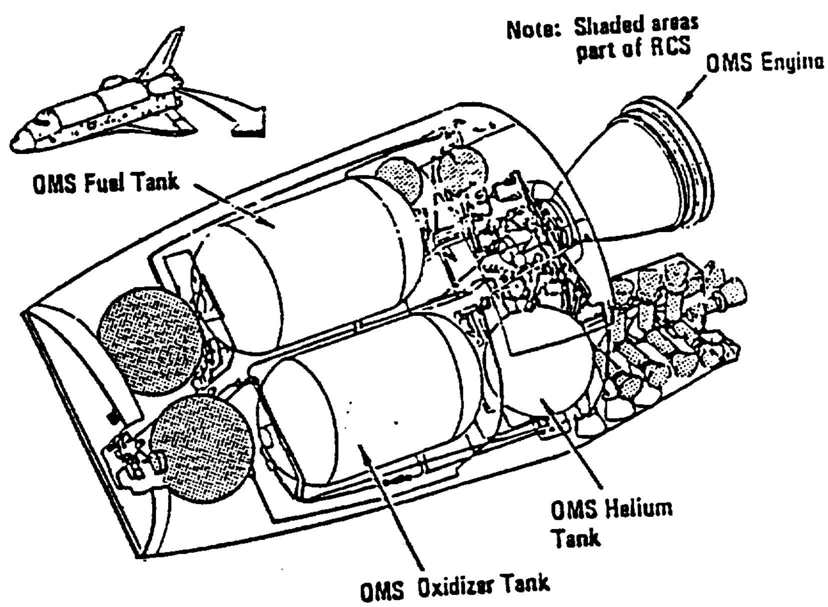


Figure 2.4-1. STS Orbiter/OMS

yaw. Each OMS engine receives pressure-fed propellants at its bipropellant valve assembly. The bipropellant ball valve assembly is actuated pneumatically from its gaseous nitrogen system. The engine is purged with nitrogen after the thrusting period.

The engine thrust chamber assembly is regeneratively cooled. A platelet injector is used to mix propellants. Propellants are NTO and MMH which are hypergolic.

Thermal control is achieved by insulation on the interior surface of the pods that enclose the OMS hardware and the use of strip heaters. The heaters prevent propellant from freezing in the tanks and lines.

The OMS engine Fault Detection and Identification (FDI) detects and identifies off nominal performance of the OMS engine, such as off-failures during OMS thrusting periods, on-failures after or before a thrusting period, and high or low engine chamber pressures. The OMS gimbal actuator FDI detects and identifies off-nominal performance of the pitch and yaw gimbal actuators on the engines.

## 3.0 OPERATIONS DATABASE

Four different vehicle propulsion systems were selected for the operations database. Two were cryogenic propellant systems, and two were earth storable propellant systems. All were space environment operable, and, outside of satellites and their relatively small "kick stages," these systems offered the best opportunity for studying space propulsion systems that were on par with present conceptual Lunar Return vehicle systems.

Very early in the study attempts to collect operations data on these systems at Kennedy Space Center resulted in the realization that, except for the STS orbiter OMS, desired data was not as readily available as assumed. Figure 3.0-1 summarizes data availability for each of the propulsion systems. Also it became apparent that the majority of the data resided with the responsible design centers as follows; LM Ascent with Johnson Space Center, Centaur/RL10 with Lewis Research Center, S-IV B with the University of Alabama, Huntsville and STS/OMS with Johnson Space Center. Visits were made to each of the centers, and the data search was initiated. Further impediments were encountered, which made the search more difficult and time consuming, including: 1) no single area where data is filed under the system category; 2) data was in personal files, libraries, history files, repositories, on micro-fiche, or missing; 3) Apollo era data was archived in regional storage facilities; and 4) regional storage facilities would require extensive travel to suspected data sources.

Figure 3.0-2 pictorially summarizes where the search thus far has indicated that the data resides. The focal point for the data collected was the NASA center listed. The scope of the data presented for each propulsion system is an indication of their relative availability.

The Space Systems Division of Rockwell International assembled a wealth of pertinent systems data, and Johnson Space Center personnel provided documents and interviews to further enhance the OMS information. The Centaur/RL 10 system is the next most complete databook. General Dynamics Space Systems and NASA LeRC provided summary material, and Rocketdyne supplemented representative Centaur/RL 10 system data from other sources. It was agreed by both Rocketdyne and NASA that due to resource constraints the data collection and presentation should be concentrated on the STS/OMS and Centaur/RL 10 areas. However, data that was available and collected on the Apollo S IV-B and LM Ascent systems has been summarized in separate databooks.

The format of the systems information databooks is shown in Table 3.0-1.

# Baseline Data Availability Summary

- STS Orbiter / OMS
  - Readily available
- Centaur / RL-10
  - Exists, competition sensitive
- Saturn SIVB / J-2 and Apollo LEM / Ascent
  - Partially available
    - "System" - archived
    - Engine - readily available

Figure 3.0-1. Chart A

# Propulsion System Baselines Data Sources

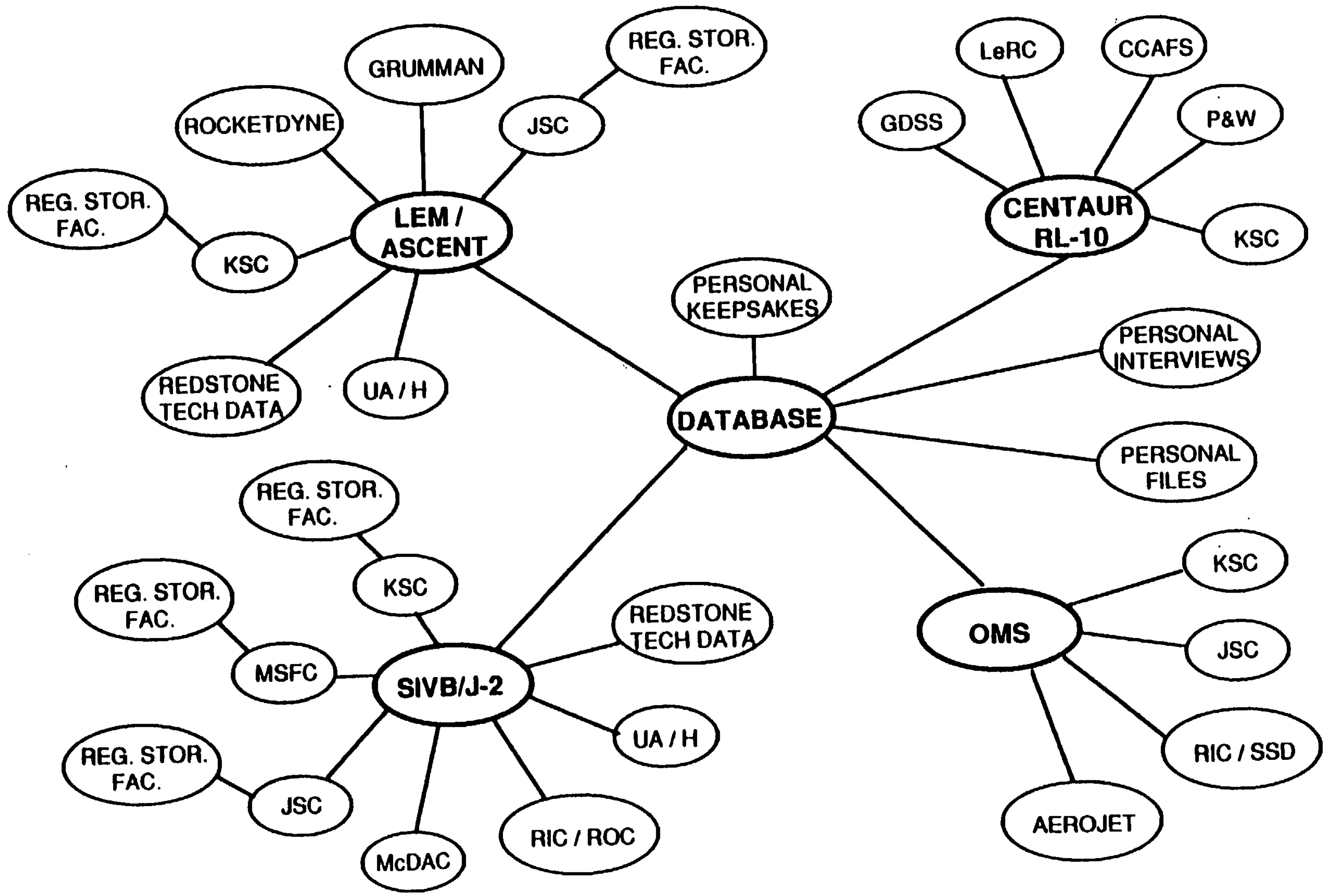


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Figure 3.0-2. Chart B

**Table 3.0-1, STPOES Databook Format**

| |
|---|
| Introduction |
| System Evolution |
| System Description |
| System Processing |
| Ground Support Equipment |
| Major Anomalies |
| Flight Operation Information |
| Operational Issues |
| Data Source Listing |

The databooks exist as separate volumes for each of the four systems. In the case of the STS/OMS, Centaur/RL10 and S IV B, multiple books were required to make them physically manageable.

The databases are limited because of the available resources but do present an initial volume of pertinent data. The study has also indicated the likely sources of further data from government, industry, and academic libraries. The institutions which generated the data would be the most effective in retrieving it. Further database materials should be generated in the context of operational concerns and issues. Persons who have experience with issues for a specific propulsion system will be able to describe the issue, relate and have access to documentation which can substantiate the concern, and evolve the concern to a broader issue more applicable to other propulsion systems than the one on which they encountered the problem.

An example of using database documentation for determining an operability issue is the "Liquid/Vapor Management" concern category. Liquid/Vapor Management Issues include propellant acquisition, propellant gaging and zero G venting. In the OMS databook, in the OMS Operational Issues Section, Propellant Acquisition is listed and documented as an issue/consideration. This issue becomes part of the documented operability concerns list, and design features are formulated for mitigating the issue. Other issues/concerns documented in the databooks can be derived with more effort.

# 4.0 PROPULSION OPERABILITY METHODOLOGY

An operations indicator or index is needed to provide a quantifiable measurement to permit assessment of operability along with other system characteristics such as performance, reliability, and unit cost. Operability accountability would be applied in the same manner as performance, reliability and cost assessments are used. The operations assessment, concurrently used in the design process, would identify operations issues for mitigation and/or elimination. The designer, engineering review boards, proposal review boards and others could use an operations index tool to enable comparisons and enhance operability. This task developed a methodology development process, defined in-space operability, and formulated a road map for the construction of an In-Space Operations Index.

## 4.1 METHODOLOGY DEVELOPMENT

Operability methodology development ultimately would produce an Overall Operations Index. The result would be a technique for rating the operability of an entire vehicle and mission by evaluating and combining the operabilities of the elements (systems) composing the vehicle. The scope of an overall operability index is exceedingly broad (reference Figure 4.1-1 for a lunar mission example) and beyond the scope of this task. Accordingly, this effort focused on the launch and in-space aspects of propulsion system operations. Mission control and management elements were considered as they relate to the propulsion system.

Although an operations calculation of cost for a vehicle is theoretically possible, it is difficult to obtain. Operations overall costs should be obtainable on existing systems, however, supportive data is not readily available. Operations cost data for a new design is less firm and is generally an estimate with large uncertainty. Further, an operations cost approach is tedious because of the many elements involved, and results would be subject to challenge as many subjective assumptions must be made.

An Operations Index can guide the overall or global direction of a new transportation system by providing an operations focus in the conceptualization process. This approach considers operability concurrently at the beginning of the design process, whereas in many systems operability is considered later in the process. Once the design is complete, operational enhancements are difficult to add into the system.

The approach taken was to work towards a top level, strategic index which would serve as a tool for operability evaluations mainly in the early stages of design. The index would not require operations experience on the part of the user of the tool; but would embody that experience within it. The index provides designers with a means of evaluating the operability of their concepts. Evaluators and designers could use the index to assess relative operabilities of alternate concepts. Future refinements of the index would sharpen its resolution.

The methodology described contains a first draft of an In-Space Operations Index approach. It is intended to stimulate thought by those experienced with operations and to be improved over the long term, through workshops, seminars, and in-space

# Lunar Program Elements

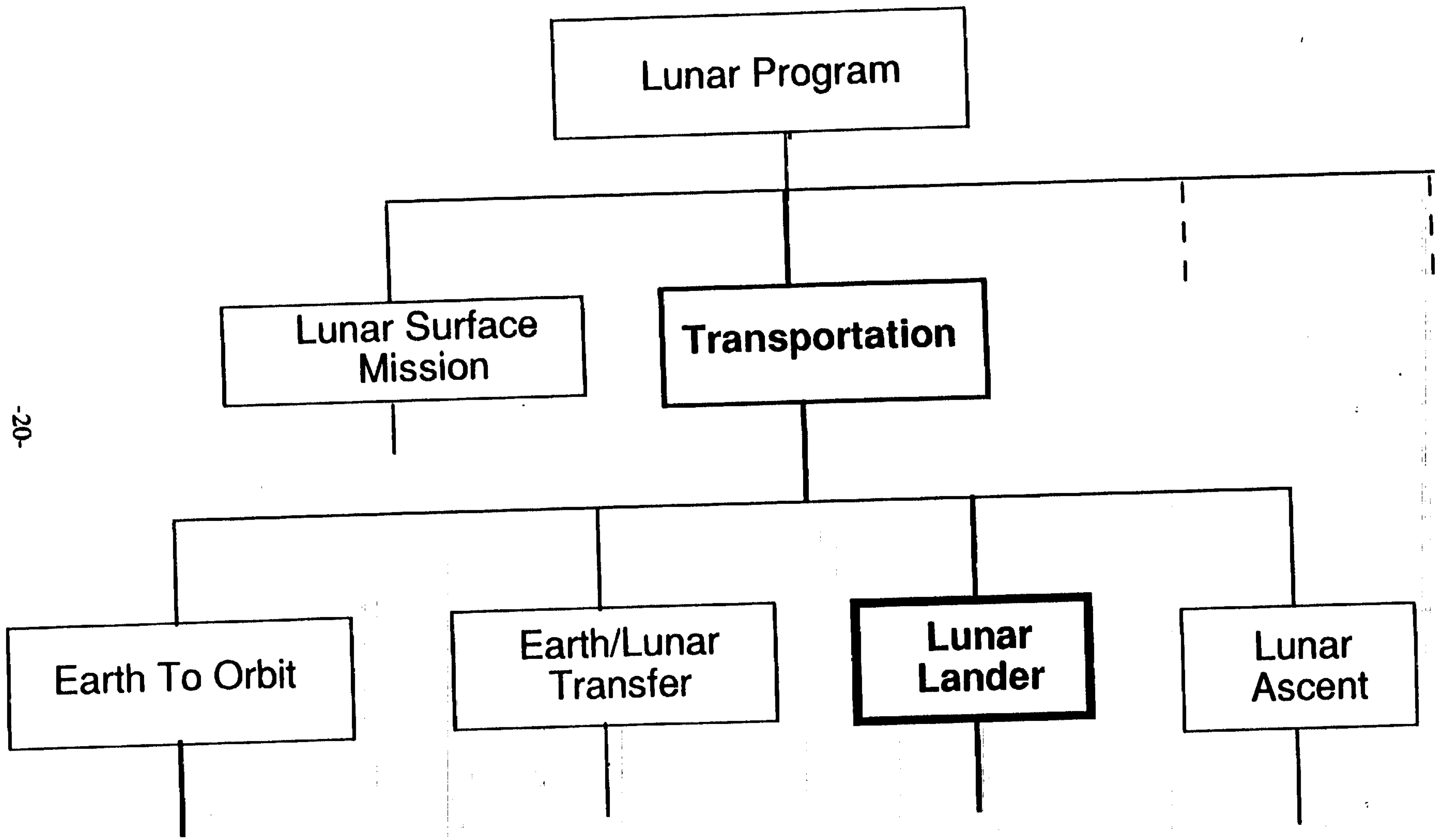



Figure 4.1-1a

# Lunar Program Elements (continued)

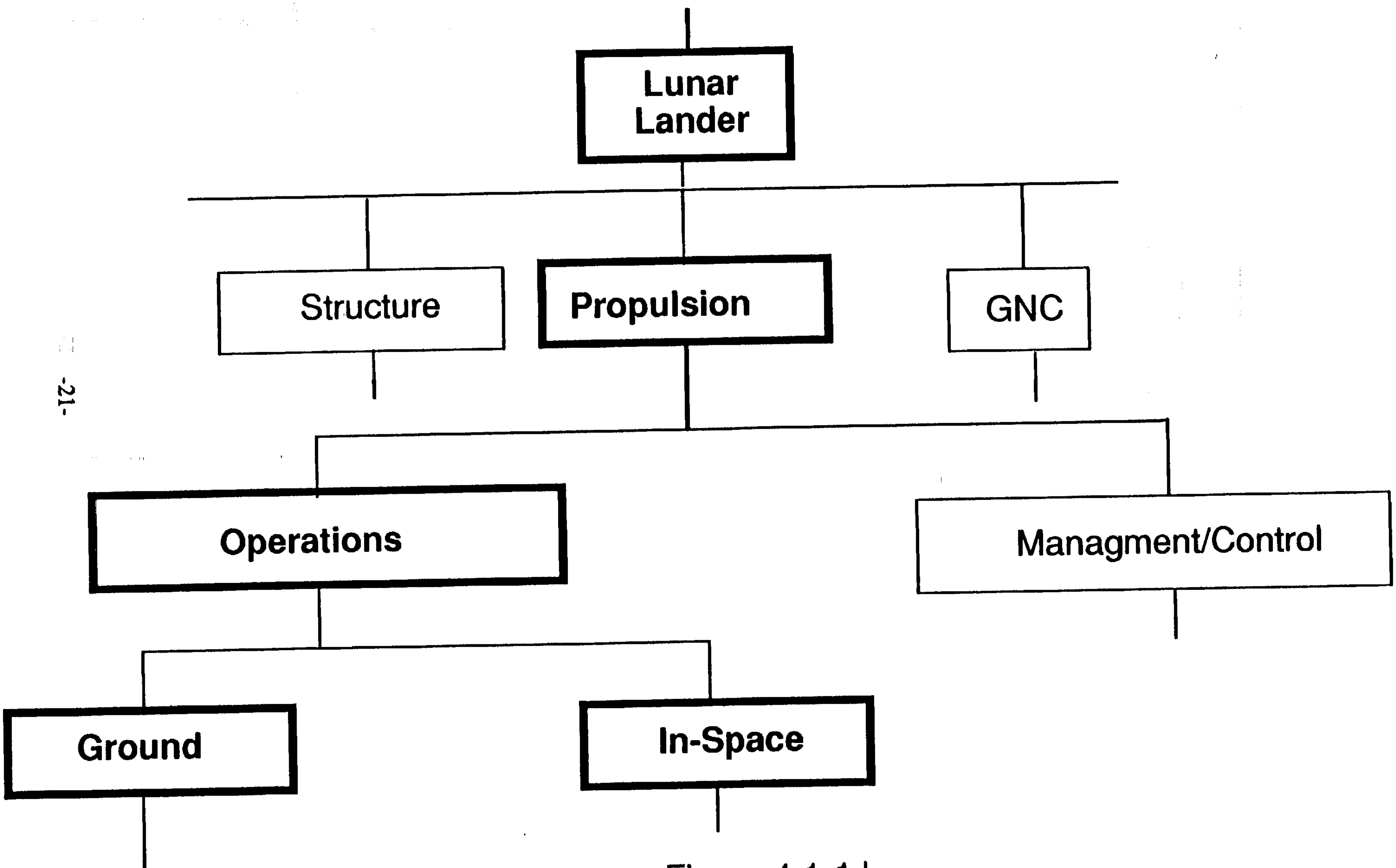



Figure 4.1-1 b

# Lunar Program Elements (continued)

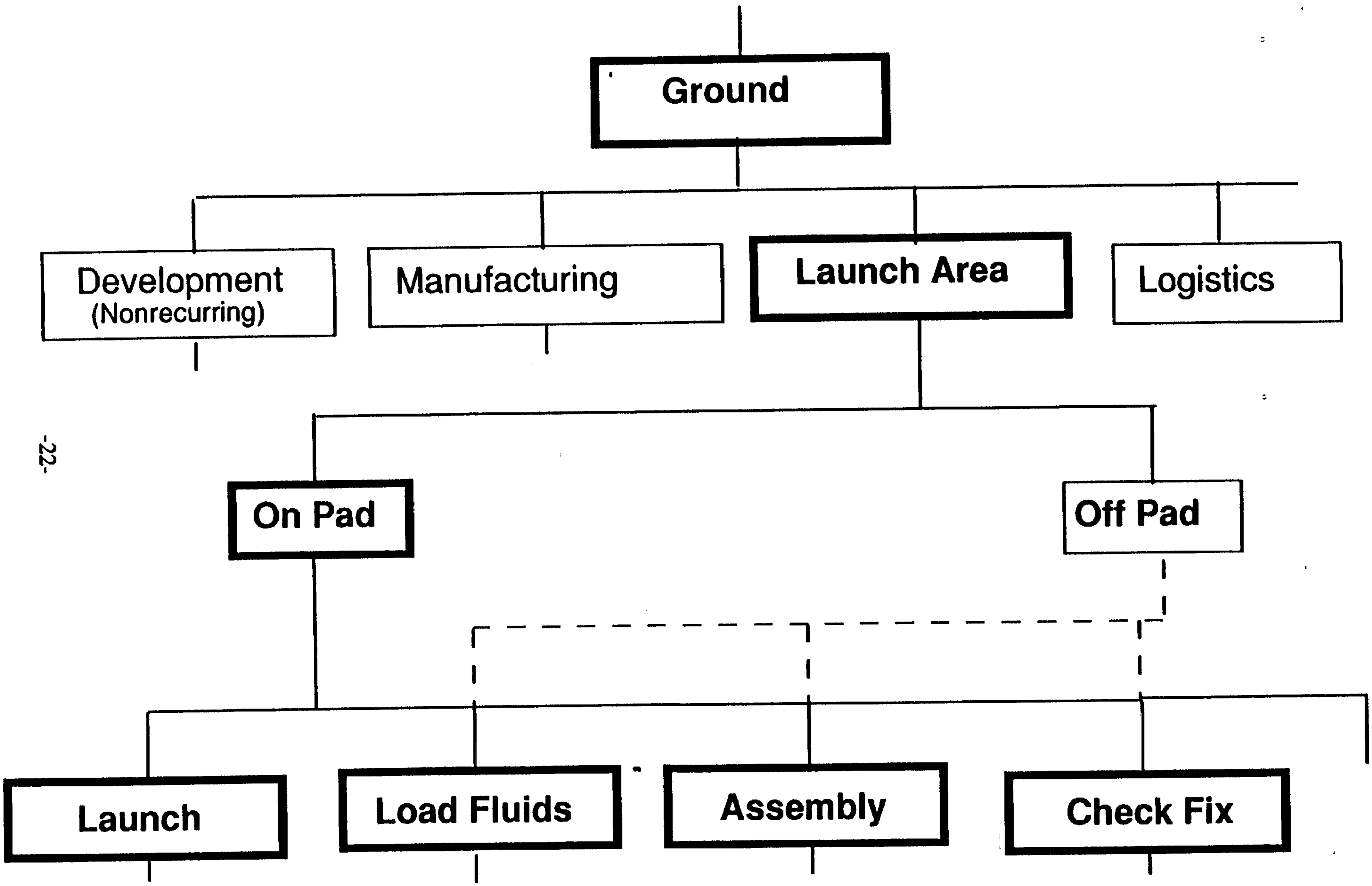



Figure 4.1-1c

# Lunar Program Elements (continued)

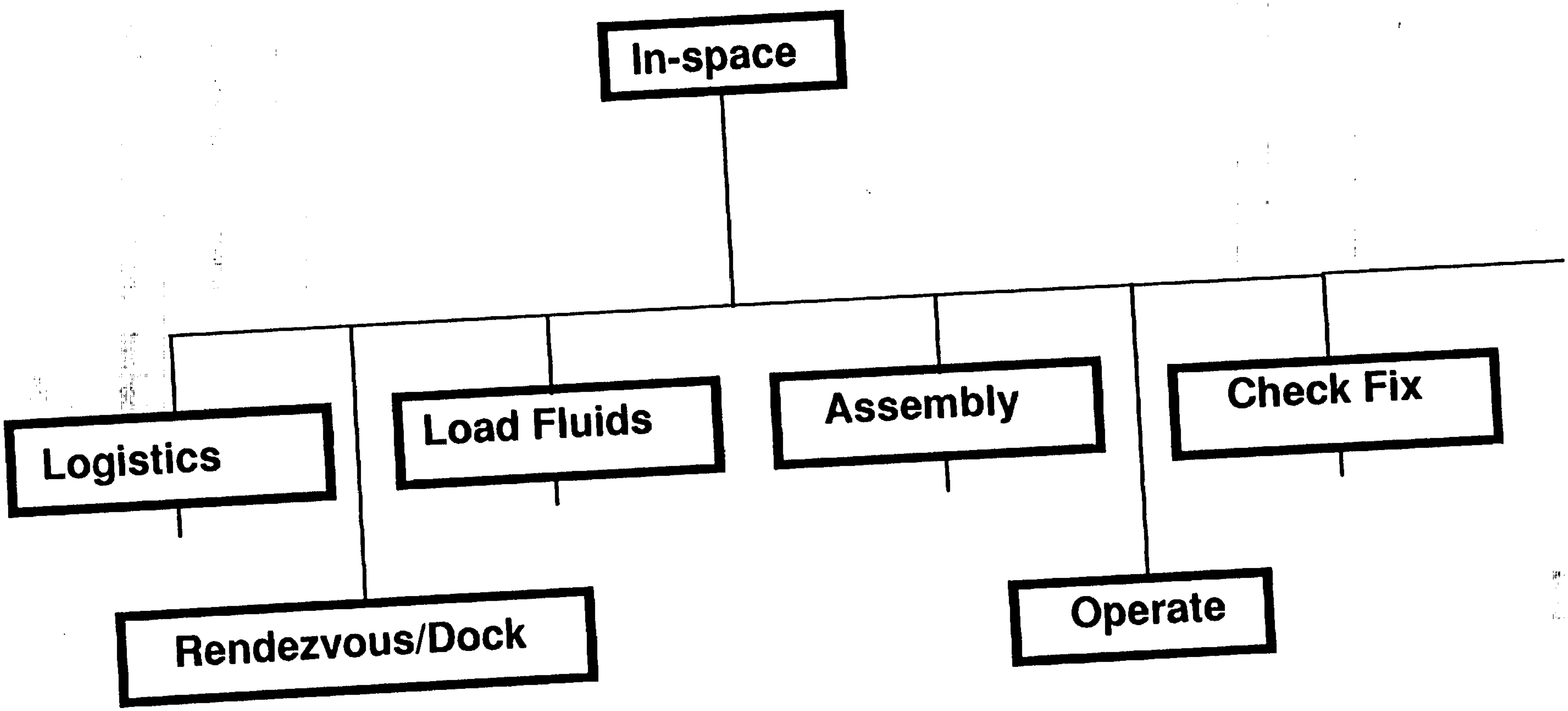



Figure 4.1-1d

operations database refinements and evaluations. A similar approach was used to develop the Launch Operations Index.

## 4.2 IN-SPACE OPERATIONS INDEX

The STPOES task plan was to develop a database, evaluate this raw data and evolve a list of in-space operations concerns and issues. An In-Space Operation Index would then be developed measuring how well these concerns and issues were addressed. This approach was not successful because of time, manpower and schedule constraints. Therefore, the task to develop an In-Space Operations Index (ISOI) for a propulsion system evolved from earlier effort in the OEPSS study where the task of developing a Launch Operations Index (LOI) was initiated. The LOI approach has been under review for several months and favorably received. Thus, the ISOI task approach follows the same general approach with the understanding that improvements were to be incorporated. The following documents the progress to date towards completing the ISOI.

1) Defining in-space operability.

2) Delineating in-space propulsion operations boundaries between the launch operations that precede in-space operations and the actual operation that succeeds it. By definition, in-space operations includes only the preparation leading to and not the actual functional operation itself.

3) Developing a methodology to construct an index.

4) Constructing the index.

The first two tasks have been completed, and progress made on the fourth is described. The third task still needs work and agreement by the community to allow reaching a consensus (through workshops). Note the initial ISOI is meant to be the starting position for an in-space operations index. Refinements to the ISOI would follow the same approach used in developing the LOA; i.e., conducting workshops and seminars with industry, NASA and the Air Force to supplement database information (Section 3.0), define concerns, and evolve design features.

### 4.2.1 DEFINING IN-SPACE OPERABILITY

Overall, three main uses were envisioned for the in-space operations index. First, the index would provide a measure of goodness of a propulsion system's operability. Ideally, the operations index would serve as a indicator of how close a propulsion subsystem's operability is to an optimal design for operability. With this insight, a designer would be able to improve the operability of a propulsion design.

The intention of creating an operations index is to measure an attribute of a spacecraft that has been mostly ignored in the past during design -- how operable is it? A clear, common definition of the term operability is important if it is to have meaning and usefulness in designing better spacecraft. The ISOI study used the following as the definition of in-space operations. "In-space operations includes preparing and placing a propulsion system (that is already in space) into operation (but not including the intended functional operation itself) and keeping it ready for its next use".

Second, a minimum standard of operability performance can be defined. By correlating the ISOI score with specific propulsion subsystems and their historical operational performance, a minimum standard might be developed that would separate acceptable from unacceptable hardware designs and provide a sense of direction as well as a rough measure of "goodness." Then, in the future, spacecraft specifications for propulsion would include the requirement that their level of operability be above some minimum standard or targeted at some goal.

Third, operability could become an evaluation criteria in tradeoff studies to assist in selecting the best overall design when there is more than one choice. Ideally, operability could be a criteria with the same level of importance as the traditional propulsion system objectives, see Figure 4.2-1. However, this application is only feasible if the definition of operability criteria does not overlap with any of the other major criteria such as hardware cost, reliability, performance, safety, etc. In this study, operations cost was included in the definition of operability.

### 4.2.2 DELINEATING IN-SPACE PROPULSION OPERATIONS BOUNDARIES

The distinction between launch and in-space operations is one of definition. In-space operations were defined as starting after insertion into a safe earth orbit. Thus, the propulsion subsystems of concern in space are those that operate after launch when the spacecraft is in space. For simplification purposes, the starting point for any in-space operation is a dormant state. The approach to developing an In-Space Operability Index was to divide in-space operations into five classes of operations that represent different purposes and activities. The classifications include test and checkout, pre-operations, assembly, service, and maintenance, as shown in Figure 4.2-2. Flow charts for test and checkout and pre-operations are presented in Figures 4.2-3 and 4.2-4. All five classes of in-space operations are presumed to start their sequence of activities from a common dormant state. Furthermore, four of these classes end with a dormant state. Pre-operations, on the other hand, ends after the start of the propulsion subsystem. At the end of the propulsion operation, if the mission requires repetitive cycles, then the subsystem also returns to a dormant state. A flow chart of the activity sequence for In-Space Operations for a propulsion system is presented in Figure 4.2-5.

The advantage of breaking in-space operations into five separate classes is the possibility of measuring the operability of each. Then, each separate index provides the details of operability within each class of operation. This information gives the designer and operators a better indication on how well a design will do on each particular operation.

For a top level index, which includes consideration of a propulsion design's overall operability, a composite index would be developed from its separate components. This composite operability index would be derived from each of the five in-space operation classes and combining them into the composite index.

# Space-Based Propulsion System Operability

Desire to develop a quantifiable independent figure of merit for evaluating space-based propulsion system operability

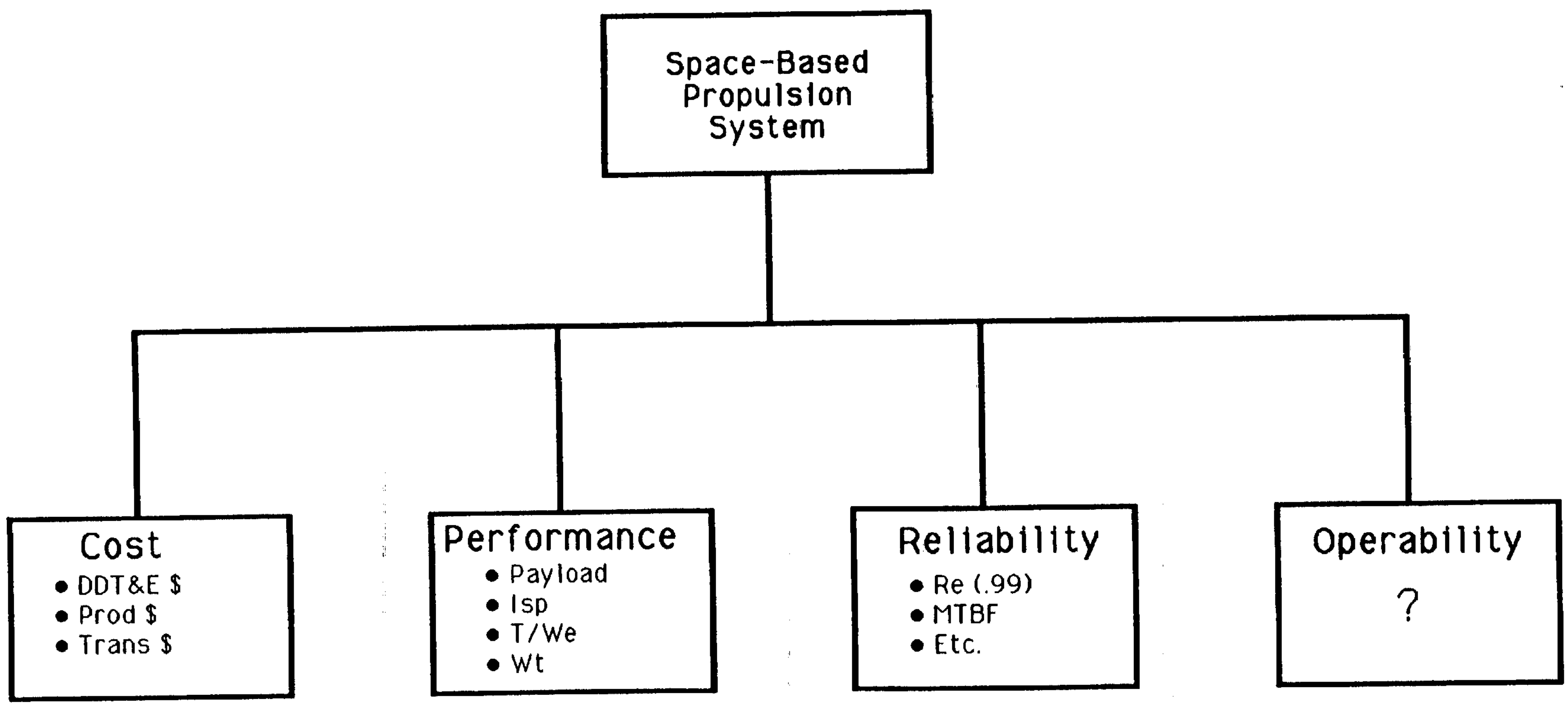



Figure 4.2-1

# Space Based Operations Major Categories

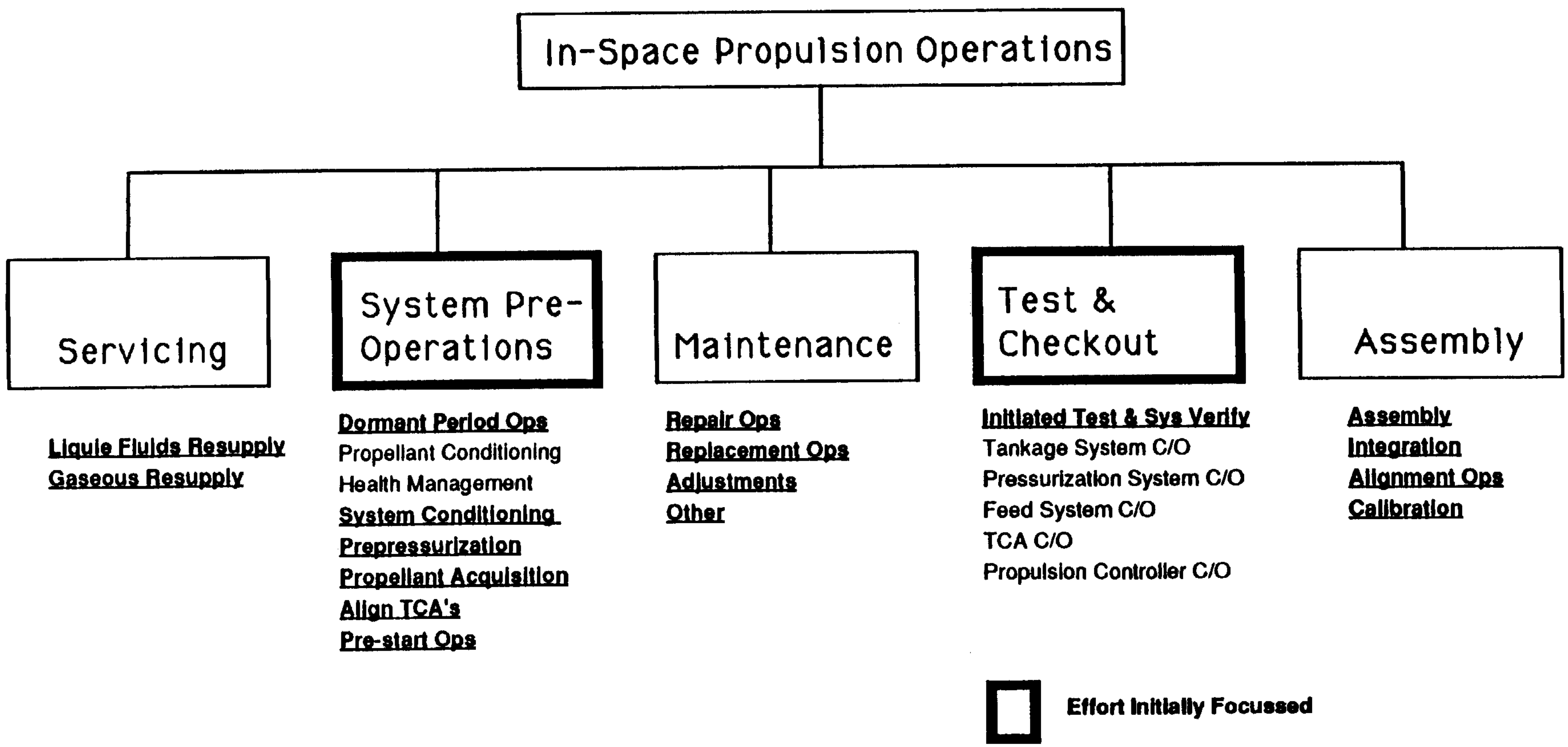



Figure 4.2-2

## Flow Chart of Activity Sequence for Test & Checkout

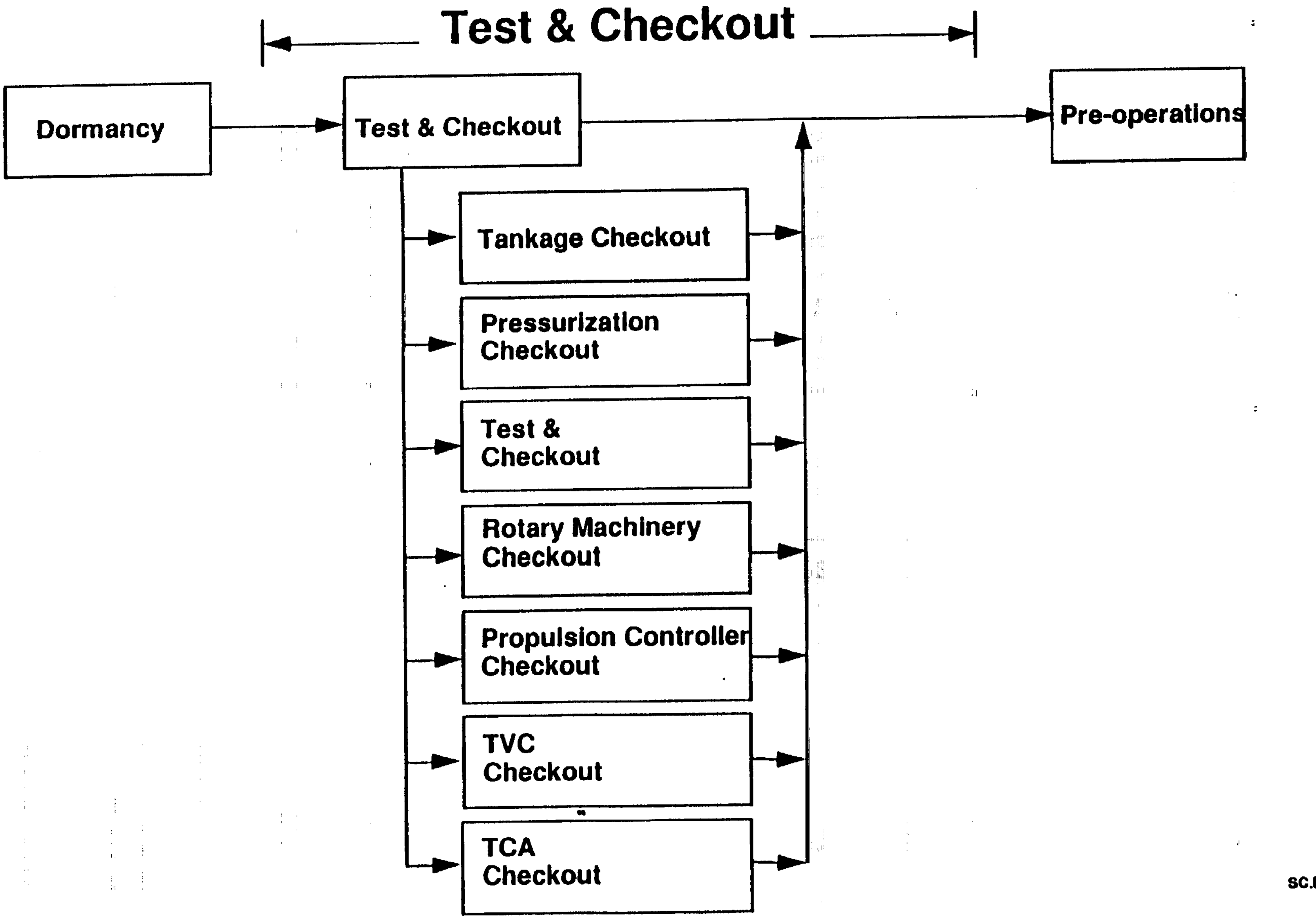



Figure 1.2-3

# Flow Chart of Activity Sequence for Pre-Operations

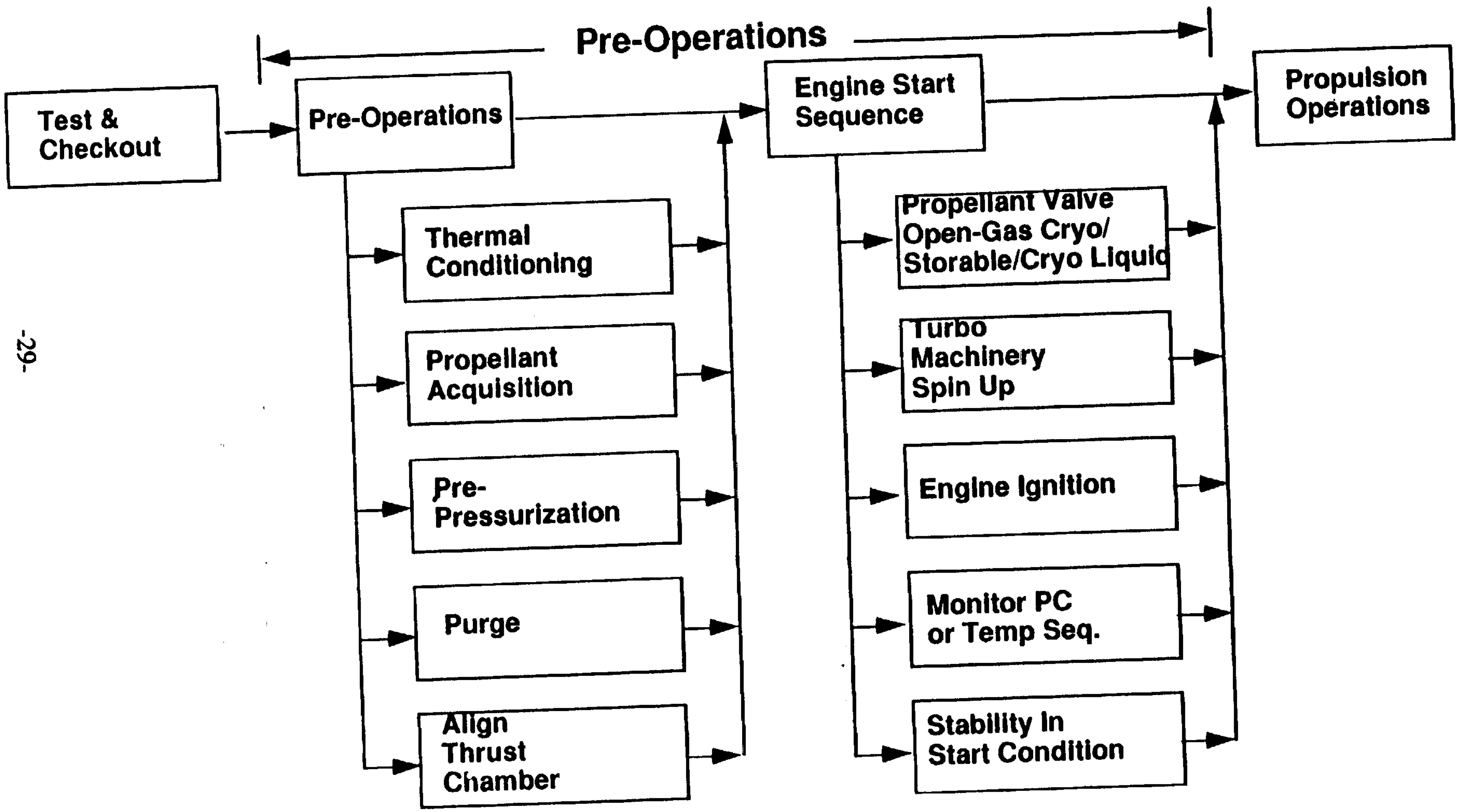
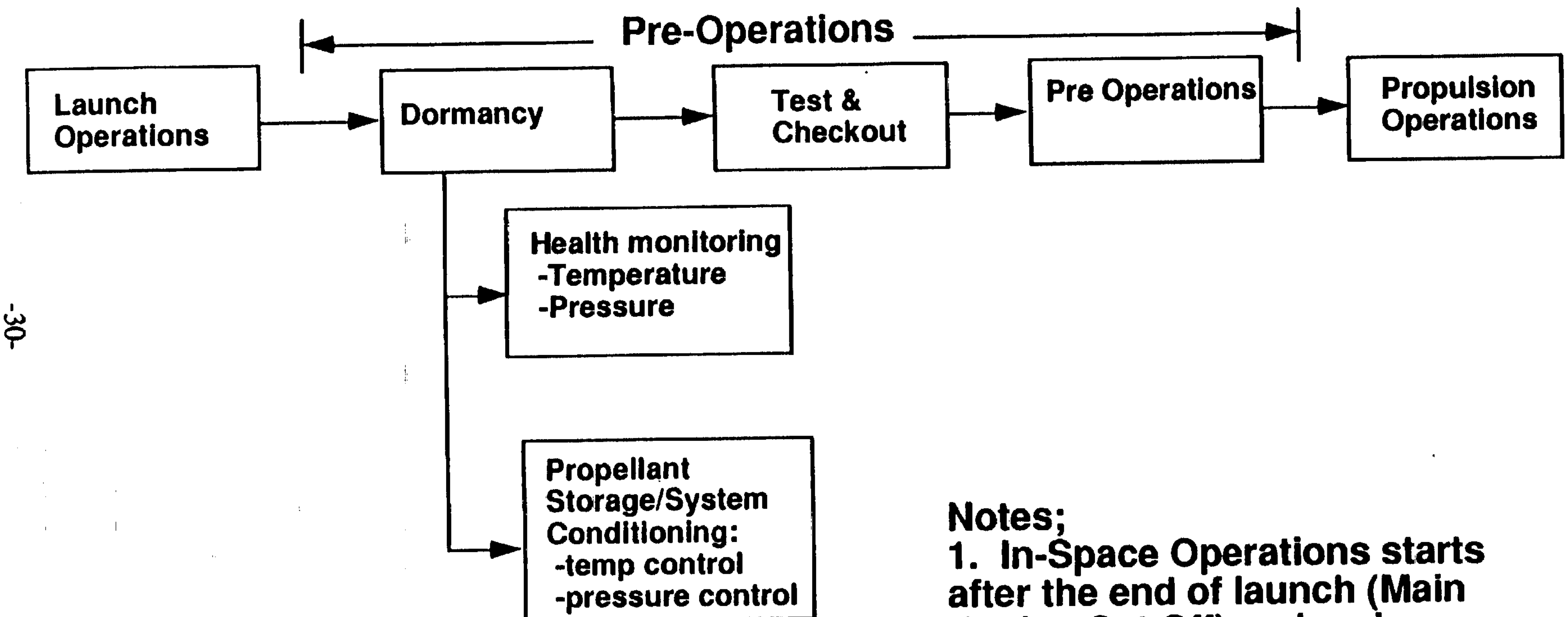



Figure 4.2-4

## Flow Chart of Activity Sequence for In-Space Operations of a Propulsion Subsystem




Notes;
1. In-Space Operations starts after the end of launch (Main Engine Cut Off) and ends just before start of the propulsion operations in space
2. Assembly, Servicing, and Maintenance are also In-Space Operations

SC.II

Figure 4.2-5

**Table 4.2-1 STPOES In Space Concerns/Issues**

1. Liquid/Vapor Management
    - Propellant Acquisition
    - Propellant Gaging
    - Zero G Venting
2. Hardware Dependability
3. Dormant Standby -- Monitoring
4. Fault Tolerant
5. Maintenance
    - Automated
    - EVA
6. Limited Commodities
7. Environment
    - Space Debris
    - Thermal Management
    - Pressure
8. Health Management
9. Tools/Equipment -- Robotics
10. Logistic Support
11. Autonomy
    - Automatic/Manual Operation
12. System Conditioning

### 4.2.3 DEVELOPING A METHODOLOGY

The task of developing the In-Space Operability Index applied the same basic approach used to develop the Launch Operations Index (LOI). The approach for developing the LOI was to: 1) develop a list of concerns/issues that were of importance to operators; 2) devise a list of design features which address the concerns/issues; 3) construct a stepwise scale of options for each design feature; and finally, 4) devise an algorithm for consolidating all the individual scores into a single index.

The above approach was also applied to developing the In-Space Operability Index. A list of design concerns/issues that have caused propulsion problems in the past was also developed, reference Table 4.2-1. This list of concerns/issues evolved from reviewing operations reports (database information) and operators' (NASA, Air Force, and Industry) experience. While the list of concerns/issues for the ISOI are different from the list for the LOA, they both focus on conditions that can result in poor operability if not designed properly. So both the LOI and the ISOI indices start with a list of concerns. For each design feature a stepwise scale is constructed. At the top of the measurement scale is a set of design conditions, which if present, would give the best operability outcome. On the bottom of the measurement scale are a set of conditions, which, if present, would result in the worst operability outcome for that design feature. To assess the score on any particular concern, the designer or decision-maker (who may be unfamiliar with operability) need only match his propulsion system's design with the descriptions of the options on the stepwise scale for a given feature. Allied with each description on the measurement scale is a value associated with how operable that design option is with respect to other choices. By summing the rating scores for each design feature on the list of concerns and taking into account that some design features may be more important than others, an ISOI score can be calculated. The relative importance of different design features is accommodated by assigning them different weighting factors.

While the launch and in-space operations indices have much in common, in-space operations is multifaceted, so in-space operations were divided into five separate classes (see Figure 4.2-2) to acknowledge the fact that different spacecraft missions may require different combinations of these in-space propulsion operations. The intent was to calculate a separate index for each of the five in-space operation classes with a provision for consolidating them into a composite index if appropriate.

In-space operability is a measure of the ability to prepare and to place a propulsion system into operation and to keep it ready for its next use. What are the characteristics of this ability to perform operations in space and how can one tell if it is efficiently completed. While the answers were many and different, they tended to converge into a somewhat common set of characteristics. By examining each benefit and asking why it was important, a chain of reasoning led from many instrumental values to a smaller set of three terminal values, see Figures 4.2-6 and 4.2-7. Thus, the benefits of operability were ultimately reduced to three major categories called attributes.

The following description of developing a methodology against attributes is a first cut devised be Rockwell International and admittedly requires additional work to attain

# What Factors Influence Operability?

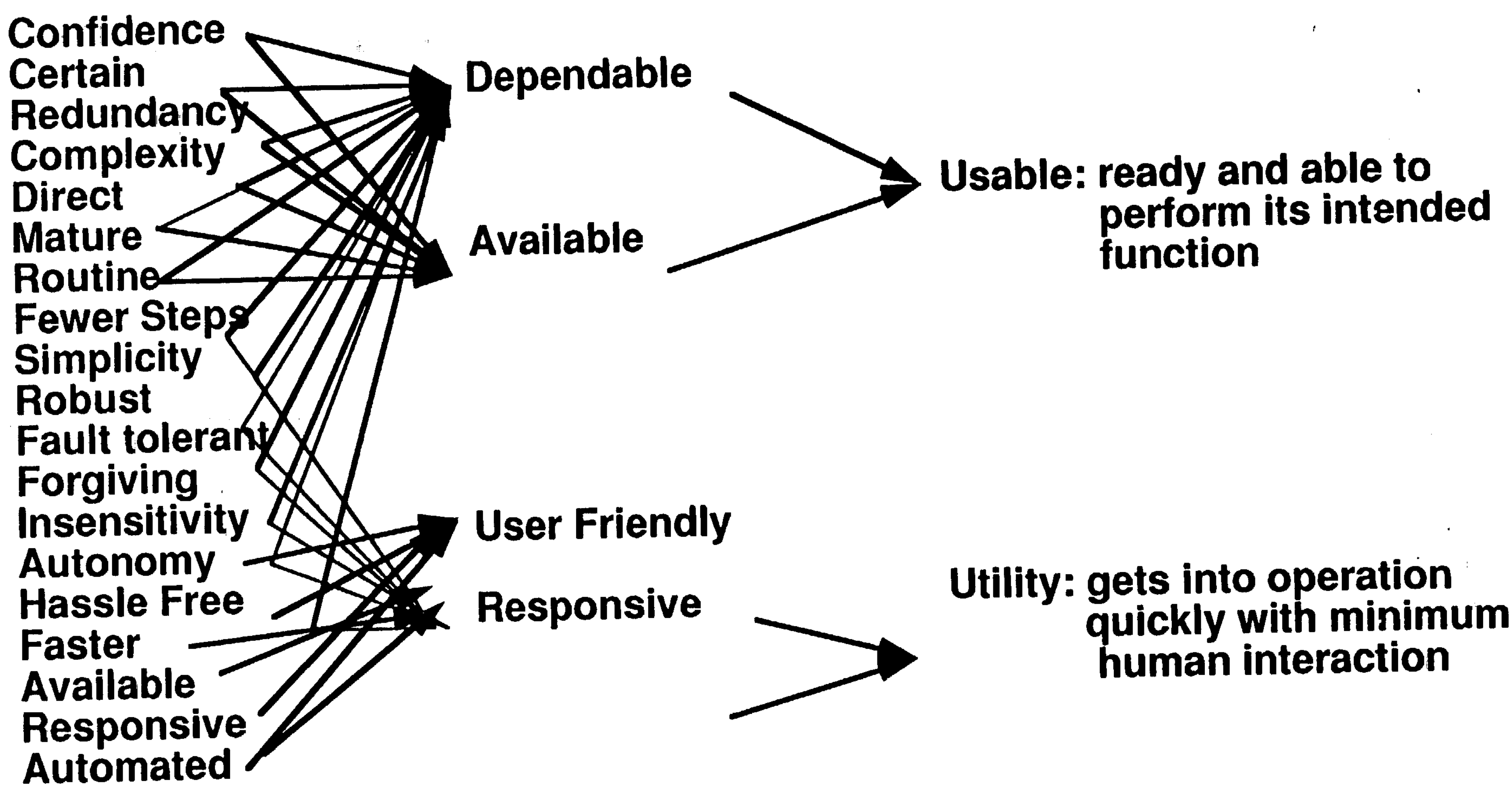



Figure 4.2-6

## Guidance for Constructing an Operability Index for any Space Based Asset

| Worst | Best |
|---|---|
| Operability Conditions | |
| - Many steps required to perform the task | - Few Steps |
| - Slow/time consumming process | - Fast/Responsive |
| - Manual/labor intensive | - Automated |
| - Complex process/tasks (requires high skills or sophisticated tools, equipment, and facilities) | - Simple |
| - Undependable process/ lacking maturity/verfication | - Dependable |
| - Uncertain outcomes (process doesn't always work) | - Certain |
| - Delayed response (operator must wait for prerequisite conditions) | - Immediately response |
| - Significant hassle involved to get the job done | - Routine |
| - Problems often develop | - Few problems |
| - Indirect/many peripherial steps required prior to desired action | - Direct access |

Figure 4.2-7

agreement by the community to allow building an In-Space Operability Index on this foundation. These attributes, their characteristics and weighting requires review by the NASA centers, and the vehicle and propulsion community.

The first attribute of operability is dependability. It was derived from the many different but related terms the experts used to describe the benefits of an operable system. The following terms were used to describe this aspect of operability: confidence: certainty of use, lack of complexity, maturity, robustness, failure resistance, and insensitivity. Redundancy and fault tolerance are measures of accomplishing system reliability because of low hardware reliability and are counter-productive to operability and dependability. Easily supportable systems with minimum interfaces and infrastructures required to support functions are very operable. In addition, fully automated assessments of both systems and hardware health are very operable. The common denominator of all these terms is that the user wants the propulsion subsystem to be available and to work (operate) as planned. Normally, this is a concern over reliability or how likely is the subsystem to fail. In an attempt to make operability an independent variable, the following distinction has been made between dependability and reliability. Failures that occur before the start of the propulsion subsystem are deemed to be a dependability problem. Failures that occur after engine start are deemed to be a reliability problem. Thus, if a propulsion subsystem failure occurs during the dormancy period and the system becomes unavailable until it is repaired (if this is possible), then this failure is a dependability issue. Likewise, if an otherwise available propulsion subsystem fails during the pre-operation activities prior to start-up, this failure is also a dependability issue. But if the failure occurs during the actual operation, then the issue is one of reliability, and not one of dependability. In summary, an operable system is usable; it is ready and able to perform its intended function.

The second attribute, availability, is a combination of terms that are directly related: responsiveness and autonomy. They were derived from the following set of descriptions used by the experts: fast, responsive, automated, simple, autonomous, hassle/worry-free, few steps, mature and direct. Availability is a function of dependability. This characteristic is important for some selected manned missions or unmanned missions whose success depends on a quick response to unplanned events. An example of a manned spacecraft with an unplanned mission schedule is the ACRV, an escape capsule for the Space Station crew. Its primary purpose is to provide a means for an emergency departure. Since the crew may be injured or otherwise less capable due to the nature of the emergency, the ACRV must be able to operate with minimum of human interaction. An example of an unmanned spacecraft with an unplanned mission schedule is a DoD satellite that must react to hostile threats against it such as an incoming anti-satellite missile. In both cases, time is of the essence, making responsiveness an essential feature. Autonomy may also be an essential feature of responsiveness because man-in-the-loop operations slow down the operation. To escape or to evade successfully may require a reaction time in seconds or less. In summary, an operable system is easy to use and fast to respond: it gets into operation quickly with a minimum of human interaction.

The third attribute is directly related to the impact of operations on affordability (cost), especially for ground operations. As previously mentioned, in defining operability, the operations and propulsion experts tended to focus on problems and typically described systems that were difficult and costly to operate. Practically all these descriptions were of ground operations and fraught with delays; unplanned failures and repairs; lengthy and labor intensive tasks; expensive tools, equipment, and facilities; and large "standing armies" to support the operations. All these conditions are associated with high operating cost. In summary, an operable system is inexpensive to operate: it can be operated and is affordable. The cost picture is magnified when operating in space as there are no spare parts, no maintenance capability, no access, and operability problems can result in mission failure.

Each of the five different classes of in-space operations differ in degree of impact on these three attributes. However, all in-space operations within the five classes must be focused upon and resolved. For instance, the in-space operations of test and checkout and pre-operations is considerably less costly than assembly, service, and maintenance. The latter three classes of in-space operations will require an expensive infrastructure to enable them. This infrastructure will be extremely expensive. Hence, there is an incentive to design a system that doesn't need to perform these in-space operations. Once in space, a spacecraft is difficult and thus costly to work on, whether planned (assembly and service) or unplanned (repairs). At present, a spacecraft either works or it doesn't. Most of the cost aspect of space operations is borne by the hardware that must be designed for dependable and reliable in-space operations.

The system's conceptual approach to integrate, simplify and automate using the three attributes, dependability, availability, and affordability, are variables the designer and program manager can influence to improve in-space operability within the five classes of in-space operations. The methodology for measuring how well operability is improved evolved from the process used to develop the Launch Operations Index. For in-space operations a concerns list of design features are defined. Each design feature has multiple steps or options with a stepwise operability rating or rank (to be consistent with the nomenclature of the LOI). Each of these design features is weighted based on its perceived contribution of problems in achieving the three attributes of operability (dependability, responsiveness, and affordability). Design features that have historically caused big problems are given larger weights. A selected list of just those design features that cause the big problems is defined as the List of Concerns. For example, the LOI was compiled out of a list of 17 design features that are significant enough to be concerns based on the experience of the users.

Each of the designs featured in the list of concerns has a measurement scale developed for it similar to the LOI. The feature's stepwise scale is bounded by the conditions that make for the best and the worst operability conditions. Those conditions driving the rating score are directly linked to hardware design. The reason both the ISOI and LOI will be useful in improving operability is their direct link to controllable hardware design options. Operations design features and options can be evaluated and implemented early in the conceptual design process. The design features and design options can be selected to create the operability conditions desired: dependability, responsiveness, and low operating costs. So, at the heart of the ISOI is a measurement scale based on design features and design options that are controllable by the designer. Each item on the list of concerns will be graded (measured) on how well it achieves the three attributes that constitute operability.

### 4.2.4 CONSTRUCTING THE INDEX

The ISOI will be constructed from selected design features that have been identified as past sources of operational problems or drivers. Thus, the first step is to select what design features to include in the index. The design features in the ISOI can be selected from all the activities/steps required to complete one of the operational classes (such as pre-operations). Since there are many possible processes to choose from, the most likely operating process is chosen as the default. This reference process is likely to include the longest and most troublesome activities, typical of past experience. The better way to perform the same operational process would simply be to eliminate activities and/or select design options which facilitate the process.

For any one of the five operational classes, the weights assigned to the design features will be based on the following algorithm. The assignment of weight to each design feature will be based on the relative degree to which it does not normally/currently achieve each of the desired benefits/attributes of operability (dependability, responsiveness, and affordability) adjusted for each attribute's relative importance. The first step is to assign a weight to reflect each attribute's relative importance (for a given operational class). For example, the three attributes for pre-operations were assigned the following weights as shown below:

| Attribute | Weight |
|---|---|
| Dependability | 40% |
| Responsiveness | 30% |
| Affordability | 30% |

The second step is to rank-order the design features based on how well they achieve each of the three attributes. In other words, the rank-order process is repeated three times, once for each attribute. A rank of 1 means that the design feature is seldom a source of problems; it seldom fails to perform its activity and the failure of that activity seldom causes a failure of that operation. Thus, a high rank score means that the design feature is often a source of problems. Then, the rank-order is multiplied by each attribute's weight of importance and then summed.

The last step is to normalize these composite scores so that the design feature with the highest score is converted to a weight of 10. Each of the other design features is given a score based on the ratio of its composite score to that of the highest score and then multiplied by 10. For example, if the design feature has a composite score of 350, it would be assigned the highest weight of 10. If the next highest score were 320, it would be assigned a weight of 9 (320/350 * 10 = 9). With this approach, higher weights will be assigned to those design features that cause the most problems and hence are of the highest concern.

Design features with the higher weights (an indicator of poor operability) will be included in the list of concerns. Each one of these design features must then have a measurement scale constructed for it so that a propulsion design can be rated as to how well it compares to the best and worst conditions. The constructed scale is derived by first defining both the worst and best hardware designs for satisfying the design feature. These two extreme designs are given operability rating scores of 1 for the worst and 10 for the best. Typically, the highest score goes to the design option that completely avoids the activity. If an activity doesn't have to be done at all, then problems cannot stem for it. Design features examples, including stepwise design options, which address in-space concerns and issues are presented in Figures 4.2-8 through 4.2-16. These examples represent a first cut and need to be reviewed and edited by the larger propulsion community.

Given the list of concerns (for each in-space operational class) and the constructed scale for each, the ISOI can now be calculated, see Figure 4.2-17. Taking each design feature one at a time, the evaluator compares the propulsion design being rated to each of the hardware descriptions on the constructed scale, looking for a match. When the hardware descriptions do match, the evaluator gives it the corresponding operability rating. When they do not match, the evaluator gives the design the score of the design option to which it is most similar. The operability score on each design feature is multiplied by its weight and then summed for all items on the list of concerns. The perfect score is 10 times the number of items on the list of concerns. The ISOI score is a ratio of the design score to that of the perfect score, and it is a number less than one. A perfect ISOI score is one.

An in-space operability score is calculated for each of the five operational classes only if the mission calls for it. This ISOI score gives the designer information on how well his design compares to the best design possible. Furthermore, the designer knows how operable his design is in each operational classification. If a composite score were desired across all operational classifications, a composite index could be generated by combining the separate lists into one master list of concerns. The design being rated would then receive a rating score relative to the highest score possible (which is the number of items on the master list of concerns times 10).

### 4.2.5 CONCLUSIONS/RECOMMENDATIONS

The combined LOI and ISOI studies have provided a road map for developing an operations index for propulsion systems. The main task remaining is to get the operations experience into the indices. The knowledge from people with experience and from data bases collected from prior missions needs to be included in the constructed scales for the design features that have historically caused operational problems or that are drivers of probable problems or higher cost operations.

Some of the tasks that need to be done to complete the ISOI include concurrence on definition, classifications, attributes (characteristics), design features of concern to propulsion operators, and the design options for improving them. The best way to collect the combined experience of experts in propulsion and operations is to follow the path used in developing the LOI, that is, initiate and host workshops on the relevant facets of operability with an initial focus on propulsion systems.

## Design Feature # - Storage Conditioning Propellant Thermal Management

(Pressure Control, Prop Quality, Temp Control Feed System)

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10.0 | NOT REQUIRED |
| 9.0 | SPACE HEATERS |
| 8.0 | TANK WALL HEATERS |
| 7.0 | TANK WALL HEATERS W/LINES |
| 6.0 | LOW LEAK BLEED SYSTEM, VCS |
| 5.0 | TVS, IN-TANK MIXER, MULTIPLE VALVES, NO VCS |
| 4.0 | TVS, IN-TANK MIXER, MULTIPLE VALVES, VCS |
| 3.0 | TVS, RECIRC PUMP, MULTIPLE VALVES, NO V CS |
| 2.0 | TVS, RECIRC PUMPS, MULTIPLE VALVES, VCS |
| 1.5 | PARTIAL RELIQUIDATION SYSTEM, OVERBOARD VENT |
| 1.0 | FULL REFRIGERATION SYSTEM, RADIATORS, POWER SOURCE |

Figure 4.2-8

CS.I

# Design Feature # - System Thermal Conditioning

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10 | NOT REQUIRED (VEHICLE DESIGN SUFFICIENT) |
| 7 | PRE-HEAT THRUSTERS |
| 5 | THERMAL CONDITIONING OF LINES (SIMPLE VENT, LINE HEATERS, ETC.) |
| 1 | PRE-CHILL OF LINES & ENGINE INTERFACE (RECIRC SYSTEM REQUIRED) |

Figure 4.2-9

CS.I

## Design Feature # - Propellant Acquisition

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10 | ELASTOMERIC DIAPHRAGM |
| 9 | BLADDER |
| 6 | VANES |
| 5 | START BASKET (SIMPLE PAD) |
| 3 | SCREEN GALLERY (COMPLEX PAD) |
| 1.5 | CRYO LIQUID PAD & PUMP SYSTEM |
| 1 | METAL DIAPHRAGM (SINGLE USE DEVICE) |

Figure 4.2-10

## Design Feature # - Pre-Pressurization

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 1.0 | PRESSURANT CONTAINED IN PROPELLANT TANK (NOT REQUIRED) |
| 9.5 | SINGLE TANK - SINGLE ISOLATION VALVE SET BETWEEN PRESSURANT & PROPELLANT (BLOW DOWN SYSTEM) |
| 9.0 | MULTIPLE TANK - SINGLE ISOLATION VALVE SET BETWEEN PRESSURANT & PROPELLANT (BLOW DOWN SYSTEM) |
| 7.0 | REGULATED PRESSURANT, SINGLE ISOLATION VALVE |
| 4.0 | HOT GAS GENERATOR - LIQUID PROPELLANT (AUTOGENOUS) |
| 5.0 | HOT GAS GENERATOR - SOLID PROPELLANT |
| 2.0 | SIMPLE CRYO VENT |
| 1.5 | SUB-COOLED HELIUM, HEAT EXCHANGER, PRESSURE FED |
| 1.0 | SUB-COOLED HELIUM, HEAT EXCHANGER W/PUMPS |
| .5 | EXTERNAL HOOK-UP TO PRESSURANT (EVA OR ROBOTIC) |

Figure 4.2-11

CS.I

## Design Feature # - Purge

| OPERABILITY RATING | FEATURE OPTION |
| --- | --- |
| 10 | NONE REQUIRED |
| 7 | BUILT-IN W/SENSOR - AUTOMATIC ACTIVATION |
| 5 | BUILT-IN W/SENSOR - MANUAL ACTIVATION |
| 2 | PORTABLE SYSTEM (EVA OR ROBOTIC) |
| 1 | CONTINUOUS |

Figure 4.2-12

## Design Feature # - Propellant Valve Open - Gas Cryo/Storable/Cryo Liquid

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10 | ENERGIZE PYRO ONE-SHOT NORMALLY CLOSED VALVE |
| 9 | ENERGIZE SOLENOID VALVE/TALKBACK OK (REMOTE & CLOSEABLE) |
| 7 | LINE PRESSURE ACTIVATED VALVES |
| 5 | BALL VALVE - REMOVE ACTIVATION |
| | o TAILORED START TRANSIENT |
| | o LARGER ENGINES |
| 3 | LIQUID CRYO VALVE (GAS ACTUATED) |
| 2 | LIQUID CRYO VALVE (ELEC MOTOR ACTIVATED) |

Figure 4.2-13

## Design Feature # - Propellant Utilization & Mixture Ratio Control

| OPERABILITY RATING | FEATURE OPTION |
| --- | --- |
| 10 | SELF-CONTAINED IN ENGINE |

Figure 4.2-14

## Figure 4.2-15

## Design Feature # - ESS - Turbomachinery Spin-Up

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10 | ENGINE TAP-OFF OF DRIVE FLUID |
| 7 | GAS GENERATOR DRIVE (ENGINE PROPELLANTS) |
| 4 | HELIUM/$N_2$ GAS |
| 1 | ONE-SHOT CARTRIDGE (J-2 TYPE) |

Figure 4.2-15



CS.I

## Design Feature # - Engine Ignition

| OPERABILITY RATING | FEATURE OPTION |
|---|---|
| 10 | NO IGNITOR REQUIRED (HYPERGOLIC PROPELLANTS OR MONOPROP/CATALYST) |
| 7 | SPARK PLUGS |
| 6 | AUGMENTED SPARK IGNITORS |
| 4 | HYPERGOLIC SLUG IGNITORS |
| 1 | PYROTECHNIC |

Figure 4.2-16

OEPSS

Operationally Efficient
Propulsion System Study

## Calculating the Space Based Operability Index (SBOI)

Space Based Operations

Flight Control/ Monitor | Service | Maintain | Test/ Checkout | Assemble

↓

Calculate an Operability Score for each Operation

↓

$\Sigma$ ( Weight of Activity) X ( Operability Score for the Activity)

Summation
(for all Activities
in the Sequence)

↓

Normalize to 0 – 1 Scale

↓

SBOI = (Score on, Control  Score on, Service  Score on , Maintence  Score on, Checkout  Score on ) Assembly

Rockwell International
Space Systems Division

Figure 4.2-17

# 5.0 PROPULSION SYSTEM CONCEPTUAL DESIGNS

The Propulsion System Conceptual Designs task objective was to develop conceptual designs which minimize operability concerns and issues. Three subtasks were completed; Requirements Definition, Conceptual Layouts, and Conceptual Designs and Analysis. NASA LeRC developed the conceptual design requirements. Propulsion system descriptions were produced for the concept design layouts. The conceptual designs and analysis task was of sufficient depth to indicate that concept designs were viable and requirements could be met.

## 5.1 REQUIREMENTS DEFINITION

The STPOES propulsion system conceptual design requirements were developed by NASA. The mission /vehicle is a Lunar Lander Descent Stage. Major propulsion requirements were: Thrust between 60,000 to 80,000 pounds, $LOX/LH_2$ propellants and 10:1 throttling. Table 5.1-1 lists the propulsion system characteristics and requirements. Information from the First Lunar Outpost (FLO) workshop, held at NASA JSC on August 13-14, 1992, was the point of departure the propulsion system design characteristics used to produce conceptual layouts and design descriptions.

## 5.2 CONCEPTUAL LAYOUTS

Lunar Lander requirements developed in section 5.1 were used to define several Lunar Lander propulsion system concept designs. The primary design objective was to eliminate or mitigate propulsion system operability concerns and issues, both launch and in-space. The in-space concerns and issues are described in Section 4, reference table 4.2.1. A propulsion system is defined as propellant tankage, propellant distribution and the rocket engine(s). The Lunar Lander baseline concept architecture includes separate descent and ascent propulsion systems. Descent stage propulsion systems were designed. The propulsion system is ground based and expendable. The total thrust range requirement was from 60,000 to 80,000 lbf. Rocketdyne chose an 80,000 lbf. thrust level for the design point.

The envelope for the propulsion system came from JSC's selected baseline FLO Lunar lander vehicle. The stage diameter is 10 meters (393.6 in.) not including landing legs. The nozzles of the thrust chamber may extend below the tanks to within 0.8 meters (31.5 in.) of the ground. The bottom of the vehicle, except for the nozzles, has "walk under" clearance of 2.1 meters (82.68 in.). A central core diameter, for the ascent propulsion system, is 4.5 meters (175.73 in). The height of the envelope is assumed to be whatever is required for the propellants, with a design goal to be as short as possible.

An initial concept of a six thrust chamber modular assembly and one turbopump set was revised to a four thrust chamber and two turbopump sets. Four thrust chambers with the same chamber pressure were lighter and less complex. Two turbopumps sets enhance throttling, i.e., when the propulsion system is throttled to 50%, one turbopump set will be shut down and held in a redundant standby mode. This approach reduces the turbopump throttling requirement to 5:1. Having a pump set in the stand by mode also enhances system reliability, i.e., the system has turbopump out

Table ں.1-1

# STPOES Conceptual Design Requirements

- **Mission/Vehicle Requirements:**

| | |
|---|---|
| • Application: | Lunar Lander - Descent Stage Evolution |
| • Staging: | Two Stage (Separate Descent and Ascent Prop. Modules) |
| • Mission Profile: | Circularization Burn, Deorbit Burn, Terminal Descent and Landing |
| • Descent Payload: | 35 MT (Includes Ascent Stage, Plus Crew, Plus Payload) |
| • Hardware Reuse: | Expendable |
| • Total Stage Vac. Thrust: | 266.9 to 355.9 MN (60,000 to 80,000 lbf) |
| • Propellants: | LOX/LH2 |
| • Fault Tolerance: | Zero Fault Tolerance for Descent Stage (Single Fault Tolerance for Ascent Stage) |
| • Throttling: | 10:1 |
| • Stage Diameter: | 10 Meters (32.8 ft) Not Including Legs |

- **Operational Attributes:**

| | |
|---|---|
| • Operability: | Operability Indices >0.9 |
| • Reliability: | >0.99 |
| • Cost: | Lowest Recurring and Non-Recurring Cost |

capability. The engine length was restrained by ground clearance requirements. A thrust chamber nozzle expansion ratio of 440:1 was used to stay within the engine length restraint.

Several propellant tankage arrangements were proposed around the four thrust chamber/two turbopump set modular engine arrangement. These propellant tankage arrangements led to the development of four concept designs. Each design iteration evolved to a more operational system. A brief description of the four concept designs is presented below.

Concept A consists of four thrust chambers and two sets of turbopumps integrated into a modular configuration with four hydrogen tanks and a single oxygen tank. Concept A is shown in Figure 5.2-1. This Integrated Propulsion System (IPS) uses a LOX/$LH_2$ hybrid power cycle with an integral Reaction Control System (RCS) and is capable of throttling to 10% of the nominal thrust. Gaseous Oxygen and Hydrogen RCS thrusters provide vehicle reaction control, replacing a separate monopropellant or storable hypergolic propellant RCS system. This approach eliminates the use of multiple propellants on the vehicle. In addition, the high pressure gaseous RCS propellants can be used to spin start the turbopumps, increasing the available power to the turbines. Figure 5.2-2 presents a schematic of the Concept A propulsion system.

The Concept A tank arrangement consists of a single spherical oxidizer tank surrounded by four hemispherical end cylindrical fuel tanks mounted horizontally. Turbopumps are essentially tank mounted with minimal inlet plumbing. Tank mounting the turbopumps simplifies pump cryogenic conditioning. Concept A did not address propellant acquisition or liquid/vapor issues. Each thrust chamber was positioned in the interstice between the fuel tanks and the oxidizer tank. A list of the operability features incorporated into the design is presented in Table 5.2-2. An isometric view of the Concept A propulsion system is shown in Figure 5.2-3. Concept A system dimensions are presented in Figure 5.2-4. Table 5.2-3 presents a weight breakdown of this propulsion system concept.

Concept B (not shown) is a design variation of Concept A using supercritical cryogenic propellant and propellant tanks. Having supercritical propellants eliminates concerns with propellant acquisition, propellant settling and sloshing. Propellant tank pressures of 200 and 750 psia were assumed for the hydrogen and oxygen supercritical tanks. The higher pressure tanks simplify the propulsion system by eliminating boost pumps and separate RCS $GH_2$ and GOX tanks. The Concept B design (not shown) is similar in configuration to the Concept A design except propellant tanks are heavier, to accommodate supercritical propellant conditions, and the boost pumps and RCS tanks are eliminated, which mitigates to some extent the heavier tank weights.

A weight estimate was made of a typical tank for propellants at supercritical pressures and temperatures. The Hydrogen and Oxygen properties data are taken from NBS data and the propellant conditions used are presented in Table 5.2-4. Weight estimations were made for a cylindrical, hemispherical end hydrogen tank and a spherical oxygen tank at nominal cryogenic conditions and at the supercritical conditions. Aluminum tank material was assumed with a design stress of 50 Kpsi and a density of .1 lb/in^3. Wall thicknesses were calculated for the cylindrical and spherical

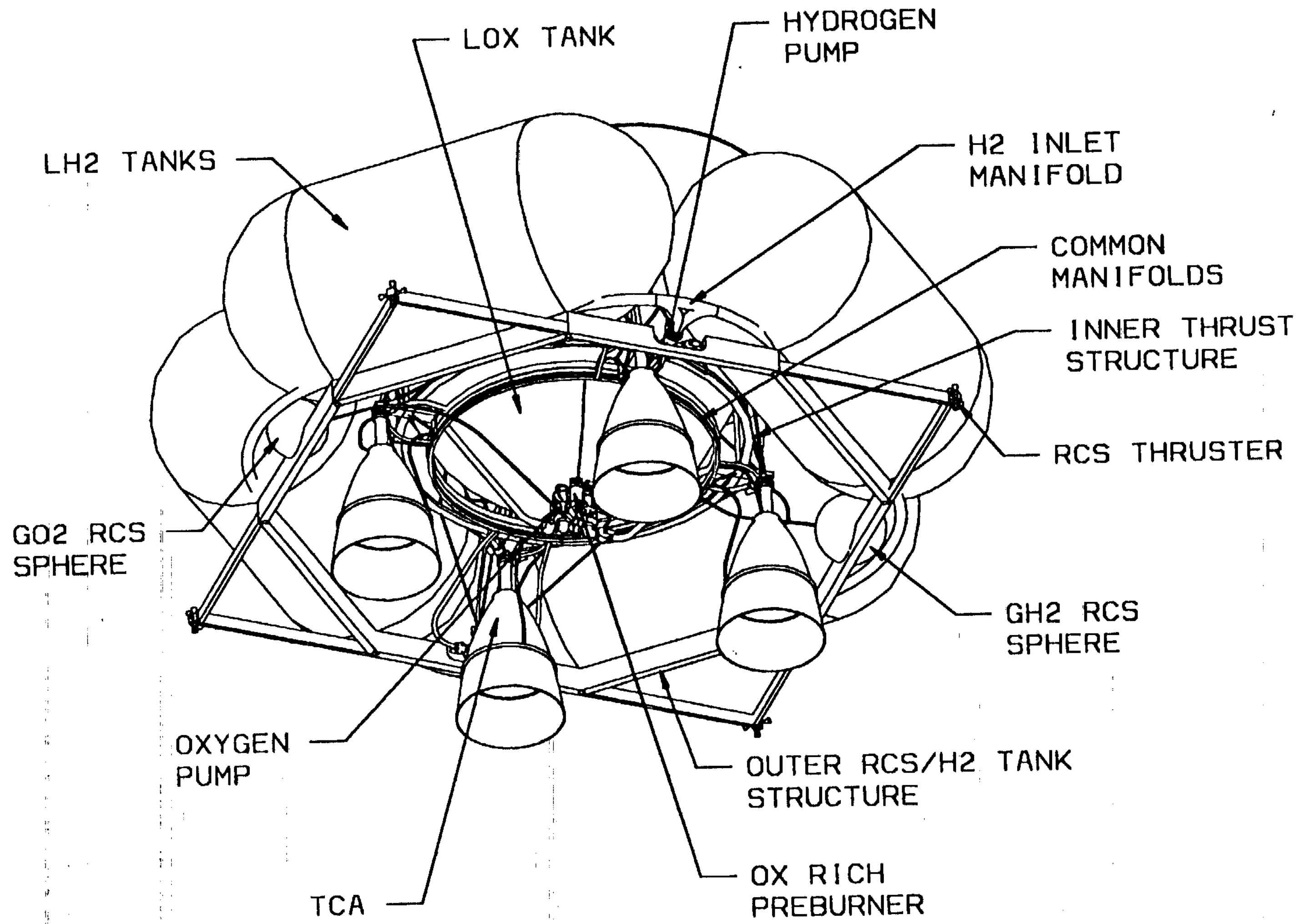



Figure 5.2-1 Design Concept A

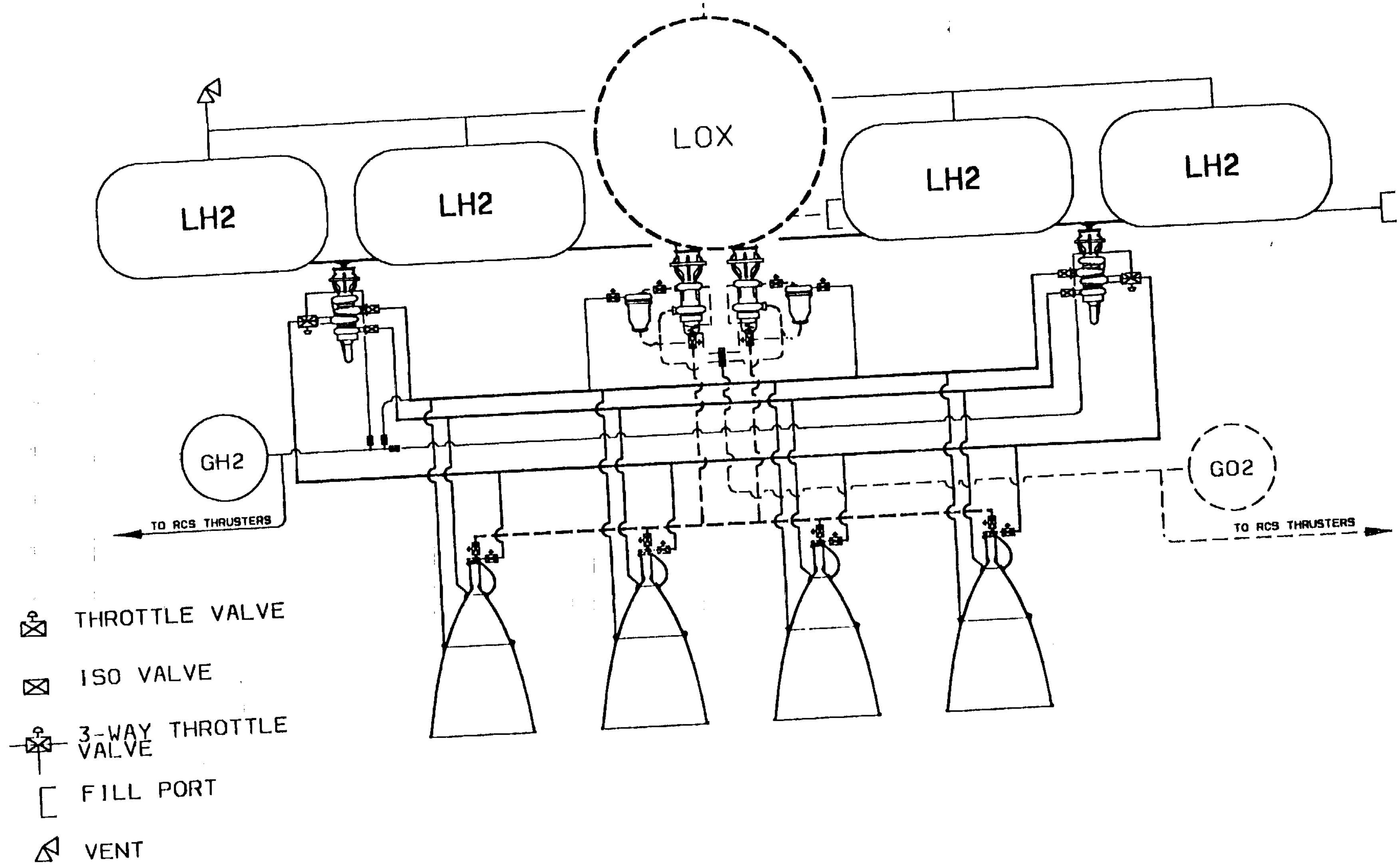



Figure 5.2-2 Design Concept A -- Propulsion System Schematic

## Table 5.2-2 IME/Lunar Lander Design Features

- Open Propulsion Compartment
- Automated Checkout
- Two Fluid System -- $LOX/LH_2$
- $O_2/H_2$ RCS
- Laser Ignition (Engines & Ordnance)
- EMA Actuators
- Differential Throttling TVC (no Gimbal)
- Zero NPSH Pumps (no Tank Pressurization)
- Integrated Systems
- Accessible Components
- Turbopumps Interfaced Directly to Propellant Tanks (no preconditioning)
- No Hydraulics, Pneumatics, Helium, Hypergolics, Monopropellants APU's, Gimbal Systems, Flex Lines

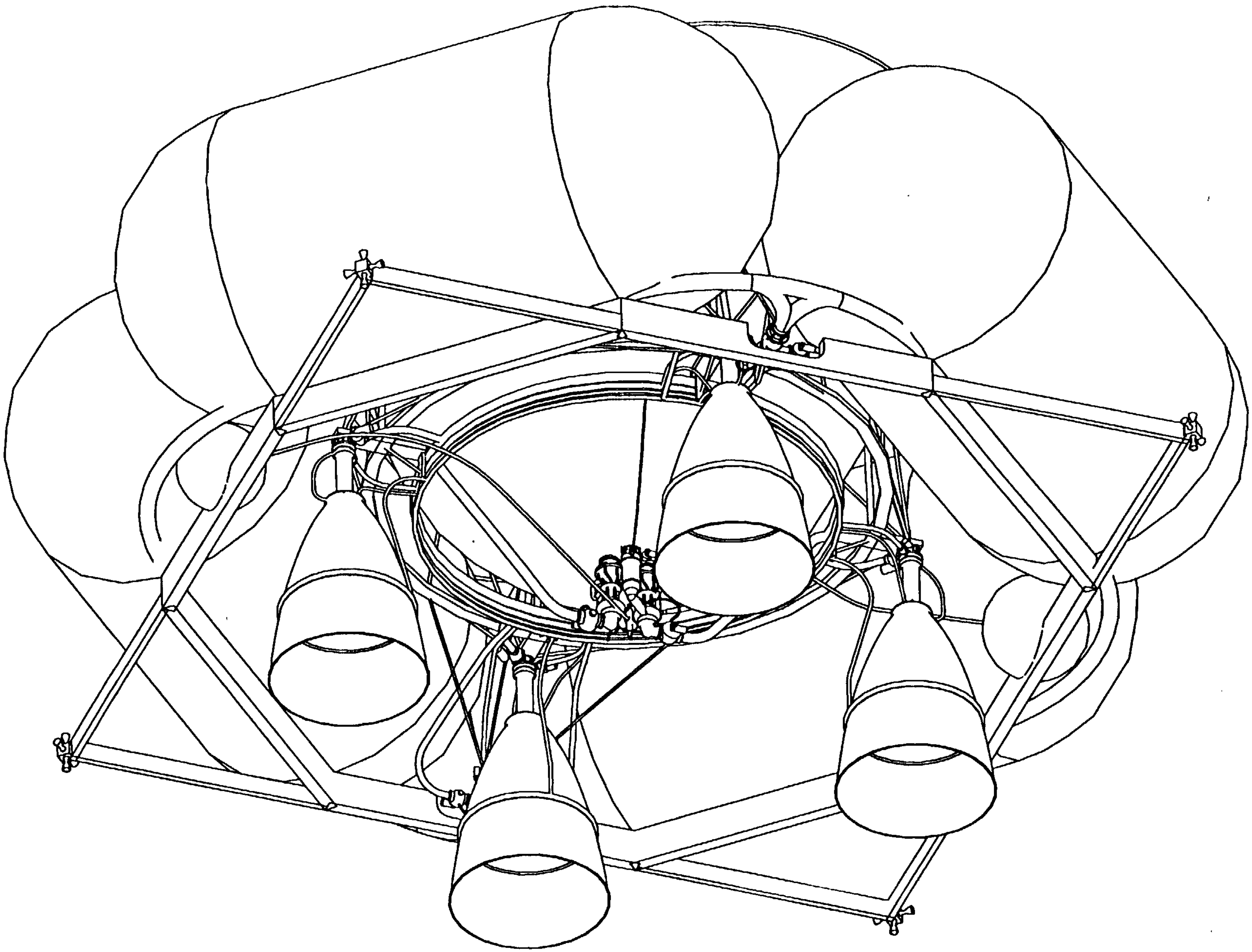


Figure 5.2-3  Design Concept A -- Bottom Side Isometric View

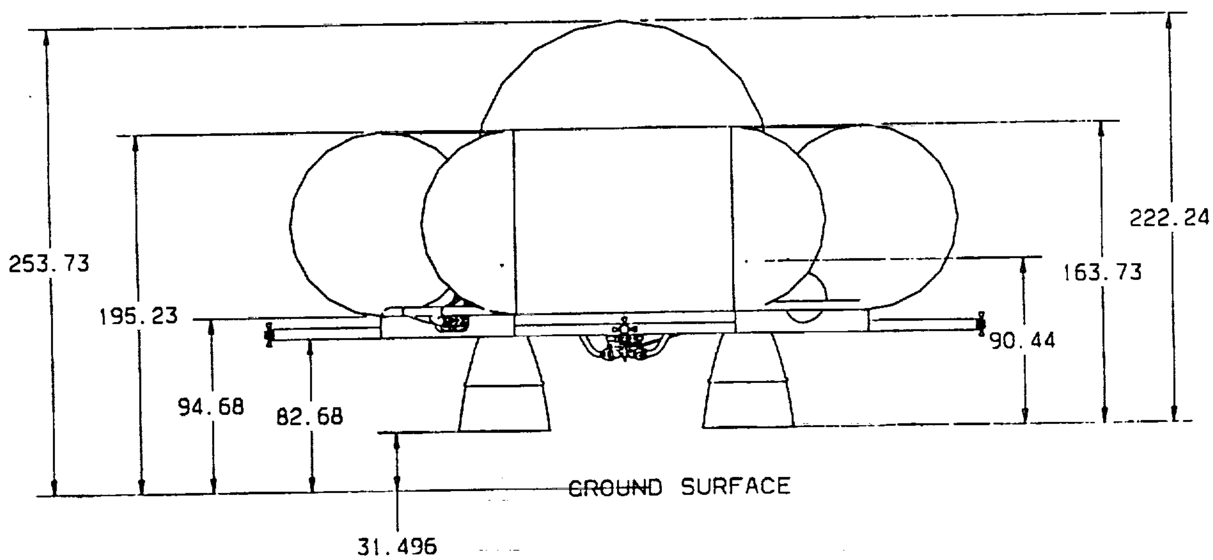



**Figure 5.2-4 Design Concept A -- Dimensions (Inches)**

Table 5.2-3

# Design Concept A
# STPOES Weight Breakdown

**WITH COMPOSITE MATERIALS**

- **80 Klbs 4 Modular Bells 2 Pump Sets**
- **H2 - Expander, O2 - PreBurner Cycle**
- **Pc = 2297 psia, exp 440**

| | |
|---|---|
| **Combustion Chambers (4)**<br>Graphite epoxy overwrap | **199** |
| **Regenerative Cooled Nozzle (4)** | **429** |
| **Film / Radiation Cooled Nozzle (4)** | **116** |
| **Turbopumps (SLIC w/ integrated jet Boost) (2)** | |
| **Hydrogen**<br>Advanced material | **48** |
| **Oxygen** | **124** |
| **Valves**<br>H2 MMC | **85** |
| **Propellant Ducts**<br>H2 Superplastically formed MMC | **203** |
| **Oxidizer Preburner (2)** dia = 4.69 | **45** |
| **Harness, Sensors, & Ignition** | **42** |
| **Controller** | **24** |
| **Thrust Cap & Pump Structure** | **37** |
| **Misc. parts** | **68** |
| **TOTAL** | **1420 lbs** |
| **T/W =** | **56.4** |

Table 5.2-4

# STPOES Supercritical Propellants (Concept B)

**Hydrogen nominal liquid cryogenic properties**

P = 32 psia T = 38°R rho = 4.37 lb/ft^3

**Hydrogen critical properties**

P = 187.5 psia T = 59.4°R rho = 1.962 lb/ft^3

**Selected Hydrogen Supercritical properties**

P = 200 psia T = 40.4°R rho = 4.37 lb/ft^3

**Oxygen nominal liquid cryogenic properties**

P = 47 psia T = 164°R rho = 70.98 lb/ft^3

**Oxygen critical properties**

P = 731.4 psia T = 278.2°R rho = 27.23 lb/ft^3

**Selected Oxygen Supercritical properties**

P = 750 psia T = 168°R rho = 70.98 lb/ft^3

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ends separately. A minimum material wall thickness of .045 was assumed. A safety factor of of 1.5 was used. Delta weight increases were 1,267 lbs. each and 12,364 lbs. for hydrogen and oxygen tanks respectively. Composite wrapped tank weights were also estimated using a PV/W of 900,000. Delta tank weight increases were 408 lbs. each and 2290 lbs. for hydrogen and oxygen tanks respectively. These properties and calculations are only intended for comparison of the two estimated weights of nominal cryogenic and supercritical tanks. The weight estimations for the individual tanks are summarized in Table 5.2-5. Supercritical propellants provide significant operability enhancements, however, there are significant propellant tank weight penalties associated with this approach. Mitigating the tank weight increases are weight reductions from eliminating RCS propellant tanks, reducing the number of fill and drain valves (only one set would be needed), eliminating propellant acquisition systems, and eliminating boost pumps. The total propulsion system delta weights need to be evaluated, however, schedule and funding did not permit this analysis.

Concept C modifies the original concept by substituting concentric toroidal cylindrical tanks for conventional cylindrical tanks. Concept C is shown in Figure 5.2-5. A propulsion system schematic is presented in Figure 5.2-6. The toroidal tank arrangement allows tank mounted turbopumps, eliminates multiple propellant tanks, simplifies propellant tank loading and venting, and incorporates an open central core for the ascent engine. Figure 5.2-7 presents a shaded image of the toroidal tank system.

The toroidal tanks were sized to contain the same volume of propellants as Concept A. The outer diameter of the hydrogen tank was set at 10 meters (392.7 inches). Both the hydrogen and oxygen toroidal tanks have the same internal radii. A straight wall section was added to the hydrogen tank to increase its volume. The overall height is greatly reduced, and a large open area is provided in the center core for the ascent engine. The launch vehicle shroud is simplified as the outer diameter of the hydrogen tank is the stage outside surface. The large center core allows room for the ascent engine, payload, and the crew module. Figure 5.2-8 shows a side view with dimensions.

The Integrated Propulsion System weights are similar to Concept A with an additional weight of 18 lbs due to a larger diameter for mounting the thrust chambers and turbopumps. Toroidal tanks are not as weight efficient as spherical or cylindrical tanks for the same volume and pressure. However, these propellant tanks would be low pressure tanks ( < 32 psi $H_2$, < 47 psi $O_2$ ) designed for minimum fabrication wall thickness.

Concept D (not shown) is similar to Concept C except the propellant tanks are sized for a single descent and ascent lunar lander stage. A combined descent and ascent propulsion system would have a higher, total vehicle, overall operations index. The inner toroidal oxidizer tank would be replaced with the appropriately sized spherical tank and the hydrogen tank would be stretched for the added propellant required for descent and ascent.

Table 5.2.5

# STPOES Supercritical Propellants (Concept B)

**TANKAGE WEIGHT for 4 H2 - 1 O2 CONCEPT B**

**Hydrogen**

| | |
|---|---|
| nominal cryogenic Aluminum tank | 321 lbs ea. |
| Supercritical Aluminum tank | 1588 lbs |
| Composite wrapped tank PV/W = 900,000 nominal cryogenic | |
| nominal composite tank | 78 lbs |
| Supercritical composite tank | 486 lbs |

**Oxygen**

| | |
|---|---|
| nominal cryogenic Aluminum tank | 827 lbs ea. |
| Supercritical Aluminum tank | 13,191 lbs |
| Composite wrapped tank PV/W = 900,000 nominal cryogenic | |
| nominal composite tank | 153 lbs |
| Supercritical composite tank | 2,443 lbs |

STPOES SCT TDC 10/8/92

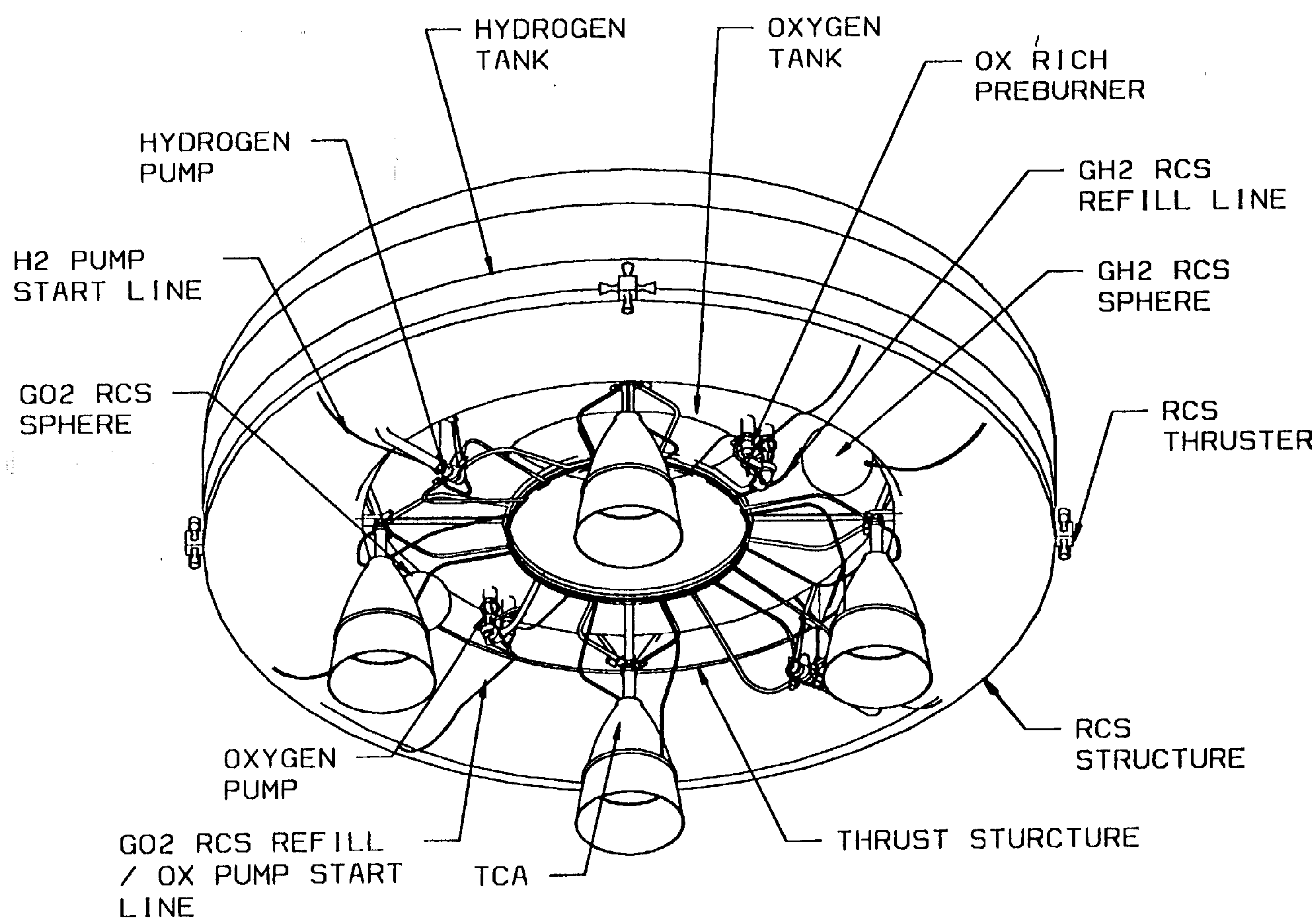


**Figure 5.2-5 Design Concept C**

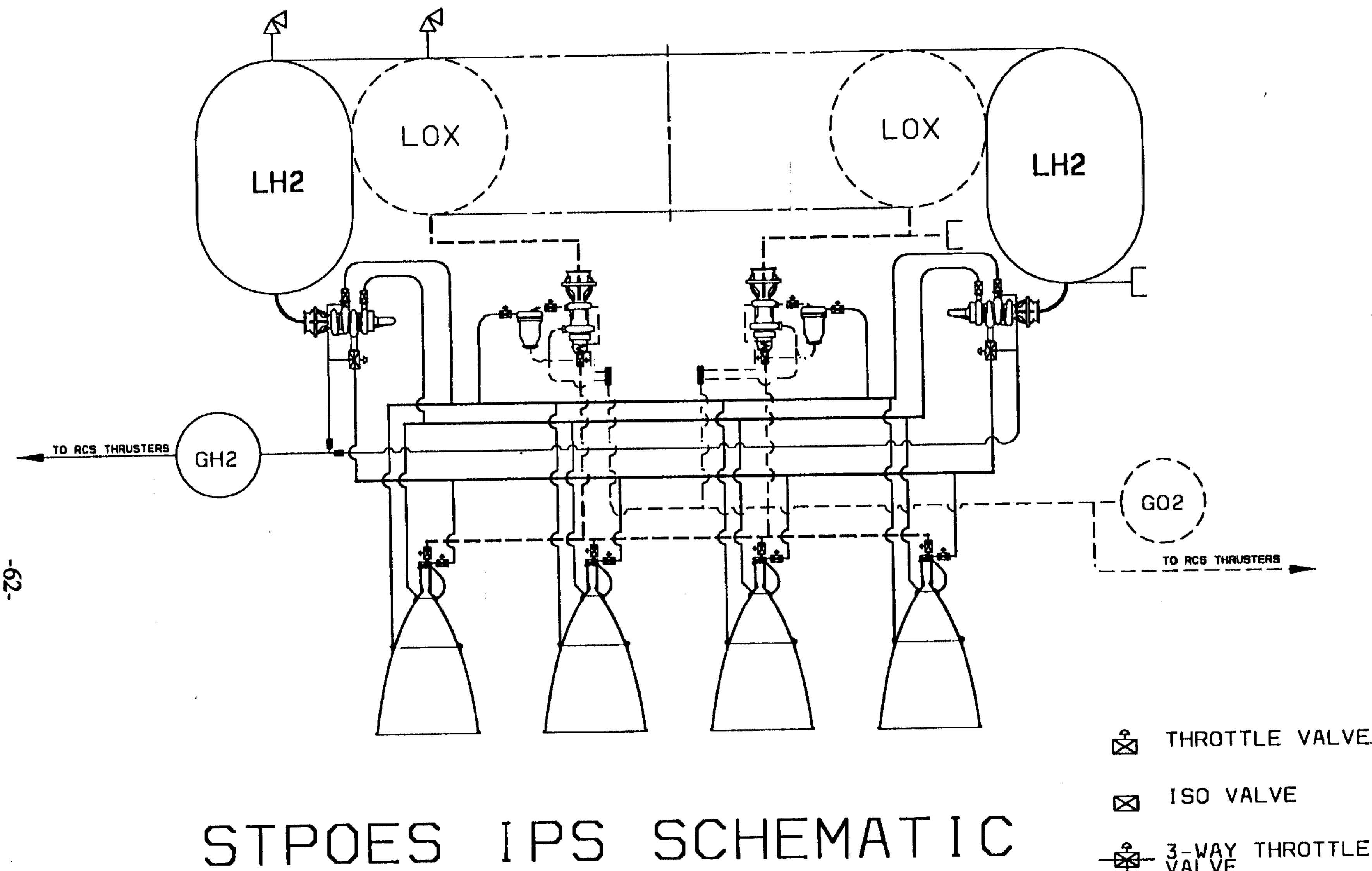


**Figure 5.2-6**
**Design Concept C Propulsion System Schematic**

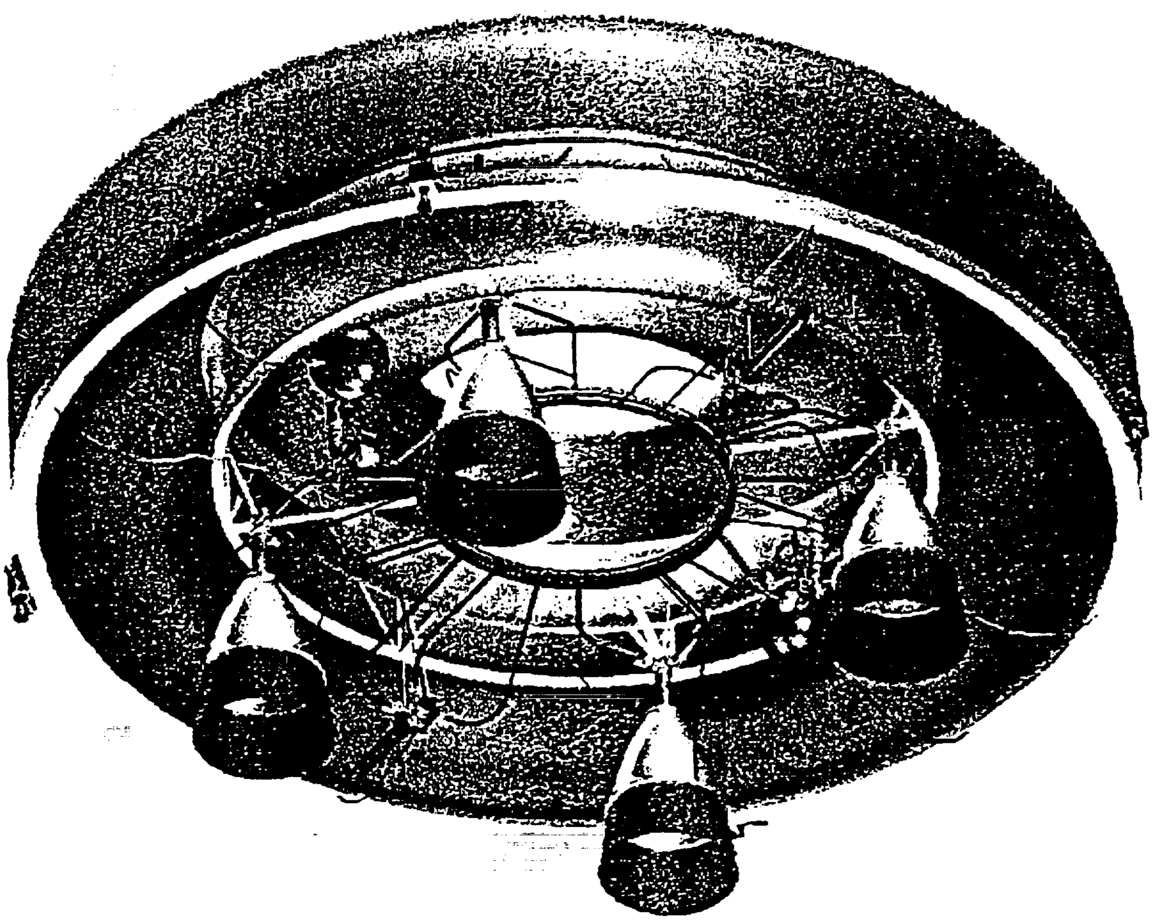

Figure 5.2-7 Concept C -- Toroidal Design

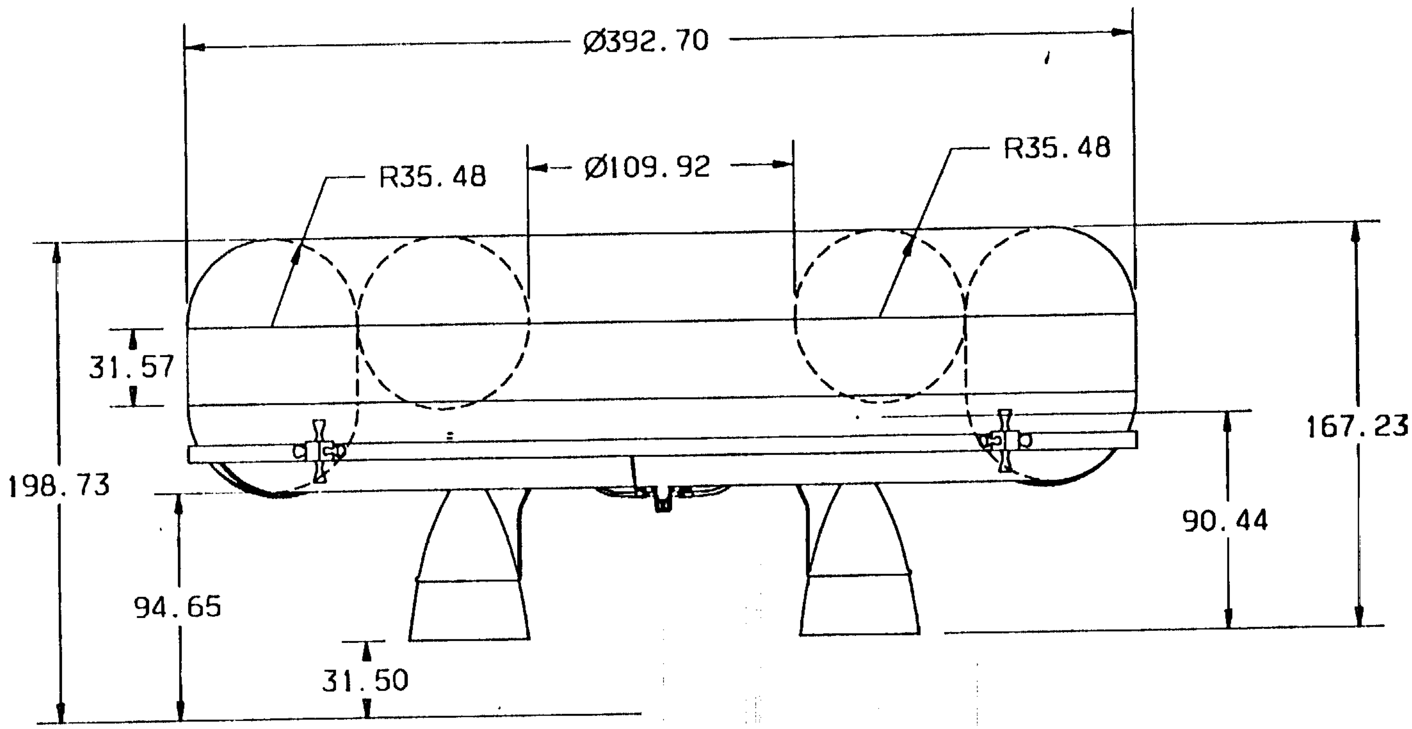


Figure 5.2-8 Concept C -- Dimensions (Inches)

Concept D is beyond the study scope, but is presented to further illustrate the advantages of an operations focus. This system maximizes integration (using one stage to replace two), minimizes parts count, and could be a reusable vehicle (assuming highly reliable hardware and an automated system).

The same Integrated Propulsion Module engine components were used on all the concept designs. Common manifolds feed hydrogen and oxygen to the individual thrust chambers, duct hydrogen coolant to the thrust chambers, and duct heated hydrogen to the fuel turbines. The turbopumps are zero NPSH SLIC pumps with integrated jet boost pumps. The oxygen preburners are a separate components. All valves on the system are Electromechanical Actuated (EMA) valves. Vehicle thrust vector control is achieved by differential throttling; i.e., the thrust chambers are fixed. A laser fiber optic device is used for ignition.

An integrated Reaction Control System (RCS) and descent propulsion system further simplifies the system. Common RCS propellant tanks can be realized by operating the RCS system at a mixture ratio of 16. In Concept A, high pressure (≈ 4500 psi) gaseous propellants can also be used to spin start the turbopumps. The gaseous propellants used during start-up would be replaced by the main propulsion system during main engine operation. The gaseous hydrogen is replaced by tapping off the common turbine inlet manifold. Gaseous oxygen replenishment would be from a heat exchanger in the turbine exhaust manifold. This gaseous oxygen can also be used in the fuel cells, the life support system, and the oxidizer tank pressure control system..

An additional operational enhancement for the Integrated Propulsion Module has a hydrogen (or oxygen) gas-only RCS. This would simplify the RCS propulsion system by eliminating the gaseous oxygen feed system, tanks and igniter. Another RCS possibility is a multi-level RCS thruster capable of operating on $GH_2/GO_2$, $GH_2$ only, or $G0_2$ only. This multi-level capability could reduce the impact of RCS jet exhaust impingement during vehicle rendezvous.

Internal baffles, liquid propellant traps and venting systems are incorporated into Concepts A, C and D propellant tank designs. The supercritical tanks in Concept B would only require a vent system. The simplest operational approach occurs with supercritical tanks on Concept D.

## 5.3 CONCEPTUAL DESIGNS & ANALYSES

### 5.3.1 REQUIREMENTS AND GROUND RULES

The STPOES conceptual design is a $LOX/LH_2$ propellant, expendable, Lunar Lander propulsion system providing 60 Klbf to 80 Klbf of thrust. The terminal descent and landing maneuvers require the propulsion system to throttle down to 10% of the nominal thrust with zero fault tolerance. Additional ground rules not specifically defined in the requirements list, but which are representative of typical SEI needs, were also adopted. These include operation with nearly zero propellant NPSH, nominal MR of 6:1, and use of advanced technology and materials in design. These top level requirements are in addition to the operational attributes specified in Section 5.1

### 5.3.2 BASELINE EVOLUTION

An integrated modular approach was adopted as the baseline configuration for the Lunar descent engine. This type of configuration has multiple turbopumps (T/P) combined with multiple conventional bell thrust chambers and the T/P's feed common collection manifolds which in turn feed the T/C's.

A hybrid power cycle was chosen for the STPOES engine. In this cycle the fuel turbopumps are driven with warm hydrogen heated in the cooling circuits, while the LOX T/P is powered by a LOX-rich preburner. This cycle provides the operational advantages of simplicity, the benign environment of an expander cycle in the fuel turbine, and the elimination of an inter-propellant seal in the LOX T/P. In addition, the gas-gas injection in the main combustion chamber facilitates deep throttling.

Considerable difficulties were found in several of the components at the deep throttled condition when a single turbopump set was used. These included $LH_2$ and LOX pump operation at unstable points, difficult control of an extremely low flowrate of $GH_2$ to the LOX-rich preburner and low LOX injection pressure drop in the preburner possibly leading to combustion instability.

These potential problems were alleviated with the addition of a second preburner and T/P. By introducing the second preburner/turbopump set, the deep throttling problems could be avoided by turning one set off and obtaining the 10:1 overall thrust reduction by throttling the remaining set down to only 5:1. The turned off preburner/turbopump set becomes backup components in this approach. In addition, the effective throttling required of the preburner/turbopump is reduced by half.

### 5.3.3 BASELINE CONCEPT

The baseline concept incorporates simple, low cost, innovative concept (SLIC) turbopumps for the main fuel pump. The Rocketdyne SLIC turbopump is a design breakthrough enabling a reduction in the component piece count while enhancing T/P performance. The SLIC design concept can also be used for the LOX pump. The low propellant NPSH requirements are met through the use of jet boost pumps. These simple design pumps provide the necessary NPSP without any moving parts.

The main combustion chambers incorporate conformal cooling channel geometry and ribs providing sufficient heat load to power the fuel T/P with minimal pressure drop in the cooling circuit. The nozzles are regeneratively cooled down to an area ratio of 308 and dump cooled with 4% of the hydrogen flow from there to an area ratio of 440 at the exit. The main chambers operate with gas-gas injection and readily throttle down in thrust without experiencing combustion instability. The LOX-rich preburners have a small amount of gaseous $H_2$ injection with the majority being liquid $O_2$. It was necessary to increase the pressure drop in the LOX injectors at full thrust in order to provide sufficient pressure drop for combustion stability at the minimum thrust.

Engine balances at on-design full thrust and off-design operating points down to the system minimum of 10% were generated. Propulsion system thrust vector control at full thrust can be accomplished using differential throttling controlled by EMA manifold valves. Off design operation for deep throttling (Lunar landing) requires more sophisticated control. A flow schematic of this system at nominal thrust is provided in Figure 5.3.3-1, and an engine balance printout in Figure 5.3.3-2. The hybrid power cycle attains a chamber pressure of 2288 psia and delivers a vacuum Isp of 478 sec with a nozzle expansion ratio of 440:1. Only half the system (one T/P and two T/C's at 40 Klbf) was modeled for simplicity, but the individual operating parameters are identical to the full system with double the components.

Closed loop control on thrust, engine mixture ratio and preburner temperature was incorporated for the throttled balances. A variable resistance two-way bypass valve was used in fuel turbine bypass for thrust control. A single valve in the bypass line did not provide sufficient shunting of flow around the turbine for the lowest thrust condition. Engine mixture ratio control was effected through the preburner fuel valve and preburner temperature was controlled with the preburner LOX valve. All valve resistance ratios were within acceptable ranges for all operating points.

Operating maps for both the main propellant pumps are presented in Figures 5.3.3-3 and 5.3.3-4 along with the operating points at full thrust, 2:1, and 5:1 throttling. The fuel pump operates with all points in the stable region to the right of the zero slope line. The LOX pump operates just to the left of this line in the potentially unstable region at the 5:1 throttled condition. This potential problem would be alleviated with a small fraction of pump recirculation, effectively moving the operating point to the right on the map and into the stable region.

Preburner injection pressure drops are presented as a function of thrust in Figure 5.3.3-5. A minimum of 5% of preburner Pc for LOX injection and 7.5% of preburner Pc for the $GH_2$ were set for combustion stability. It was necessary to set the LOX pressure drop at 22.3% of preburner Pc (790 psia) at nominal to fulfill this requirement at the minimum thrust level.

Preburner temperature and mixture ratio as functions of thrust are presented in Figures 5.3.3-6 and 5.3.3-7 respectively. It was necessary to reduce both the flow and temperature to the LOX turbine to sufficiently reduce the power at the low thrust condition. A summary of the preburner propellant inlet conditions and operating parameters at full thrust, 2:1, and 5:1 throttling are provided in Figure 5.3.3-8.

This effort represents a "first cut" at performance and control on an operationally efficient propulsion system meeting the requirements of a Lunar lander vehicle. Additional trade studies and system optimizations such as performance, control systems, weight, envelope, and payload versus nozzle expansion ratio are recommended. An "integrated" approach to these trade studies could present new opportunities to system simplification.

## 5.4 PROPULSION SYSTEM COMPARISON

The propulsion system comparison task would ultimately use an In-Space Operations Index (ISOI) developed in Task 4.2 as the basis for comparisons. The immaturity of the ISOI presently precludes this approach, however, a Launch

FIGURE 5.3.3-1

**STPOES - HYBRID CYCLE**

**2 T/P + 4 T/C* (Fvac=80000 klb Isp=478.1 sec)**

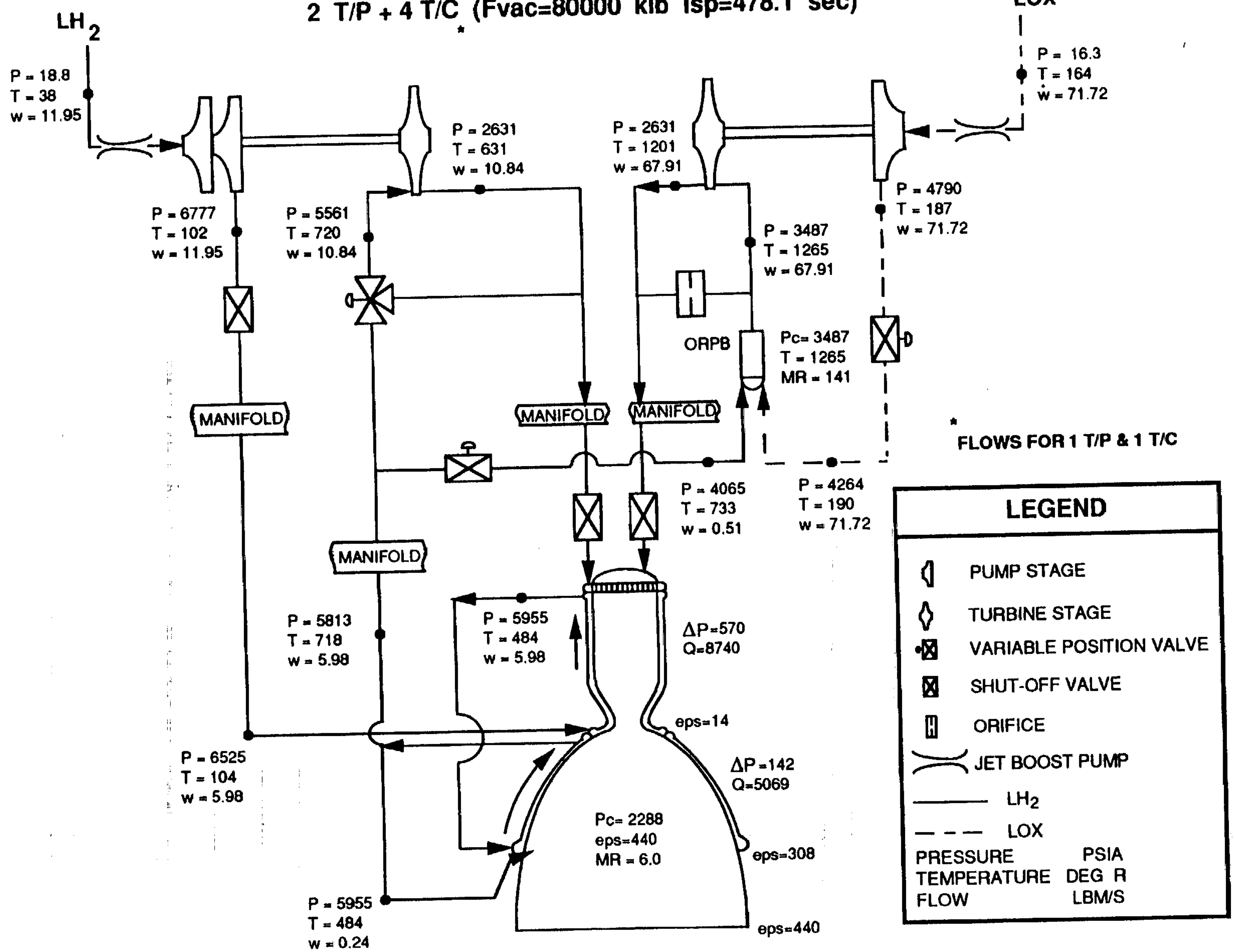


Figure 5.3.3-1

FIGURE 5.3.3-2 STPOES ENGINE BALANCE

```
*****************************************************************************************************
*                STEADY STATE ON-DESIGN ENGINE BALANCE    (PAGE NO. 1 OF 4)                        *
*                ON-DESIGN/OPTIMIZER CODE - EXPANDED OPTION Version 4.60                            *
*                              ( 9/ 2/92 - 16:22:10)                                                *
*****************************************************************************************************
*                                                                                                   *
*   CASE TITLE =    STPOES 80 K 4T/C-2T/P AS 40 K 1T/C-1T/P                                         *
*                                                                                                   *
*****************************************************************************************************
```

| ENGINE DESCRIPTION | (UNITS) | ENGINE | per T/C | dump cool |
|---|---|---|---|---|
| PROPELLANT COMBINATION | | LOX/H2 | | |
| ENGINE CYCLE | | HYBRID | | |
| MULTIPLE COMPONENTS CONFIGURATION | (YES/NO) | YES | | |
| NO. OF MULTIPLE COMPONENTS | | | 2 | |
| DELIVERED THRUST | | | | |
| VACUUM | (LBF) | 40000.00 | 20000.00 | |
| SEA LEVEL | (LBF) | -16538.41 | | |
| DELIVERED SPECIFIC IMPULSE | | | | |
| VACUUM | (SEC) | 478.06 | 478.87 | 337.0 |
| SEA LEVEL | (SEC) | -197.66 | | |
| MIXTURE RATIO BY WEIGHT | (O/F) | 6.000 | 6.250 | |
| MASS FLOWRATE | | | | |
| FUEL | (LB/SEC) | 11.95 | 5.74 | |
| OXIDIZER | (LB/SEC) | 71.72 | 35.86 | |
| DUMP COOLING | (LB/SEC) | | | 0.48 |
| GEOMETRIC ENGINE LENGTH | (IN) | 82.31 | | |
| GEOMETRIC ENGINE EXIT DIAMETER | (IN) | 49.48 | | |

| THRUST CHAMBER DESCRIPTION | (UNITS) | |
|---|---|---|
| NO. OF MULTIPLE THRUST CHAMBERS | (NONE) | 2 |
| CHAMBER PRESSURE | (PSIA) | 2288.18 |
| CHAMBER FACE PRESSURE | (PSIA) | 2288.18 |
| NOZZLE PERCENT LENGTH | (PERCENT) | 80.00 |
| FUEL INLET HEAT OF FORMATION | (KCAL/MOLE) | 0.12 |
| 1-DIMENSIONAL NOZZLE EXIT PRESSURE | (PSIA) | 0.20 |
| 2-DIMENSIONAL NOZZLE EXIT PRESSURE | (PSIA) | 0.41 |
| CONTRACTION RATIO | (NONE) | 4.00 |
| THROAT AREA DISCHARGE COEFFICIENT | (NONE) | 1.0000 |
| NOZZLE EXIT DISPLACEMENT THICKNESS | (INCHES) | 0.0000 |

| | | 1-DIM. AERO | GEOMETRIC |
|---|---|---|---|
| THROAT AREA | (IN**2) | 4.37 | 4.37 |
| NOZZLE EXIT DIAMETER - INSIDE DIAMETER | (IN) | | 49.48 |
| - PRIMARY | (IN) | 49.48 | 49.48 |
| NOZZLE AREA RATIO - INSIDE DIAMETER | (NONE) | | 440.00 |
| - PRIMARY | (NONE) | 440.00 | 440.00 |
| COMBUSTOR DIAMETER | (IN) | | 4.72 |
| THROAT DIAMETER | (IN) | 2.36 | 2.36 |
| THROAT RADIUS | (IN) | 1.18 | 1.18 |
| GIMBAL LENGTH | (IN) | | DNA |
| COMBUSTOR LENGTH | (IN) | | 11.96 |
| COMBUSTOR CHARACTERISTIC LENGTH | (IN) | | 44.31 |
| NOZZLE LENGTH | (IN) | | 70.35 |

| | | O.D.E. | DELIVERED |
|---|---|---|---|
| SPECIFIC IMPULSE | (SEC) | 497.54 | 478.87 |
| CHARACTERISTIC VELOCITY | (FT/SEC) | 7742.74 | 7704.23 |
| THRUST COEFFICIENT | (NONE) | 2.067 | 2.000 |
| ENERGY RELEASE EFFICIENCY (Isp) | (PERCENT) | 99.50 | |
| 1-DIM. KINETIC EFFICIENCY | (PERCENT) | 99.70 | |
| DIVERGENCE EFFICIENCY | (PERCENT) | 99.26 | |
| BOUNDARY LAYER EFFICIENCY | (PERCENT) | 97.78 | |

**Figure 5.3.3-2 STPOES Engine Balance**

```
*********************************************************************************************
*              STEADY STATE ON-DESIGN ENGINE BALANCE    (PAGE NO. 2 OF 4)                   *
*           ON-DESIGN/OPTIMIZER CODE - EXPANDED OPTION Version 4.60                          *
*                          ( 9/ 2/92 - 16:22:10)                                             *
*********************************************************************************************
*                                                                                           *
*  CASE TITLE =    STPOES 80 K 4T/C-2T/P AS 40 K 1T/C-1T/P                                  *
*                                                                                           *
*********************************************************************************************
```

| TURBOPUMP DESCRIPTION | (UNITS) | MAIN | PUMPS |
|---|---|---|---|
| NO. OF MULTIPLE COMPONENTS | (NONE) | 1. | 1. |
| PUMPING FLUID | (NONE) | HYDROGEN | OXYGEN |
| NO. OF STAGES | (NONE) | 2. | 1. |
| SHAFT SPEED | (RPM) | 218227.8 | 54236.2 |
| REQUIRED HORSEPOWER | (HP) | 5776.7 | 1700.1 |
| ISENTROPIC EFFICIENCY | (PERCENT) | 72.14 | 73.57 |
| INCOMPRESSIBLE EFFICIENCY | (PERCENT) | 77.45 | 73.57 |
| INLET PRESSURE | (PSIA) | 32.00 | 47.00 |
| DISCHARGE PRESSURE | (PSIA) | 6776.60 | 4789.93 |
| MASS FLOWRATE | (LB/SEC) | 11.95 | 71.72 |
| VOLUMETRIC FLOWRATE AT INLET | (GPM) | 1224.68 | 452.19 |
| INCOMPRESSIBLE HEAD RISE, OVERALL | (FT) | 206511.9 | 9622.8 |
| ISENTROPIC HEAD RISE, OVERALL | (FT) | 192348.7 | 9622.8 |
| PRESSURE RISE, OVERALL | (PSID) | 6744.60 | 4742.93 |
| STAGE SPECIFIC SPEED | (RPM*GPM**.5/FT**.75) | 1325.82 | 1187.06 |
| SUCTION SPECIFIC SPEED | (RPM*GPM**.5/FT**.75) | 18158.37 | 31327.12 |
| INLET/OUTLET DIAMETER RATIO | (NONE) | 0.600 | 0.616 |
| INDUCER TIP DIAMETER | (IN) | 1.50 | 2.05 |
| INDUCER TIP SPEED | (FT/SEC) | 1425.00 | 484.78 |
| INDUCER INLET FLOW VELOCITY | (FT/SEC) | 245.82 | 48.44 |
| INDUCER INLET FLOW COEFFICIENT | (NONE) | 0.1726 | 0.1000 |
| IMPELLER TIP DIAMETER | (IN) | 2.49 | 3.32 |
| IMPELLER TIP SPEED | (FT/SEC) | 2375.00 | 786.90 |
| IMPELLER TIP WIDTH | (IN) | 0.1917 | 0.2548 |
| IMPELLER FLOW COEFFICIENT | (NONE) | 0.2002 | 0.1259 |
| IMPELLER HEAD COEFFICIENT | (NONE) | 0.5890 | 0.5000 |
| IMPELLER BLADE ANGLE | (DEGREES) | 55.00 | 25.00 |
| INLET TEMPERATURE | (DEG R) | 38.00 | 164.00 |
| INLET DENSITY | (LB/FT**3) | 4.37 | 70.98 |
| INLET ENTHALPY | (BTU/LB) | -106.10 | -56.59 |
| INLET ENTROPY | (BTU/LB-R) | 2.00 | 0.71 |
| INLET VAPOR PRESSURE | (PSIA) | 18.66 | 16.16 |
| DISCHARGE TEMPERATURE | (DEG R) | 101.94 | 187.37 |
| DISCHARGE DENSITY | (LB/FT**3) | 4.98 | 71.60 |
| DISCHARGE ENTHALPY | (BTU/LB) | 236.62 | -39.78 |
| DISCHARGE ENTROPY | (BTU/LB-R) | 3.17 | 0.73 |

USER INPUT DESIGN LIMITS (IMPLICIT CONSTRAINTS)

| | | | |
|---|---|---|---|
| MAXIMUM INDUCER TIP SPEED | (FT/SEC) | 1425.00 | 600.00 |
| MAXIMUM IMPELLER TIP SPEED | (FT/SEC) | 2375.00 | 1000.00 |
| MAXIMUM IMPELLER HEAD COEFFICIENT | (NONE) | 0.6000 | 0.5000 |
| MAXIMUM INLET/OUTLET DIAMETER RATIO | (NONE) | 0.750 | 0.750 |
| MAXIMUM DISCHARGE PRESSURE | (PSIA) | 9000.00 | 9000.00 |
| MAX. STG. SPEC. SPEED | (RPM.GPM**.5/FT**.75) | 2500.00 | 1800.00 |
| MAX. SUC. SPEC. SPEED | (RPM.GPM**.5/FT**.75) | *********************** | |

USER INPUT LIMITS ON OPTIMIZED PARAMETERS (EXPLICIT CONSTRAINTS)

| | | | |
|---|---|---|---|
| MINIMUM SHAFT SPEED | (RPM) | 139000.00 | 28000.00 |
| MAXIMUM SHAFT SPEED | (RPM) | 280000.00 | 80000.00 |

**Figure 5.3.3-2 STPOES Engine Balance (Cont'd.)**

```
*****************************************************************************************
*                    STEADY STATE ON-DESIGN ENGINE BALANCE    (PAGE NO. 3 OF 4)          *
*                    ON-DESIGN/OPTIMIZER CODE - EXPANDED OPTION Version 4.60             *
*                                ( 9/ 2/92 - 16:22:10)                                   *
*****************************************************************************************
*                                                                                       *
*   CASE TITLE =    STPOES 80 K 4T/C-2T/P AS 40 K 1T/C-1T/P                              *
*                                                                                       *
*****************************************************************************************
```

| MAIN TURBINE DESCRIPTION | (UNITS) | FUEL | OXIDIZER |
|---|---|---|---|
| NO. OF MULTIPLE COMPONENTS | (NONE) | 1. | 1. |
| DRIVING GAS | (NONE) | HYDROGEN | LOX/H2 |
| TYPE | (NONE) | P/C | REAC. |
| NO. OF STAGES | (NONE) | 1. | 1. |
| SPEED | (RPM) | 218227.8 | 54236.2 |
| PRESSURE RATIO, INLET-TO-OUTLET | (NONE) | 2.113 | 1.325 |
| PITCH LINE (BLADE MEAN) VELOCITY | (FT/SEC) | 2114.00 | 885.48 |
| ADMISSION | (PERCENT) | 100.00 | 100.00 |
| DELIVERED HORSEPOWER | (HP) | 5776.7 | 1700.1 |
| ISENTROPIC EFFICIENCY | (PERCENT) | 68.80 | 78.89 |
| MASS FLOWRATE | (LB/SEC) | 10.84 | 67.91 |
| VELOCITY RATIO | (NONE) | 0.4036 | 0.8352 |
| PITCH LINE (BLADE MEAN) DIAMETER | (IN) | 2.218 | 3.739 |
| BEARING DN, E-6 | (MM*RPM) | 5.616 | 1.456 |
| ANNULUS AREA*N**2, E-10 | ((IN*RPM)**2) | 9.649 | 2.802 |
| HUB/TIP RATIO - INLET | (NONE) | 0.802 | 0.674 |
| HUB/TIP RATIO - EXIT | (NONE) | 0.768 | 0.643 |
| BLADE HEIGHT - INLET | (IN) | 0.244 | 0.728 |
| BLADE HEIGHT - EXIT | (IN) | 0.291 | 0.811 |
| MAX. BLADE TEMPERATURE - INLET | (DEG R) | 640.3 | 1227.8 |
| MAX. BLADE TEMPERATURE - EXIT | (DEG R) | 640.3 | 1227.8 |
| INLET TEMPERATURE | (DEG R) | 720.02 | 1265.00 |
| INLET PRESSURE | (PSIA) | 5561.20 | 3487.66 |
| DISCHARGE TEMPERATURE | (DEG R) | 631.49 | 1200.80 |
| DISCHARGE PRESSURE | (PSIA) | 2631.40 | 2631.40 |
| SPECIFIC HEAT | (BTU/LB-R) | 4.018 | 0.262 |
| SPECIFIC HEAT RATIO (GAMMA) | (NONE) | 1.390 | 1.331 |
| MOLECULAR WEIGHT | (LB/LB-MOLE) | 2.016 | 30.499 |
| **USER INPUT DESIGN LIMITS (IMPLICIT CONSTRAINTS)** | | | |
| MINIMUM HUB/TIP RATIO | (NONE) | 0.600 | 0.600 |
| MINIMUM BLADE HEIGTH | (IN) | 0.000 | 0.000 |
| MINIMUM PITCH LINE(BLADE MEAN) DIAMETER | (IN) | 0.000 | 0.000 |
| MAXIMUM HUB/TIP RATIO | (NONE) | 0.920 | 0.920 |
| MAXIMUM BLADE HEIGTH | (IN) | 100.000 | 100.000 |
| MAXIMUM PITCH LINE(BLADE MEAN) DIAMETER | (IN) | 100.000 | 100.000 |
| MAXIMUM AN**2, E-10 | ((IN*RPM)**2) | 9.650 | 9.250 |
| MAXIMUM BEARING DN, E-6 | (MM*RPM) | 1000.000 | 100.000 |
| **USER INPUT LIMITS ON OPTIMIZED PARAMETERS (EXPLICIT CONSTRAINTS)** | | | |
| MINIMUM PRESSURE RATIO | (NONE) | 1.80 | 1.33 |
| MINIMUM PITCH LINE VELOCITY | (FT/SEC) | 2114.00 | 880.00 |
| MAXIMUM PRESSURE RATIO | (NONE) | 2.20 | 1.33 |
| MAXIMUM PITCH LINE VELOCITY | (FT/SEC) | 2114.00 | 900.00 |

**Figure 5.3.3-2 STPOES Engine Balance (Cont'd.)**

```
*****************************************************************************
*            STEADY STATE ON-DESIGN ENGINE BALANCE    (PAGE NO. 4 OF 4)      *
*            ON-DESIGN/OPTIMIZER CODE - EXPANDED OPTION Version 4.60         *
*                        ( 9/ 2/92 - 16:22:10)                               *
*****************************************************************************
*                                                                           *
*  CASE TITLE =    STPOES 80 K 4T/C-2T/P AS 40 K 1T/C-1T/P                  *
*                                                                           *
*****************************************************************************
```

| ENGINE INLET DESCRIPTION | (UNITS) | FUEL | OXIDIZER |
|---|---|---|---|
| PROPELLANT TYPE | (NONE) | HYDROGEN | OXYGEN |
| INLET PRESSURE | (PSIA) | 32.00 | 47.00 |
| INLET TEMPERATURE | (DEG-R) | 38.00 | 164.00 |
| INLET DENSITY | (LB/FT**3) | 4.37 | 70.98 |
| INLET ENTHALPY | (BTU/LB) | -106.10 | -56.59 |
| INLET HEAT OF FORMATION | (KCAL/MOL) | -2.149 | -3.089 |
| MOLECULAR WEIGHT | (LB/LB-MOL) | 2.016 | 32.000 |

| COOLING JACKET DESCRIPTION | | NOZZLE | COMBUSTOR |
|---|---|---|---|
| NO. OF MULTIPLE COMPONENTS | (NONE) | 1. | 2. |
| COOLANT FLUID TYPE | (NONE) | HYDROGEN | HYDROGEN |
| COOLANT FLOWRATE | (LBS/SEC) | 5.98 | 5.98 |
| COOLING JACKET PRESSURE DROP | (PSID) | 142.00 | 570.00 |
| COOLING JACKET HEAT LOAD | (BTU/SEC) | 5069.00 | 8740.00 |
| NON-BOUNDARY LAYER HEAT LOSS | (BTU/SEC) | N/A | 1713.04 |
| INLET TEMPERATURE | (DEG-R) | 484.38 | 104.33 |
| INLET PRESSURE | (PSIA) | 5954.90 | 6524.90 |
| INLET ENTHALPY | (BTU/LB) | 1703.57 | 236.62 |
| DISCHARGE TEMPERATURE | (DEG-R) | 718.46 | 484.38 |
| DISCHARGE PRESSURE | (PSIA) | 5812.90 | 5954.90 |
| DISCHARGE ENTHALPY | (BTU/LB) | 2554.37 | 1703.57 |

| PREBURNER DESCRIPTION | | FUEL |
|---|---|---|
| NO. OF MULTIPLE COMPONENTS | (NONE) | 1 |
| PROPELLANT COMBINATION | (NONE) | LOX/H2 |
| CHAMBER PRESSURE | (PSIA) | 3487.66 |
| MIXTURE RATIO BY WEIGHT | (O/F) | 141.195 |
| CHAMBER TEMPERATURE | (DEG R) | 1265.00 |
| TOTAL GAS FLOWRATE | (LBS/SEC) | 72.23 |
| FUEL FLOWRATE | (LBS/SEC) | 0.51 |
| OXIDIZER FLOWRATE | (LBS/SEC) | 71.72 |
| FUEL INLET HEAT OF FORMATION | (KCAL/MOL) | 0.83 |

**Figure 5.3.3-2 STPOES Engine Balance (Cont'd.)**

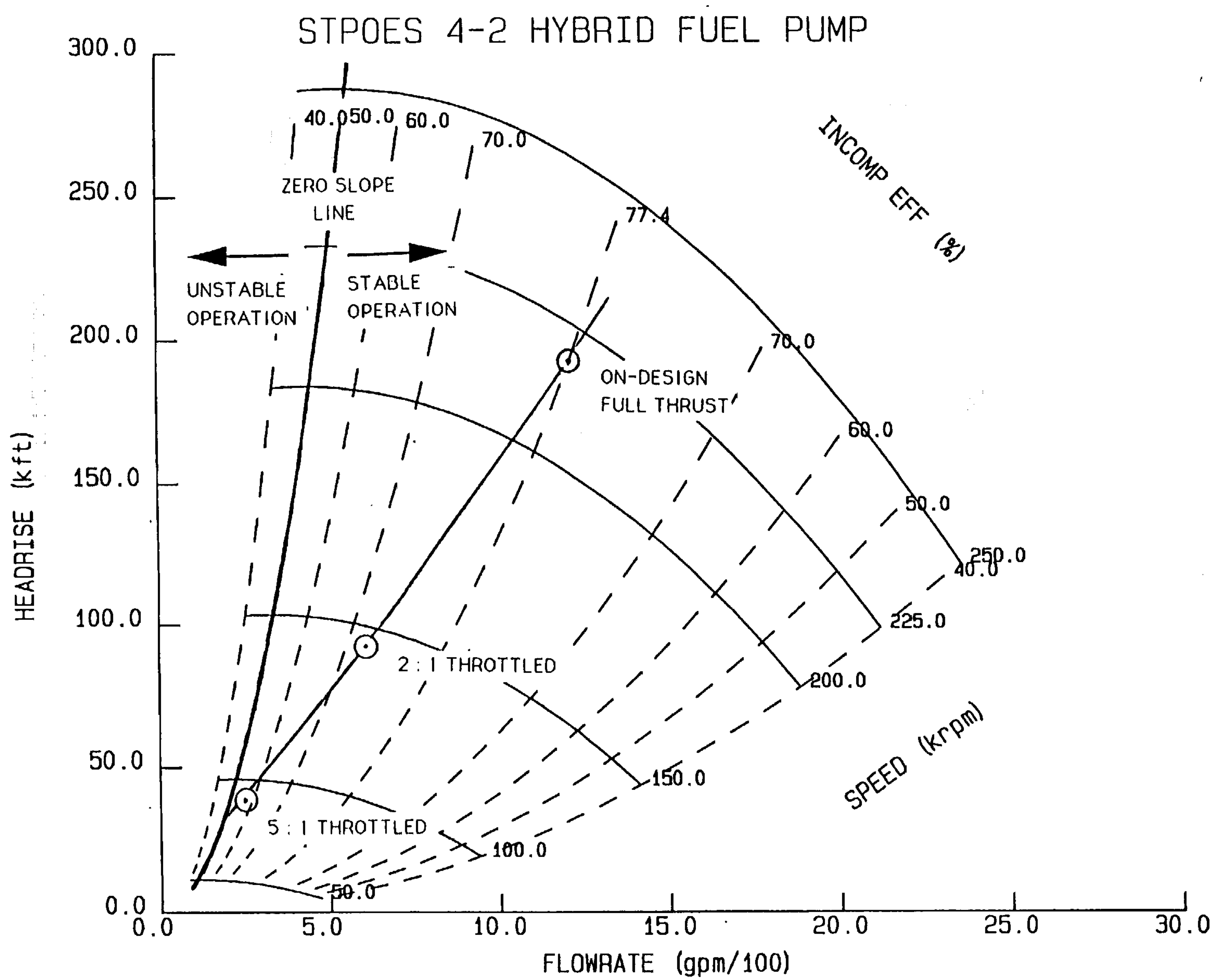
STPOES 4-2 HYBRID FUEL PUMP
INCOMP EFF (%)
40.0
50.0
60.0
70.0
77.4
70.0
60.0
50.0
40.0
ZERO SLOPE LINE
UNSTABLE OPERATION
STABLE OPERATION
ON-DESIGN FULL THRUST
2 : 1 THROTTLED
5 : 1 THROTTLED
SPEED (krpm)
250.0
225.0
200.0
150.0
100.0
50.0
HEADRISE (kft)
300.0
250.0
200.0
150.0
100.0
50.0
0.0
FLOWRATE (gpm/100)
0.0
5.0
10.0
15.0
20.0
25.0
30.0

**Figure 5.3.3-3**

## FIGURE 5.3.3-4

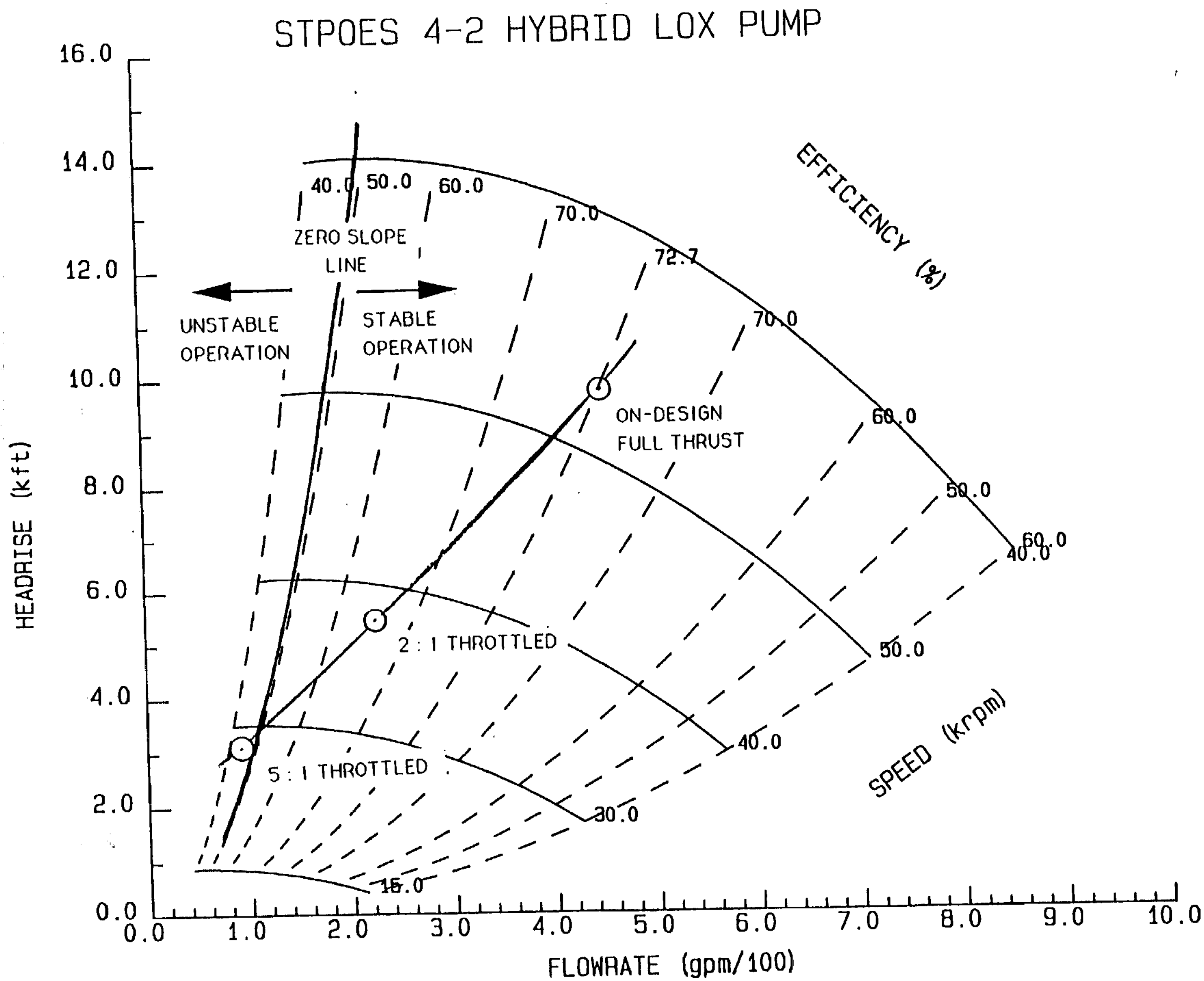


Figure 5.3.3-4

# STPOES 4 T/C - 2 T/P HYBRID CYCLE
# PREBURNER INJ PRES DROP vs THRUST

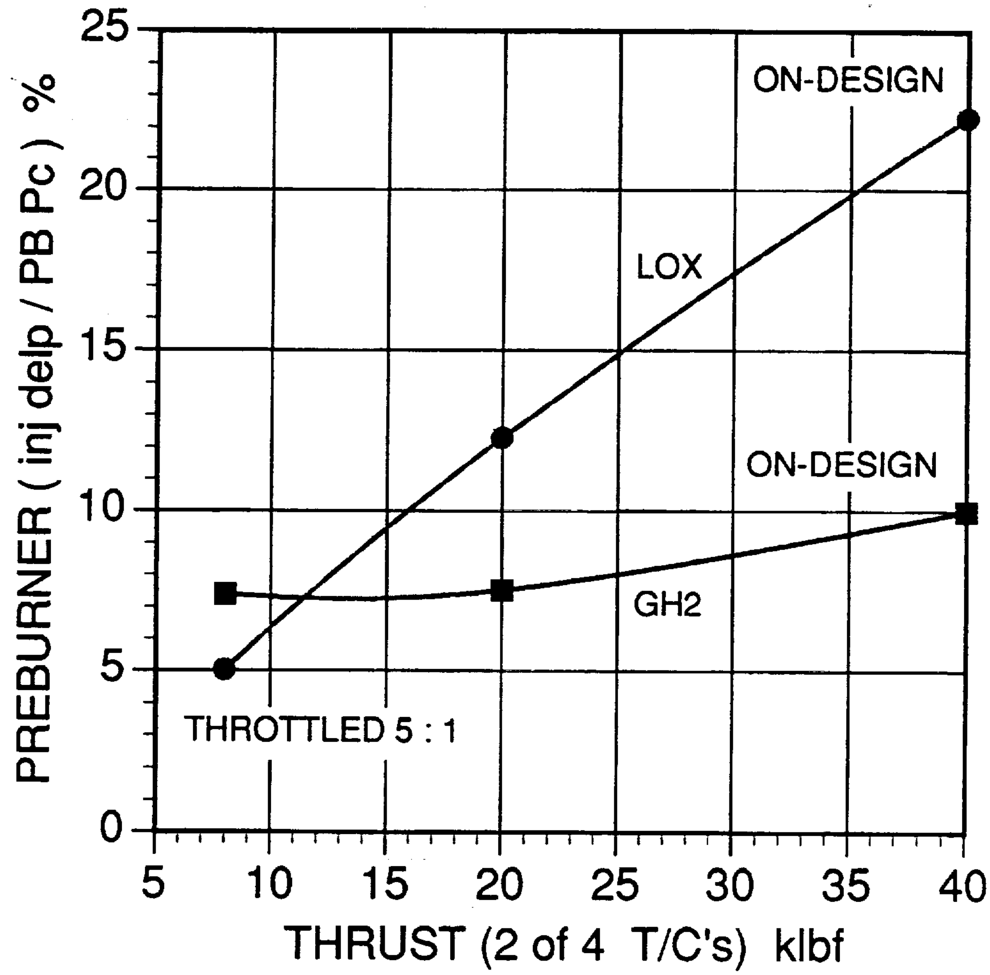


**Figure 5.3.3-5**

## STPOES 4 T/C - 2 T/P HYBRID CYCLE PREBURNER TEMPERATURE vs THRUST

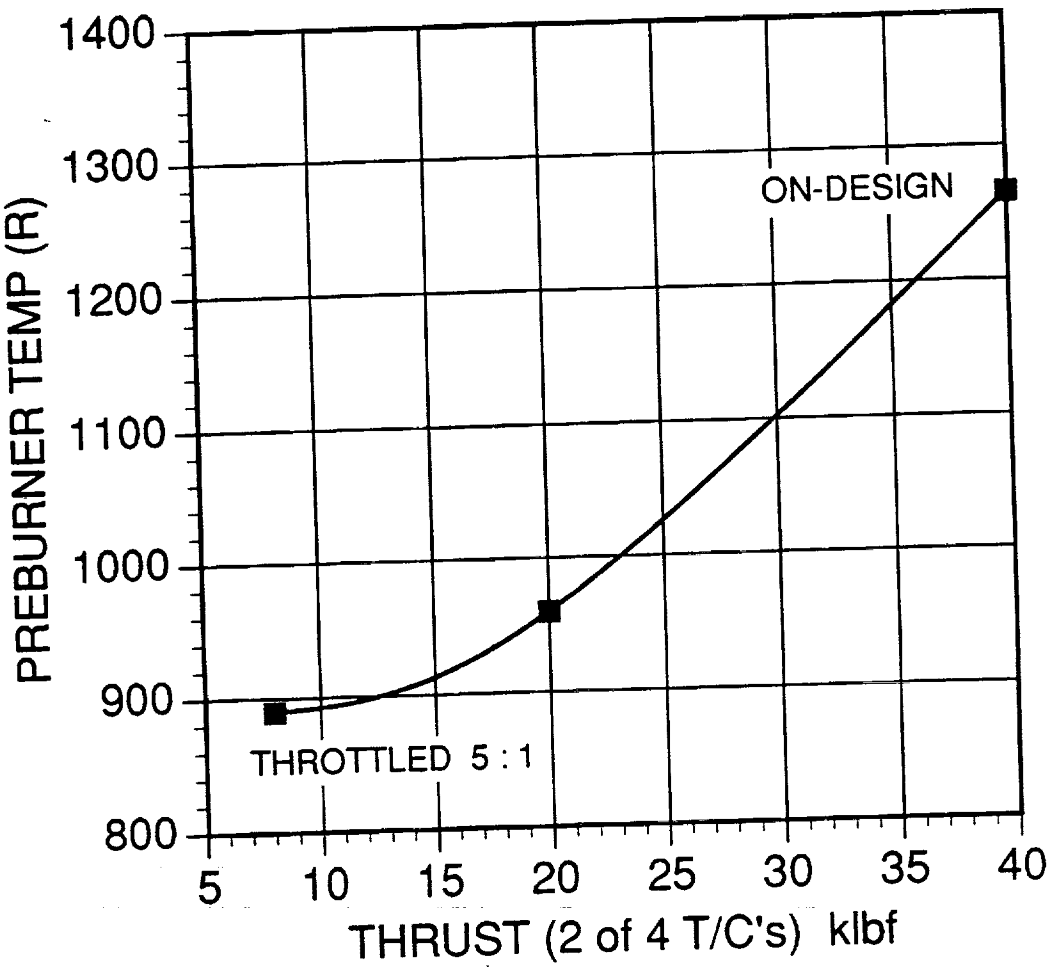


**Figure 5.3.3-6**

# STPOES 4 T/C - 2 T/P HYBRID CYCLE
# PREBURNER MIXTURE RATIO vs THRUST

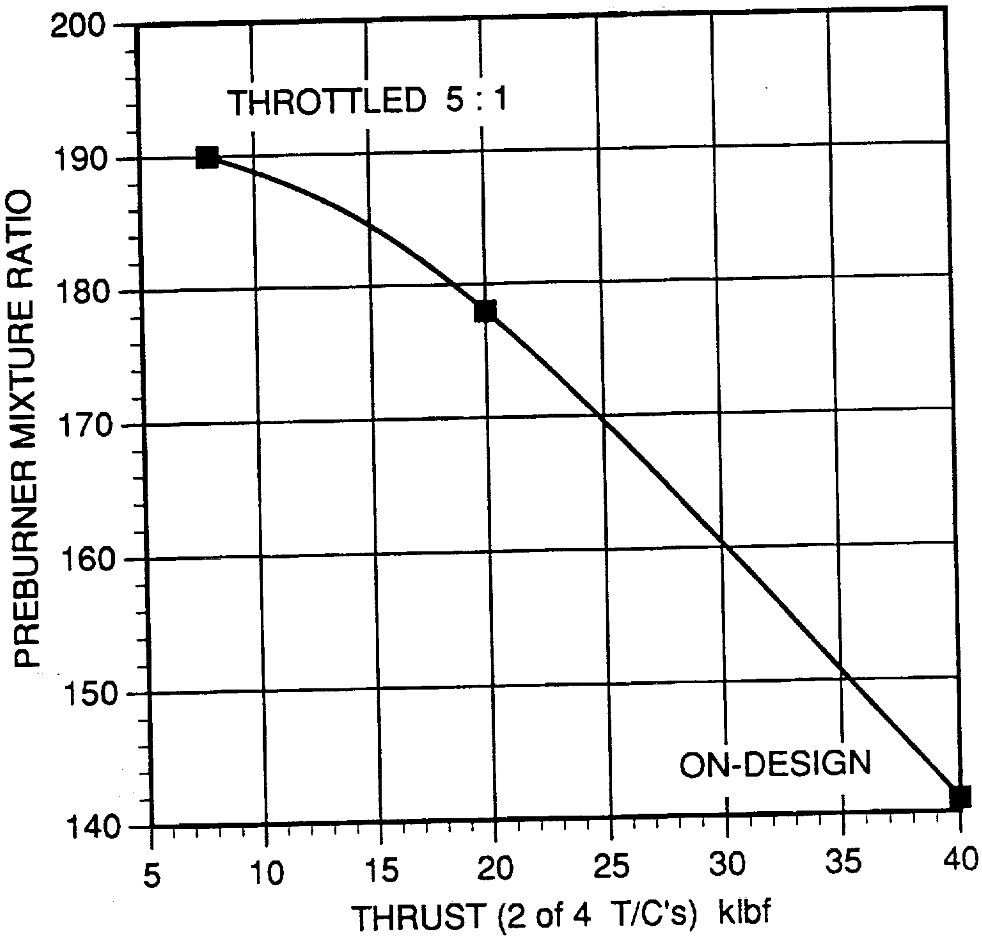


**Figure 5.3.3-7**

# STPOES 4 T/C - 2 T/P HYBRID CYCLE PREBURNER DURING THROTTLING

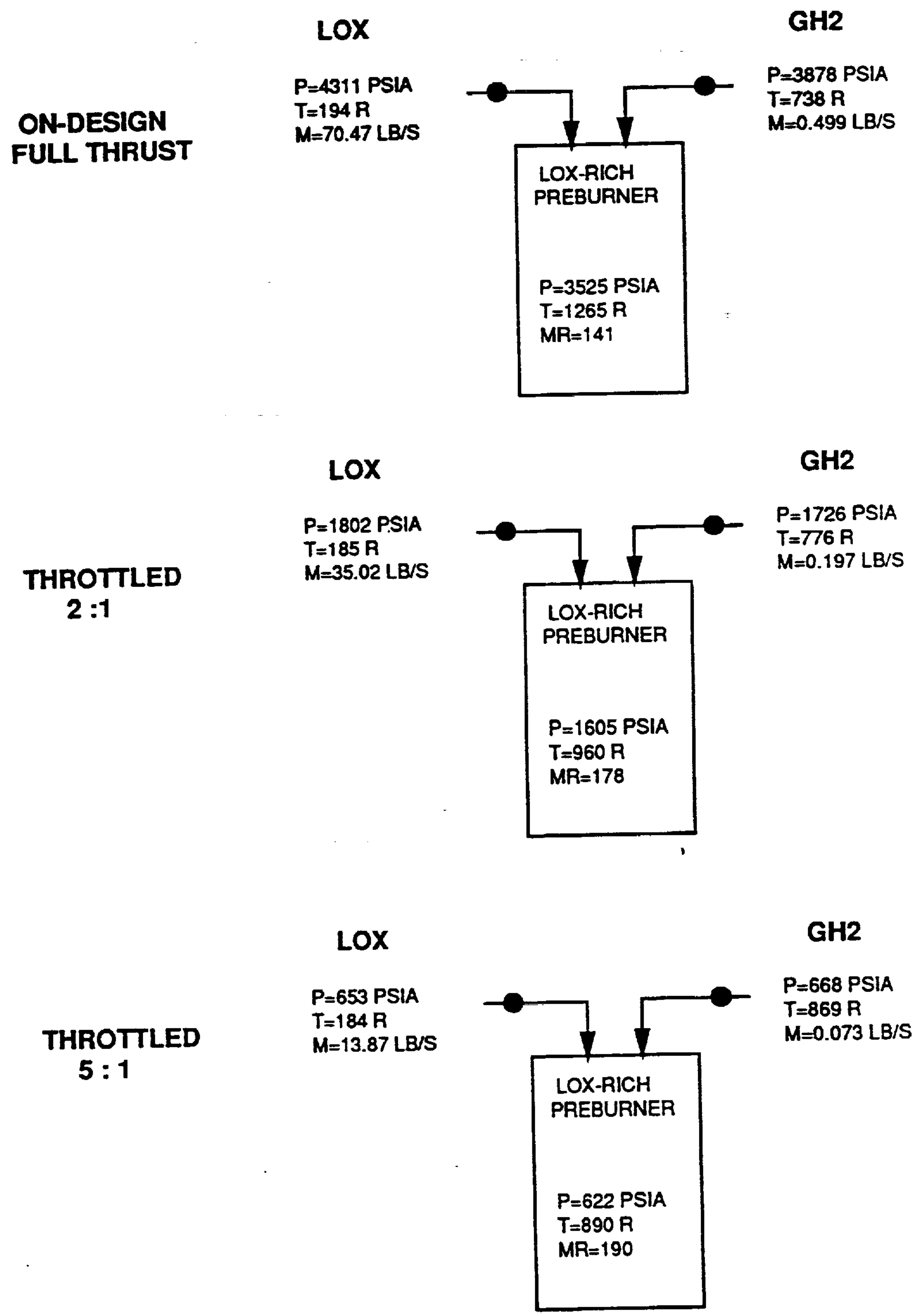


**Figure 5.3.3-8**

Operations Index comparison can be made. As all systems must be earth launched, use of the LOI has initial validity and should always be used in combination (LOI and ISOI when available).

The resources required to get a propulsion system ready to operate were broken down into a list of launch operations concerns. This concerns list reflects both nominal propulsion system preparation (no failure scenario) and launch experience where propulsion system maintenance was required. The concerns list, shown in Table 5.4-1, evolved from multiple workshops, interviews and documentation. From the concerns list a design features list was prepared which eliminates or mitigates specific concerns. The features list is shown in Table 5.4-2. There are a range of solutions for any design feature. For example, the Propulsion System Compartment Configuration design feature can range from a completely closed compartment with limited access to completely open. A propulsion system is compared against the features list and a relative value for each design feature is determined. The summation of design features make up an LOI for a particular design.

Propulsion system LOI comparisons were completed between four STPOES conceptual designs for a lunar lander propulsion system and the Centaur and S IV-B propulsion systems. These conceptual designs were all cryogenic lunar lander

## Table 5.4-1 STPOES Launch Operations Concerns

1 Closed aft compartments
2 Fluid system leakage
   - External
   - Internal
3 Hydraulic system
4 Multiple propellants
5 Hypergolic propellant (handling & safety)
6 Accessibility
7 Sophisticated heat shielding
8 Excessive components/subsystem interfaces
9 Hardware integration
10 Separate OMS and RCS
11 Pneumatic system
   - Actuation
   - Purging
   - Spin-up
   - Pressurization
12 Gimbal system
13 High maintenance hardware
14 Ordinance operations
15 Propellant tank pressurization systems
16 Excessive interfaces
17 Conditioning/Geysering (LOX tank forward)
18 Preconditioning system
19 Expensive commodity usage--helium
20 Hardware commonality
21 System contamination

# Table 5.4-2 Design Feature List

| | |
|---|---|
| 1 | Compartment Configuration |
| 2 | Degree of Checkout Automation |
| 3 | Number/Type of Propellants |
| 4 | Recovery Method |
| 5 | Auxiliary Propulsion type |
| 6 | Ordnance Systems |
| 7 | Actuator system Type |
| 8 | Heat Shield Type |
| 9 | Purge System Type |
| 10 | TVC System Type |
| 11 | Fluid Ground Interface Type |
| 12 | Tank Pressurization Systems |
| 13 | Preconditioning Reqt's |
| 14 | Accessibility |
| 15 | Potential for Leakage |
| 16 | Degree of Hardware Integration |
| 17 | Ground Support Requirements |

propulsion systems. The designs (described in Section 5.2) are delineated as follows: Concept A, An Operationally Efficient Technology design; Concept B, a variation of Concept A using supercritical propellants; Concept C, an enhanced design incorporating operationally efficient concentric toroidal propellant tanks; and Concept D, a variation of Concept C where the tanks are enlarged and the vehicle serves as both the descent and ascent stage. These concepts focus on incorporating operability features into a vehicle propulsion system. They are not definitive designs for a Lunar lander propulsion system.

The conceptual design comparisons for the Centaur, S IV-B and the four Lunar Lander conceptual designs are summarized in table 5.4-3. The Lunar Lander conceptual designs' LOI percentages were all in the low 80's. The STPOES designs focused on incorporating design features which increase propulsion system operability. This compares with LOI percentages in the mid 30's for the Centaur and S IV-B. Existing and previously designed propulsion systems were not designed with operability as the primary objective. The large gap in LOI reflects this difference in design objectives.

The four STPOES conceptual designs have similar LOI values. There are propulsion system differences with each design. Concept A incorporates operationally efficient design features within a design that, in appearance, are similar to systems shown in the First Lunar Outpost (FLO) workshop held at JSC on August 13-14, 1992. Concept B, using supercritical propellants, simplifies the vehicle systems further with fully integrated main propulsion, RCS, fuel cell, and crew oxygen. In addition, propellant acquisition and engine conditioning activities are eliminated. There is added weight with this concept, based on this admittedly incomplete first look. Concept C, using toroidal tanks, would simplify vehicle heat shielding as the outside diameter could be the vehicle skin. Concept D is a combined descent and ascent stage which would enhance operability by eliminating a complete vehicle stage.

The differences in the four conceptual designs, while significant, show only a small difference in the calculated LOI. For the case where a single stage or vehicle replaces two separate vehicles, i.e., Concepts C and D, the LOI calculation is deficient in differentiating operability improvements. This deficiency may be accounted for with multiple stages by multiplying separate LOI's so as to present a complete launch system (booster, upper stages, lander vehicles and ascent vehicles) integrated LOI.

Reliability is another comparison evaluation area. A preliminary reliability assessment for the STPOES concept design was completed on the four-thrust chamber/two-turbopump set configuration. The propulsion system has a turbopump-out capability (one turbopump set operating in a standby mode) during throttling. Reliability analysis of the STPOES concept predicts the design concept having an overall system reliability of 0.9941, exceeding the STPOES reliability goal of 0.99. Comparisons with other propulsion systems is beyond the scope of the task. Reliability model details are presented below.

The STPOES concept design is divided into four major subassemblies to simplify the reliability model analysis for the propulsion system. Each major subassembly is represented in the reliability block diagram (RBD) as the thrust chamber assembly

Table 5.4-3. In-Space Propulsion LOI Comparison

| | | Centaur | S IV-B | STPOES Lunar Lander | STPOES Lunar Lander | STPOES Lunar Lander | STPOES Lunar Lander |
|---|---|---|---|---|---|---|---|
| | Design Feature | value | value | value | dsgn B value | dsgn C value | dsgn D value |
| 1 | Compartment Configuration | 24 | 24 | 72 | 72 | 72 | 72 |
| 2 | Degree of Checkout Automati | 13.5 | 13.5 | 81 | 81 | 81 | 81 |
| 3 | Number/Type of Propellants | 17 | 15 | 40 | 40 | 40 | 40 |
| 4 | Recovery Method | 70 | 70 | 70 | 70 | 70 | 70 |
| 5 | Auxiliary Propulsion type | 16 | 16 | 56 | 68 | 56 | 56 |
| 6 | Ordnance Systems | 28 | 28 | 63 | 63 | 63 | 63 |
| 7 | Actuator system Type | 12 | 18 | 48 | 48 | 48 | 48 |
| 8 | Heat Shield Type | 42 | 42 | 42 | 42 | 48 | 48 |
| 9 | Purge System Type | 15 | 15 | 40 | 40 | 40 | 40 |
| 10 | TVC System Type | 15 | 15 | 50 | 50 | 50 | 50 |
| 11 | Fluid Ground Interface Type | 10 | 10 | 50 | 50 | 50 | 50 |
| 12 | Tank Pressurization Systems | 22 | 22 | 40 | 40 | 40 | 40 |
| 13 | Preconditioning Reqt's | 8 | 16 | 40 | 40 | 40 | 40 |
| 14 | Accessibility | 27 | 27 | 63 | 63 | 63 | 63 |
| 15 | Potential for Leakage | 24 | 24 | 56 | 56 | 56 | 56 |
| 16 | Degree of Hardware Integratic | 21 | 21 | 49 | 49 | 56 | 56 |
| 17 | Ground Support Requirement: | 21 | 21 | 63 | 63 | 63 | 63 |
| | Total | 385.5 | 397.5 | 923 | 935 | 936 | 936 |
| | LOI | 0.34 | 0.35 | 0.80 | 0.81 | 0.81 | 0.81 |

| | |
|---|---|
| **Concept A** | An Operationally Efficient Technology Design |
| **Concept B** | Variation of Concept A using Supercritical Propellants |
| **Concept C** | An Enhanced Design Incorporating Operationally Efficient Concentric Toridial Propellant Tanks |
| **Concept D** | Variation of Concept C Where Tanks are Enlarged and Vehicle Is Both Decent and Ascent Stage |

(TCA), the turbopump assembly (TMA), the control assembly, and the integrating assembly. Figure 5.4-1 illustrates the serial arrangement of the four major subassemblies including the serial arrangement of the four thrust chambers for the TCA and the two turbopump sets in the standby mode for the TMA. Each turbopump set within the TMA is arranged in series with the preburner assembly (PBA).

The TCA includes the four sets of injectors, combustion chambers, nozzles, and the fuel and oxidizer throttling valves. The TMA includes the two sets of the integrated fuel boost/SLIC pumps, integrated oxidizer boost/SLIC pumps, fuel pump isolation valve, fuel turbine bypass isolation valve, fuel turbine isolation valve, and oxidizer turbine isolation valve. The control assembly includes the controller, engine sensors, and the health monitoring system. The integrating assembly includes the four manifolds (i.e., high pressure fuel, fuel turbine inlet and outlet, oxidizer outlet), seven associated fuel ducts (i.e., high pressure fuel pump discharge, thrust chamber coolant, fuel turbine inlet and outlet, fuel turbine bypass loop, fuel injector), and six associated oxidizer ducts (i.e., oxidizer inlet, oxidizer turbine inlet and outlet, oxidizer turbine bypass loop, oxidizer injector).

The STPOES reliability prediction, shown in Table 5.4-4, is based on separate studies done by R. Biggs and F. M. Kirby of Rocketdyne. The SSME reliability values are used as the baseline for assigning reliability values to the STPOES components. These predicted SSME reliability values are identified in the AIAA 90-2712 report titled "A Probabilistic Risk Assessment for the Space Shuttle Main Engine with a Turbomachinery Vibration Monitor Cutoff System" by R. Biggs. The factors used to scale the SSME reliability values are based on the system complexity, technological risk, operating environmental conditions, and hardware producibility elements of the STPOES design. The STPOES/SSME factors used to scale for the STPOES concept are shown in Table 5.4-5. These factors are based on the RI/RD 89-136 report titled "Design Definition Document for the Space Transportation Main Engine" by F. M. Kirby.

The reliability allocation for the STPOES engine components is determined by the parts counts method. The reliability goal of 0.99 was specified for the propulsion system. Note, this reliability assessment was conducted assuming a traditional engine system. Indeed, the reliability goal (0.99) is a traditional engine system allocation. While reliability benefits are accounted for using an integrated engine system design, the total propulsion system defined by this study (tanks, lines, RCS system, turbopumps, thrust chambers, etc.) was not evaluated.

This reliability assessment was beyond the study scope. The results of the reliability allocation analysis are shown in Table 5.4-6 and are organized in a similar arrangement to the R. Biggs report. The reliability allocation is divided into five groups including the fuel turbopump (FTP) system, oxidizer turbopump (OTP) system, thrust chamber assembly (TCA), control system assembly and ducts/lines. The FTP and OTP systems include the two integrated boost/SLIC pumps and two integrated boost/SLIC pumps, respectively. The TCA includes the two preburners, and four sets of injectors, combustion chambers, and nozzles. The control system assembly includes the valves (i.e., 20 valves mentioned earlier), controller, and sensors (i.e., sensors, health monitoring system). The ducts/lines include the four manifolds and 13 propellant ducts mentioned earlier.

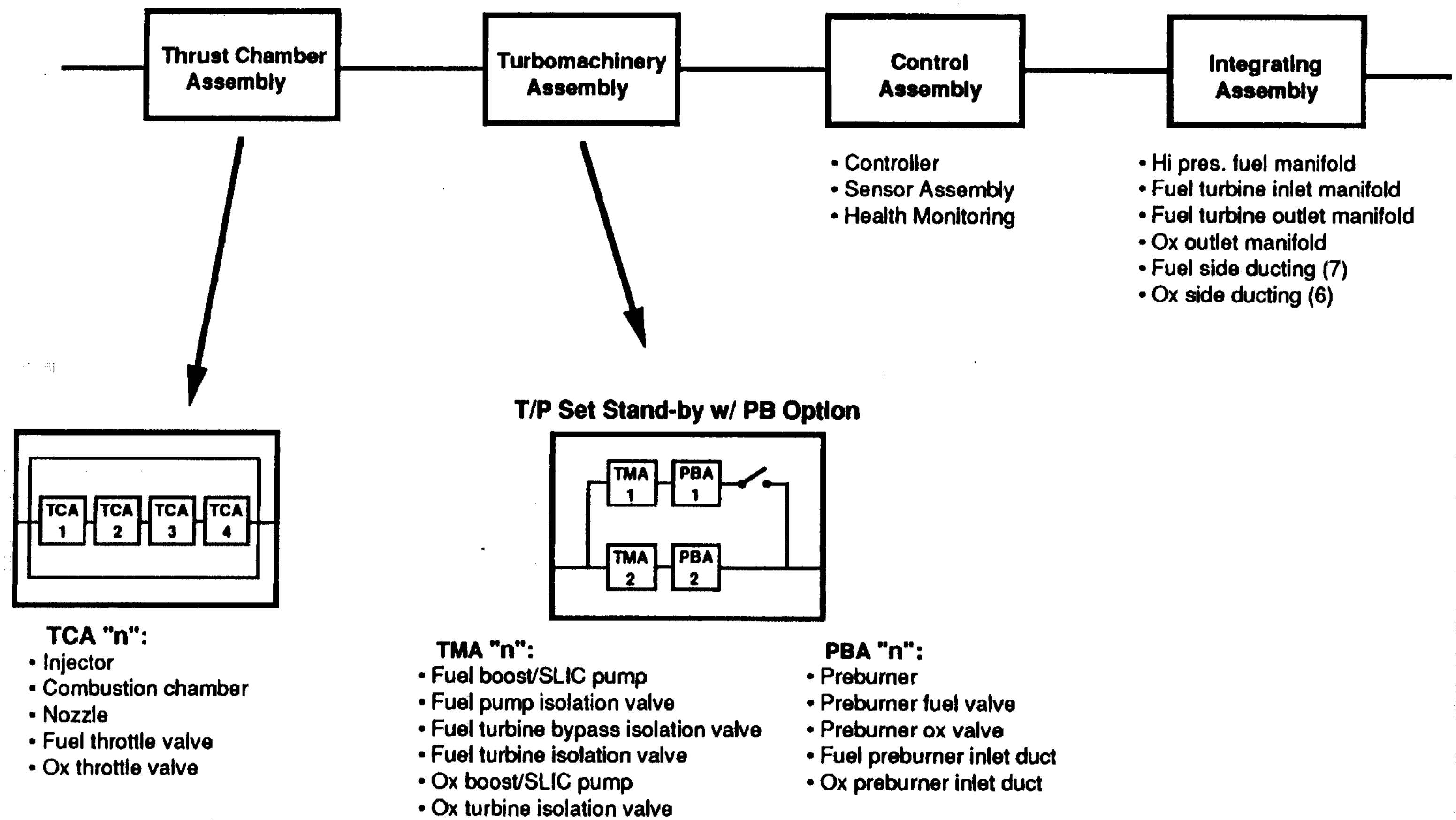


**Figure 5.4-1: Space Transfer Propulsion Operational Efficiency Study (STPOES) Reliability Block Diagram (RBD)**

STPOES RBD.5 Apr 92

# Table 5.4-4 STPOES Reliability Prediction

| 9-Oct-92 | SSME Analogy | SSME Rel. | Failure Rate (10^3) | Weighed Factor (Reduce F.R. by) | STPOES Reliability (Adjusted) | Failure Rate (10^3) (Adjusted) | Configuration: / Thrust vectoring: | STPOES 4 T/C & 2 T/P no T/C out diff throttle (Stand-by) (Adjusted) | STPOES # of Units 4 T/C & 2 T/P configuration |
|---|---|---|---|---|---|---|---|---|---|
| **Thrust Chamber Assembly** | | | | | | | | | |
| Injector | main injector | 0.998653 | 1.35 | 0.85 | .999798 | 0.20 | | | 4 |
| Combustion chamber | MCC | 0.997382 | 2.62 | 0.80 | .999476 | 0.52 | | | 4 |
| Nozzle | nozzle | 0.997382 | 2.62 | 0.85 | .999607 | 0.39 | | | 4 |
| Fuel throttle valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 4 |
| Oxidizer throttle valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 4 |
| | | | | | | | Per TCA: | 0.998849 | |
| | | | | | | | Total TCA: | 0.995404 | |
| **Turbomachinery Assembly** | | | | | | | | | |
| Fuel boost pump | LPFTP | 0.999496 | 0.50 | 0.70 | .999849 | 0.15 | | | 2 |
| Fuel SLIC pump | HPFTP | 0.995689 | 4.31 | 0.70 | .998707 | 1.29 | | | 2 |
| Fuel pump isolation valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| Fuel turbine by-pass isolation valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| Fuel turbine isolation valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| Ox boost pump | HPOTP | 0.997806 | 2.19 | 0.35 | .998574 | 1.43 | | | 2 |
| Ox SLIC pump | LPOTP | 0.999853 | 0.15 | 0.35 | .999904 | 0.10 | | | 2 |
| Ox turbine isolation valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| | | | | | | | Per TMA: | 0.996511 | |
| | | | | | | | Total TMA: | 0.999994 | |
| **Controls** | | | | | | | | | |
| Controller | electronics | 0.999538 | 0.46 | 0.75 | .999885 | 0.12 | | | 1 |
| Sensor | sensor | 0.997806 | 2.19 | 0.75 | .999452 | 0.55 | | | TBD |
| Health Monitoring | electronics | 0.999538 | 0.46 | 0.75 | .999885 | 0.12 | | | 1 |
| | | | | | | | Total CA: | 0.999221 | |
| **Integrating valves, ducts, and manifolds** | | | | | | | | | |
| High pressure fuel manifold | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel turbine inlet manifold | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel turbine outlet manifold | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox outlet manifold | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| (Fuel) Fuel inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| High pres. fuel pump discharge duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Thrust chamber coolant duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel turbine inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel turbine outlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel turbine bypass loop duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Fuel injector duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| (Oxidizer) Oxidizer inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox pump discharge duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox turbine inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox turbine outlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox turbine bypass loop duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| Ox injector duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 1 |
| | | | | | | | Total IA: | 0.999507 | |
| **Hybrid-related components** | | | | | | | | | |
| Preburner | preburner | 0.999076 | 0.92 | 0.60 | .999630 | 0.37 | | | 2 |
| Preburner fuel valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| Preburner ox valve | prop. control | 0.999890 | 0.11 | 0.85 | .999984 | 0.02 | | | 2 |
| Fuel preburner inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 2 |
| Ox preburner inlet duct | ducting | 0.999855 | 0.14 | 0.80 | .999971 | 0.03 | | | 2 |
| | | 0.978789 | | | | | Overall R: | 0.994132 | |
| | | | | | | | Overall FR (10^3): | 5.87 | |

## Table 5.4-5a: STPOES DESIGN RELIABILITY PREDICTION THRUST CHAMBER ASSEMBLY

| Similar Components STPOES/ SSME | Design Improvements STPOES/SSME | Design Improvement Factor | | | | Predicted Failure Rate Improvement | Failure Rate Comparison | |
|---|---|---|---|---|---|---|---|---|
| | | Producibility | Operating Environ/ Stress | Technological Risk | Complexity | | SSME Actual | STPOES Predicted |
| MCC | • Significantly reduced welding<br>• Welds easily inspectable<br>• Reduced number of parts<br>• Investment casting CC body<br>• Increased structural margin<br>• Improved heat flux cooling margin | 5 | 4 | 3 | 4 | 80% | 2.62 | 0.52 |
| Nozzle | • Reduced number of welds<br>• Welds easily inspectable<br>• Elimination of coolant tubes<br>• Cast manifolds<br>• Film cooling<br>• Increased structural margin<br>• Partial regenerative cooling | 5 | 5 | 4 | 3 | 85% | 2.62 | 0.39 |
| Main Injector | • Elimination of welds<br>• Laser drilled orifices<br>• Elimination of coaxial injector elements<br>• Increased structural margin<br>• More benign operating environ. | 5 | 4 | 4 | 4 | 85% | 1.35 | 0.20 |
| Gas Generator/ Preburner | • Reduced welding<br>• Welds easily inspectable<br>• Cast manifolds & body<br>• No combustor liner/simpler design<br>• Increased structural margin | 5 | 1 | 2 | 4 | 60% | 0.92 | 0.37 |

SSME Weighing Factors
DYL. 9 Oct 92

## Table 5.4-5b: STPOES DESIGN RELIABILITY PREDICTION TURBOMACHINERY ASSEMBLY

| Similar Components STPOES/ SSME | Design Improvements STPOES/SSME | Design Improvement Factor | | | | Predicted Failure Rate Improvement | Failure Rate Comparison | |
|---|---|---|---|---|---|---|---|---|
| | | Producibility | Operating Environ/ Stress | Technological Risk | Complexity | | SSME Actual | STPOES Predicted |
| LOX Turbopump | • SLIC design<br>• Elimination of welds<br>• Elimination of seal package<br>• Significantly reduced number of parts<br>• Hydrostatic bearings<br>• Oxidizer on pump & turbine ends<br>• Simplified assembly process<br>• Investment cast housings | 4 | -1 | -1 | 5 | 35% | 2.19 | 1.43 |
| Fuel Turbopump | • SLIC design<br>• Elimination of welds<br>• Elimination of seal package<br>• Significantly reduced number of parts<br>• Hydrostatic bearings<br>• Fuel on pump & turbine ends<br>• Simplified assembly process<br>• Investment cast housings<br>• Reduced turbine temperatures | 4 | 3 | 2 | 5 | 70% | 4.31 | 1.29 |

SSME Weighing Factors
DYL 9 Oct 92

# Table 5.4-5c: STPOES DESIGN RELIABILITY PREDICTION ENGINE SYSTEM

| Similar Components STPOES/ SSME | Design Improvements STPOES/SSME | Design Improvement Factor | | | | Predicted Failure Rate Improvement | Failure Rate Comparison | |
|---|---|---|---|---|---|---|---|---|
| | | Producibility | Operating Environ/ Stress | Technological Risk | Complexity | | SSME Actual | STPOES Predicted |
| Electronics/ Controls | • Closed loop performance control<br>• Fail safe system tolerance to single point failure<br>• Closed loop redline control<br>• Condition monitoring<br>• Minimum checkout<br>• Redundant redline measurements<br>• Micro controller shock mounted | 5 | 3 | 3 | 4 | 75% | 0.46 | 0.12 |
| Valves | • Pneumatic actuated sector ball valves<br>• Large force margin<br>• Plastic bushings<br>• Open/closed position indication<br>• Fail safe closing return spring<br>• Use of check valves where applicable | 4 | 4 | 4 | 5 | 85% | 0.16 | 0.02 |
| Ducting | • Increased Fs margin<br>• Reduced welding<br>• Reduced flow velocity<br>• Simplified flange geometry for vehicle inner connects<br>• Reduced number of parts | 4 | 4 | 4 | 4 | 80% | 0.34 | 0.07 |

SSME Weighing Factors
DYL 9 Oct 92

# Table 5.4-6 STPOES Allocation

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| dtd: 9 Oct 92 | | Overall System Rel: | 0.990 | | | | | |
| **STME Allocation** | **%** | **STPOES Components** | **# Units** | **%** | **Relative %** | **Fail. Rate** (1000 cycles) | **Reliability** (subsystem) | **Reliability** (component) |
| **FTP** | **0.280** | **FTP System** | 2 | **0.280** | **0.285** | **2.85** | **.997154** | |
| | | Fuel SLIC pump w/ boost pump | 1 | | | 1.42 | .998576 | .998576 |
| **OTP** | **0.170** | **OTP System** | 2 | **0.170** | **0.173** | **1.73** | **.998271** | |
| | | Ox SLIC pump w/ boost pump | 1 | | | 0.86 | .999135 | .999135 |
| **Thrust Chamber Assy** | **0.330** | **TCA** | 14 | **0.330** | **0.336** | **3.36** | **.996647** | |
| Preburner | 0.070 | Preburner | 2 | 0.070 | 0.071 | 0.48 | .999520 | .999760 |
| Injector | 0.030 | Micro Orifice Injector | 4 | 0.030 | 0.031 | 0.96 | .999041 | .999760 |
| Combustion Chamber | 0.090 | Thrust Chamber | 4 | 0.090 | 0.092 | 0.96 | .999041 | .999760 |
| Nozzle | 0.140 | Nozzle | 4 | 0.140 | 0.142 | 0.96 | .999041 | .999760 |
| **Control System** | **0.120** | **Control System** | 82 | **0.120** | **0.122** | **1.22** | **.998780** | |
| Valves/Actuators | 0.090 | Valves/Actuators | 20 | 0.090 | 0.092 | 0.30 | .999702 | .999985 |
| | | T/C oxidizer throttle valve | 4 | | | 0.06 | .999940 | .999985 |
| | | T/C fuel throttle valve | 4 | | | 0.06 | .999940 | .999985 |
| | | Fuel pump isolation valve | 2 | | | 0.03 | .999970 | .999985 |
| | | Fuel turbine bypass isolation valve | 2 | | | 0.03 | .999970 | .999985 |
| | | Fuel turbine isolation valve | 2 | | | 0.03 | .999970 | .999985 |
| | | Ox turbine isolation valve | 2 | | | 0.03 | .999970 | .999985 |
| | | Preburner fuel valve | 2 | | | 0.03 | .999970 | .999985 |
| | | Preburner oxidizer valve | 2 | | | 0.03 | .999970 | .999985 |
| Controller | 0.010 | Controller | 1 | 0.010 | 0.010 | 0.01 | .999985 | .999985 |
| Sensors/Cables | 0.020 | Sensors/Cables | 61 | 0.020 | 0.020 | 0.91 | .999092 | .999985 |
| | | Sensors | 60 | | | 0.89 | .999107 | .999985 |
| | | Health monitoring | 1 | | | 0.01 | .999985 | .999985 |
| **Ducts/Lines** | **0.083** | **Ducts/Lines** | 21 | **0.083** | **0.084** | **0.84** | **.999156** | |
| | | High pressure fuel manifold | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel turbine inlet manifold | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel turbine outlet manifold | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox outlet manifold | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel inlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | High pres. fuel pump discharge duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Thrust chamber coolant duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel turbine inlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel turbine outlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel turbine bypass loop duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel injector duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Oxidizer inlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox pump discharge duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox turbine inlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox turbine outlet duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox turbine bypass loop duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Ox injector duct | 1 | | | 0.04 | .999960 | .999960 |
| | | Fuel preburner inlet duct | 2 | | | 0.08 | .999920 | .999960 |
| | | Ox preburner inlet duct | 2 | | | 0.08 | .999920 | .999960 |
| **Others** | **0.017** | **Others** | 0 | **0.000** | **0.000** | **0.00** | **#NUM!** | |
| Gimbal | 0.008 | Gimbal | 0 | 0.000 | 0.000 | #NUM! | #NUM! | #NUM! |
| | | Hydraulic actuators | 0 | | | #NUM! | #NUM! | #NUM! |
| | | Hydraulic power | 0 | | | #NUM! | #NUM! | #NUM! |
| Pneum Control | 0.006 | Pneum Control | 0 | 0.000 | 0.000 | #NUM! | #NUM! | #NUM! |
| HEX | 0.003 | HEX | 0 | 0.000 | 0.000 | | | |
| **Total:** | 1.000 | | | 0.983 | 1.000 | 10.00 | **.99005** | .99005 |

# 6.0 TECHNOLOGIES

The approach to identifying operational efficient technologies considers the STPOES task results, space transfer propulsion system concept designs, the mission, and other factors. The other factors come from examining related programs: the OEPSS, IME (Air Force Headquarters Space Systems Division Contract FO4701-91-C-0076) and Centaur upper stage, for operational concerns.

## 6.1 OPERATIONAL EFFICIENT TECHNOLOGIES CANDIDATES

Candidate technologies are those technologies which resolve or mitigate operational concerns. Again for clarity, in this STPOES task the propulsion system definition includes the vehicle tanks, lines, RCS system, turbopumps, thrust chambers etc. The STPOES task propulsion system architectures address many of the launch and in-space operations concerns. However, the STPOES conceptual designs are not developed systems. Technology development in several areas is necessary to bring these concepts to fruition.

System simplification eliminates many concerns, for example, gimbaling and hydraulic systems were eliminated by incorporating differential throttling for Thrust Vector Control (TVC) and Electromechanical Actuators (EMA's) for hydraulic actuators. The in-space environment allows for open propulsion system compartments eliminating closed aft compartments and enclosed systems.

The focus on propulsion system simplification guided the recommended technologies choices. The goal of simplifying the propulsion system by integrating the OMS and RCS propulsion systems, eliminating purges, and providing electric power generation gases (fuel cell power) drove the design towards using gaseous hydrogen and oxygen as the main propulsion system propellants. The selected power cycle, a hybrid cycle, uses a hydrogen expander cycle to drive the fuel turbopump and an oxidizer rich preburner to drive the oxidizer turbopump. This cycle provides gaseous hydrogen and oxygen. Turbine drives are simplified because mixing of fuel and oxidizer propellants are eliminated. In addition, the lunar lander requirement of 10:1 throttling can best be achieved using gaseous propellants. The technology requiring development for the hybrid cycle is the oxidizer rich preburner.

Design simplification within the propulsion system suggested a complex component, the turbopumps, be reassessed. Again emerging technologies have shown that major simplification are possible with turbopump design. This simplified turbopump technology has been named the $SLIC_{TM}$ turbopump. In this same area the need for tank pressurization systems, to supply turbopump NPSH, adds to system complexity. A turbopump system capable of operating with zero NPSH would simplify propulsion tank design requirements. Adding a simple boost pump to the turbopump was chosen as a third technology for development.

If one considers the propulsion system as defined by the STPOES, then a systems test bed that includes propellant tanks and the RCS system etc., becomes in itself an operational efficient technology candidate. This very ambitious approach to developing operationally efficient technologies within a system environment is badly needed. This kind of test bed system, with its focus on operations, would rapidly bring promising technologies to an accepted level of maturation.

The resultant selected technologies for development are: the oxidizer-rich preburner, the $SLIC_{TM}$ turbopump, the jet boost pump/$SLIC_{TM}$ module, and the propulsion system test bed. This is not a complete list of operationally efficient technologies that need to be developed. The above recommendations evolved from this task results. However the propulsion system test bed would serve as an excellent vehicle for validating other operationally efficient technologies.

### 6.1.1 OXIDIZER-RICH PREBURNER

When the turbopump turbine is driven with the same fluid which it pumps, concerns about mixing oxidizer and fuel propellants within the unit are eliminated. This allows the fuel and oxidizer turbopumps to be designed and operated without intermediate seal purge requirements. The baselined hybrid cycle integrated propulsion module uses hydrogen thrust chamber coolant tap-off gas to drive the fuel turbopump and oxidizer-rich preburner to supply oxidizer-rich turbine drive gas. The need for oxidizer pump oxidizer-rich turbine drive gas necessitates development of oxidizer-rich preburner technology.

An oxidizer-rich preburner further enables a wider engine system operating range as the injector operates as a gas-gas (fuel and oxidizer) system. Deep throttling, on the order of 10:1 can be obtained. In addition, sources of gaseous hydrogen and oxygen are available for the RCS systems, and electric power generation (fuel cells) all of which enhance operating efficiency by simplifying the overall propulsion system.

### 6.1.2 $SLIC_{TM}$ TURBOPUMP

Liquid propulsion rocket engine turbomachinery is the integral/critical component for providing a high thrust-to-weight ratio propulsion system. Typically, turbomachinery is expensive and complex, with a large number of parts, elaborate seals, mechanical bearings, and, in some cases, gear trains. These attributes cause typical turbo-machinery to be the source of both reliability and operability concerns. A space-based transfer propulsion system needs both minimal operability demands and maximum reliability. The $SLIC_{TM}$ pump concept addresses these needs by featuring: minimum number of parts, hydrostatic bearings, simplified construction, and reduced cost to improve operability by enhancing dependability and eliminating the need for pretest manual checkouts and unplanned maintenance and inspection.

### 6.1.3 JET BOOST PUMP /$SLIC_{TM}$ TURBOPUMP MODULE

The integrated jet boost pump/$SLIC_{TM}$ turbopump module was selected because the need exists to demonstrate the capability to operate with zero NPSH at the pump module inlet. The zero NPSH capability promotes operability by eliminating propellant tank pressurization systems. Also, the individual operating behavior and characteristics of each of the module components are strongly influenced by system interactions. Thus the feasibility of the integrated jet boost/$SLIC_{TM}$ pump module should be proven through technology development and demonstration.

### 6.1.4 INTEGRATED PROPULSION MODULE TEST BED

Technology advances in several key areas that significantly impact rocket propulsion design and operation are currently being demonstrated. New materials are available that reduce the weight of high temperature and highly stressed parts. New components, such as hydrostatic bearings reduce checkout requirements, increase life and reduce weight, size, and cost. New fabrication methods ($SLIC_{TM}$) can significantly reduce production costs while enabling the production of parts with greater geometric complexity. Advanced concept demonstration programs are either planned or already underway in the areas of combustors, injectors, igniters, oxidizer-rich preburners, EMA valves, turbomachinery, and control systems. The need for system level testing to complete demonstration of these components is an accepted method for rapidly completing the technology maturation process. Experience has shown that component operation in the presence of system interactions is the only approach to truly demonstrate the feasibility of the concepts. In addition to the stated justification for this test bed, it will incorporate hardware to tap-off pressurized hydrogen and oxygen to accumulators, which are plumbed to an RCS thruster. These gas sources can also be used for propellant tank pressurization or turbine spin start demands if the system needs them. Thus the test bed provides an arena to prove the feasibility of a two-propellant integrated vehicle propulsion system. An operationally focused test approach provides total flexibility for bringing maturing technologies to the level just below that of a flight test.

## 6.2 TECHNOLOGY PLANS

Four technology plans were developed: the oxidizer-rich preburner, $SLIC_{TM}$ turbopump, jet boost pump/$SLIC_{TM}$ turbopump module, and a test bed for the integrated propulsion module. NASA's technology readiness levels (Level 1 -- basic principles observed and reported; Level 2 -- technology concept/application formulated; Level 3 -- analytic and experimental proof-of-concept for critical function and/or characteristic; Level 4 -- component and/or breadboard demonstrated in laboratory, Level 5 - component and a breadboard demonstrated in relevant environment; Level 6 -- system validation model demonstrated in simulated environment; and Level 7 -- system validation demonstrated in space) were used to define the status of each technology.. A description of each technology plan is stated below.

### 6.2.1 Oxidizer Rich Preburner Technology Development Plan

Elimination of turbopump intermediate seal purging requirements removes the need for one commodity usage and simplifies the seal, enhancing system operability aspects. In order for the oxygen turbopump to operate without seal purging, its turbine must be driven with an oxidizer-rich gas. Oxidizer-rich preburner technology development enables the simplified oxidizer turbopump. An oxidizer-rich preburner further enables a wider engine system operating range as the injector operates as a gas-gas (fuel and oxidizer) system. Deep throttling, on the order of 10:1 can be obtained. In addition, sources of gaseous hydrogen and oxygen are available for tank pressurization and RCS systems, all of which enhance operating efficiency by simplifying the overall propulsion system. Areas of injector patterns, performance and stability, and preburner design, as well as materials compatibility in an oxidizer-rich environment must be addressed.

6.2.1.1 Objectives Of Technology Development. The objective for oxidizer-rich preburner technology development is to demonstrate operation of a full scale oxidizer-rich preburner.

6.2.1.2 Benefits. Oxidizer-rich preburner technology is integral to the baselined hybrid engine cycle. The potential benefits of the hybrid cycle include improved propulsion system operability, reduced system costs, and increased power availability for increased engine performance, .

Propulsion system operability improvement is realized by allowing simplification or elimination of supporting systems. The most significant user of helium in most conventional engine systems is the oxidizer turbopump's intermediate seal purge. This seal purge separates the oxidizer from the typically fuel-rich turbine drive fluid. This purge is not required in the hybrid cycle since the only consequence of leakage from pump to turbine is a minor increase in required turbine power. With the elimination of this normally required purge comes the possibility of eliminating the need for helium. The other uses of helium may be replaceable with low level purges of one of the propellants. This feature, combined with the use of electromechanical actuators (EMA's) for valve positioning, can result in a two- fluid propulsion system. The complete elimination of pneumatics and hydraulics provides significant operational benefit and simplification over conventional systems.

Another potential operational benefit of this hybrid cycle is the possible integration of reaction control systems with main engines. Most reaction control systems currently in use operate with storable hypergolic propellants. The handling of these separate propellants causes a significant operational burden on the servicing agency, since a separate propellant infrastructure is required for these highly toxic liquids. An oxygen/hydrogen hybrid system can actually supply high pressure gaseous propellants to accumulators during engine operation, thus eliminating the need for separate RCS propellant handling. Currently, engines would require the use of maintenance intensive fuel/oxygen heat exchangers to gassify the oxygen. In addition these high pressure gases can serve other uses such as spin start power for the turbopumps and tank pressurization.

The hybrid cycle will also allow significant propulsion system cost reductions. Operational costs will be significantly decreased due to the operability improvements stated above. Propulsion system hardware cost will be reduced by simplifying and eliminating most of the components and subsystems. The oxidizer turbopump will be greatly simplified since the inter-propellant seal packages are eliminated. This not only reduces the cost of the component significantly, but it provides the opportunity to use new concept turbomachinery that decreases part count and weight while increasing efficiency, operational range, and reliability. Additional cost savings are realized by the simplification or elimination of pneumatic systems and the complete elimination of hydraulic systems. The hybrid cycle's improved power margins also improve cost effectiveness since lower turbine operating temperatures will be seen for a given design chamber pressure. These reduced temperatures allow the use of more conventional, lower cost materials in the "hot gas" portion of the engine.

Increased engine performance is realized since higher chamber pressures are attainable than those of conventional expander cycles due to the increased turbine power availability. A conventional expander cycle is limited to whatever heat can be attained by cooling the combustion chamber and nozzle to provide power to both the fuel and oxidizer pumps. With the hybrid cycle, all the energy from the cooling circuits is available for the fuel turbopump alone. The low temperature, oxidizer-rich preburner allows for safe use of oxidizer as a working fluid to power the oxidizer side. The increased chamber pressure allows higher expansion ratios for a given engine envelope, thus delivering a significantly higher specific impulse. Alternatively, perhaps more significantly, the turbine drive temperatures can be reduced, thus enhancing system durability.

**6.2.1.3 Approach**. The approach is sequential tasks starting with analysis and cold-flow testing of the preburner injector. This is followed by the tasks of oxidizer-rich preburner testing and materials compatibility testing.

**6.2.1.4 Technology Status (and Rationale)**. Fabrication designs have been completed, under an IR&D program including analytical work for critical functions. Therefore, the oxidizer-rich preburner technology, in the Rocketdyne configuration, is rated at readiness level 3.

**6.2.1.5 Relationships to Current/Ongoing Efforts**. The Russian RD-170-High Pc ox-rich preburner utilizes an intermediate mixture ratio main injector (with LOX and RP), plus a LOX-cooled channel wall chamber. LOX coolant is dumped into hot gas flow downstream of main chamber. Rocketdyne is investigating direct injection of propellants at high mixture ratio (no downstream dilution).

The use of oxidizer-rich turbine drives has been realized in the Russian RD-170 rocket engine, but, in this case, propellants are injected downstream of the preburner injector. The Rocketdyne concept instead has no downstream propellant injection, but operates at O/F mixture ratios exceeding 140. Here the combustion gases themselves are oxidizer-rich.

**6.2.1.6 Schedule**. The oxidizer-rich preburner technology development schedule is shown in Figure 6.2-1.

**6.2.1.7 Facility Requirements**. Rocketdyne's Advanced Propulsion Test Facility (APTF) can be used for the subject technology development program. It is assumed that all necessary propellants will be furnished by the government.

**6.2.1.8 ROM Cost.** The total estimated cost for oxidizer-rich preburner technology costs development is $750K.

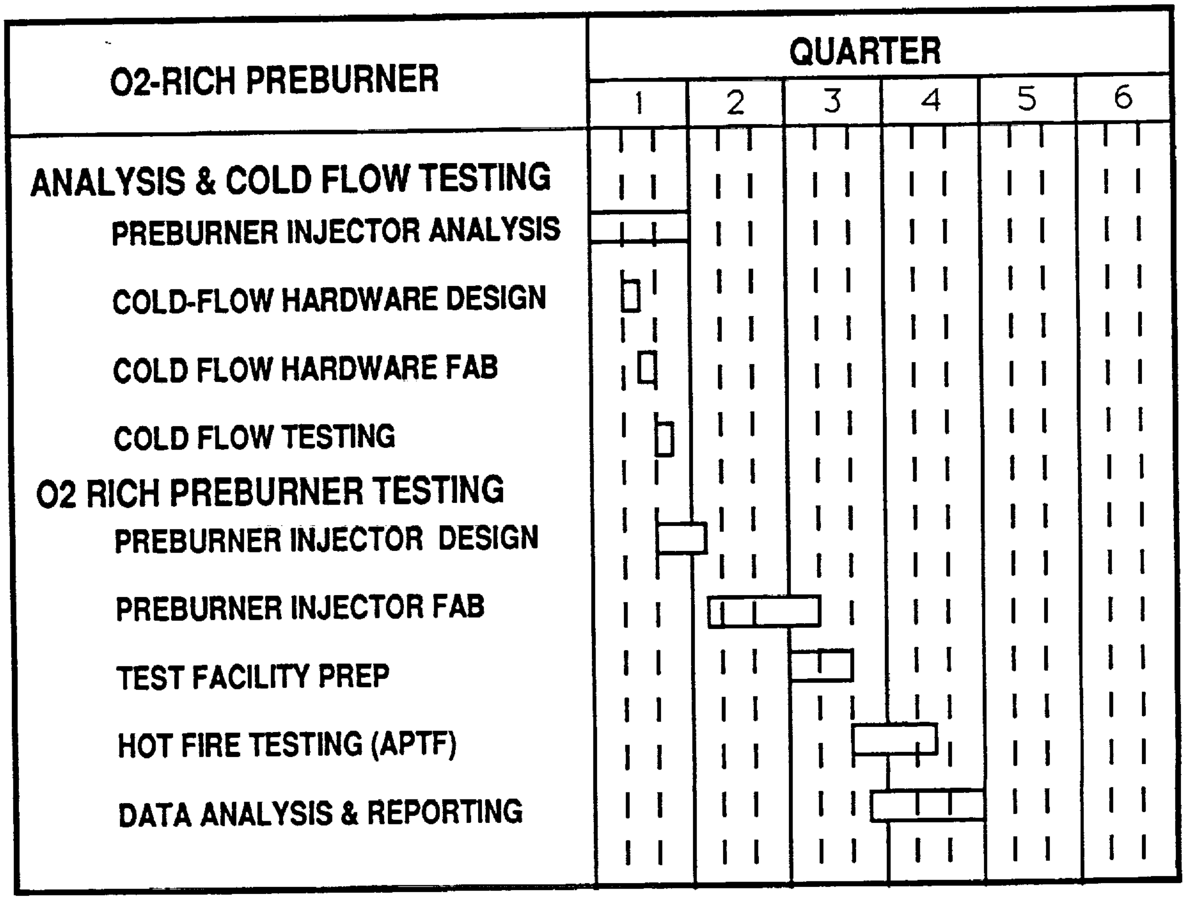


**Figure 6.2-1. Oxidizer Rich Preburner Technology Development Schedule**

## 6.2.2 SIMPLE, LOW-COST INNOVATIVE CONCEPT TURBOPUMP (SLIC$_{TM}$) TECHNOLOGY DEVELOPMENT PLAN

Liquid propulsion rocket engine turbomachinery is an integral/critical component for providing high thrust-to-weight ratio propulsion systems. Typically turbomachinery is expensive and complex, with large numbers of parts, elaborate seals, mechanical bearings, and some with gear trains. These attributes cause typical turbomachinery to be the source of both reliability and operability concerns. A space-based transfer propulsion system needs both minimal operability demands and maximum reliability. The SLIC$_{TM}$ pump concept addresses these issues by featuring; minimum number of parts, hydrostatic bearings, and simplified construction.

**6.2.2.1** __Objectives Of Technology Development__. Objectives are to prove SLIC$_{TM}$ pump operational success of $LO_2$ and $LH_2$ simplified turbopumps at the conditions required for the STPOES propulsion system application.

**6.2.2.2** __Benefits__. This patented SLIC$_{TM}$ concept replaces conventional turbomachinery, consisting of hundreds of separate parts, with only a few parts. Operational efficiency is gained in the IPM concept, since an intermediate seal purge is not required on the oxidizer pump. The rotor and housing parts are manufactured through rapid prototyping techniques and incorporate numerous subtle design refinements that enable the device to have higher efficiencies, lower weights, and higher reliabilities than conventional turbomachinery. Gains are also realized through fabrication schedule improvements and evolution of advanced materials and fabrication processes.

**6.2.2.3** __Approach__. The approach to this effort develops both the $LH_2$ and $LO_2$ SLIC$_{TM}$ turbopumps in parallel with Task 1 -- design, Task 2 -- fabrication, Task 3 -- facility preparation, and Task 4 -- testing.

**6.2.2.4** __Technology Status (and Rationale)__. The current technology status of the SLIC$_{TM}$ turbopump is NASA Level 4 to 5. Level 4 is satisfied since the component has been demonstrated in the laboratory. Level five is approached since a prototype SLIC$_{TM}$ turbopump has been tested in a relevant environment, by pumping cryogenic fluid (liquid nitrogen), with a cool nitrogen turbine drive gas.

**6.2.2.5** __Relationships to Current/Ongoing Efforts__. Rocketdyne continues development of this concept on IR&D funding. SLIC$_{TM}$ turbopump application has also been examined on the "Integrated Modular Engine" contract, under the Air Force Phillips Laboratory.

**6.2.2.6** __Schedule__. The SLIC$_{TM}$ Turbopump development schedule is shown in Figure 6.2-2.

**6.2.2.7** __Facility Requirements__. Rocketdyne's Advanced Propulsion Test Facility (APTF) can be used for SLIC$_{TM}$ pump development. It is assumed that needed propellants will be furnished by the government.

**6.2.2.8** __ROM Costs__. The total estimated cost is $6.5M to develop both the oxidizer and fuel turbopumps.

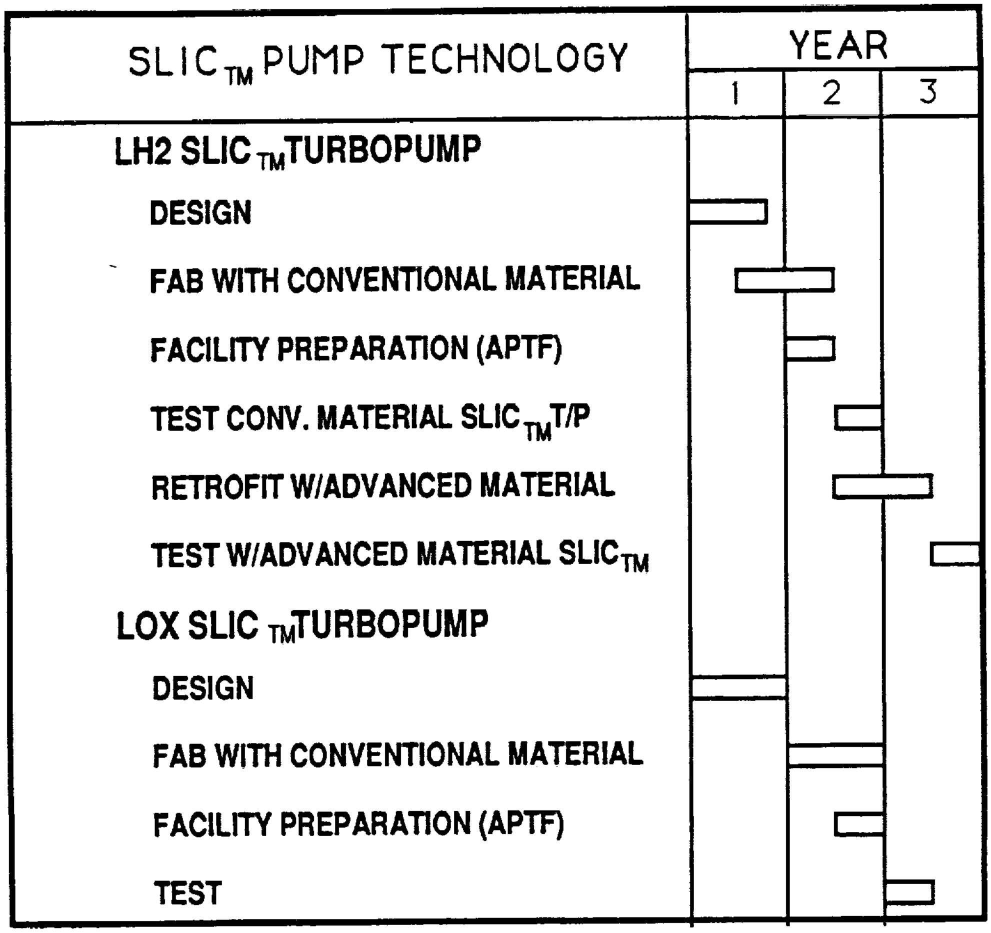


**Figure 6.2-2 SLIC Pump Technology Schedule**

## 6.2.3 PUMP MODULE TECHNOLOGY DEVELOPMENT PLAN

The space transfer propulsion application has led to a design where there is zero net positive suction head (NPSH) available at the turbopump inlet. To provide the needed NPSH, a jet boost pump is used upstream of the turbopump. Development of this technology will be accomplished using a jet boost pump in conjunction with a turbopump. This approach evaluates pump module interactions. A space-based transfer propulsion system needs both minimal operability demands and maximum reliability and performance. An integrated pump module design is presented which features a jet boost pump, with no moving parts, directly attached to a $SLIC_{TM}$ pump.

**6.2.3.1** __Objectives Of Technology Development__. The objective of pump module technology development is to prove the feasibility of pump module design and operation for the STPOES. This includes characterization of jet boost pump/$SLIC_{TM}$ turbopump system interactions.

**6.2.3.2** __Benefits__. Using the pump module allows use of zero NPSH propellant inlet conditions, since the jet boost pump provides inlet head to the $SLIC_{TM}$ turbopump. Additionally, gains are realized in: improved propulsion system operability, increased power availability for engine performance, and reduced system costs. The short schedule required for production of this module allows flexibility in planning how it will be obtained.

**6.2.3.3** __Approach__. The approach to this effort develops both the $LH_2$ and $LO_2$ $SLIC_{TM}$ turbopumps in parallel with Task 1-- design, Task 2 -- fabrication, Task 3 -- facility preparation, and Task 4 -- testing. Jet boost pump fabrication and testing is parallel to $SLIC_{TM}$ fabrication and testing. The Hydrogen $SLIC_{TM}$ turbopump will be fabricated and tested first.

**6.2.3.4** __Technology Status (and Rationale)__. The jet boost pump/$SLIC_{TM}$ turbopump module has reached the conceptual design stage. The current technology status of the pump module is NASA level 2.

**6.2.3.5** __Relationships to Current/Ongoing Efforts__. Conceptual examination of a jet boost pump/$SLIC_{TM}$ turbopump module has been made as part of the Air Force Phillips Laboratory "Integrated Modular Engine" program. Additionally the National Aerospace Plane (NASP) program is considering use of jet boost pumps.

**6.2.3.6** __Schedule__. The jet boost pump/$SLIC_{TM}$ turbopump module schedule is shown in Figure 6.2-3.

**6.2.3.7** __Facility Requirements__. Anticipated testing of the pump module can be accomplished at Rocketdyne's advanced propulsion test facility (APTF). It is assumed that all needed propellants would be furnished by the government.

**6.2.3.8** __ROM Costs__. The total estimated cost is $1.0M, assuming the $SLIC_{TM}$ development plan described in Section 6.2.2 is completed.

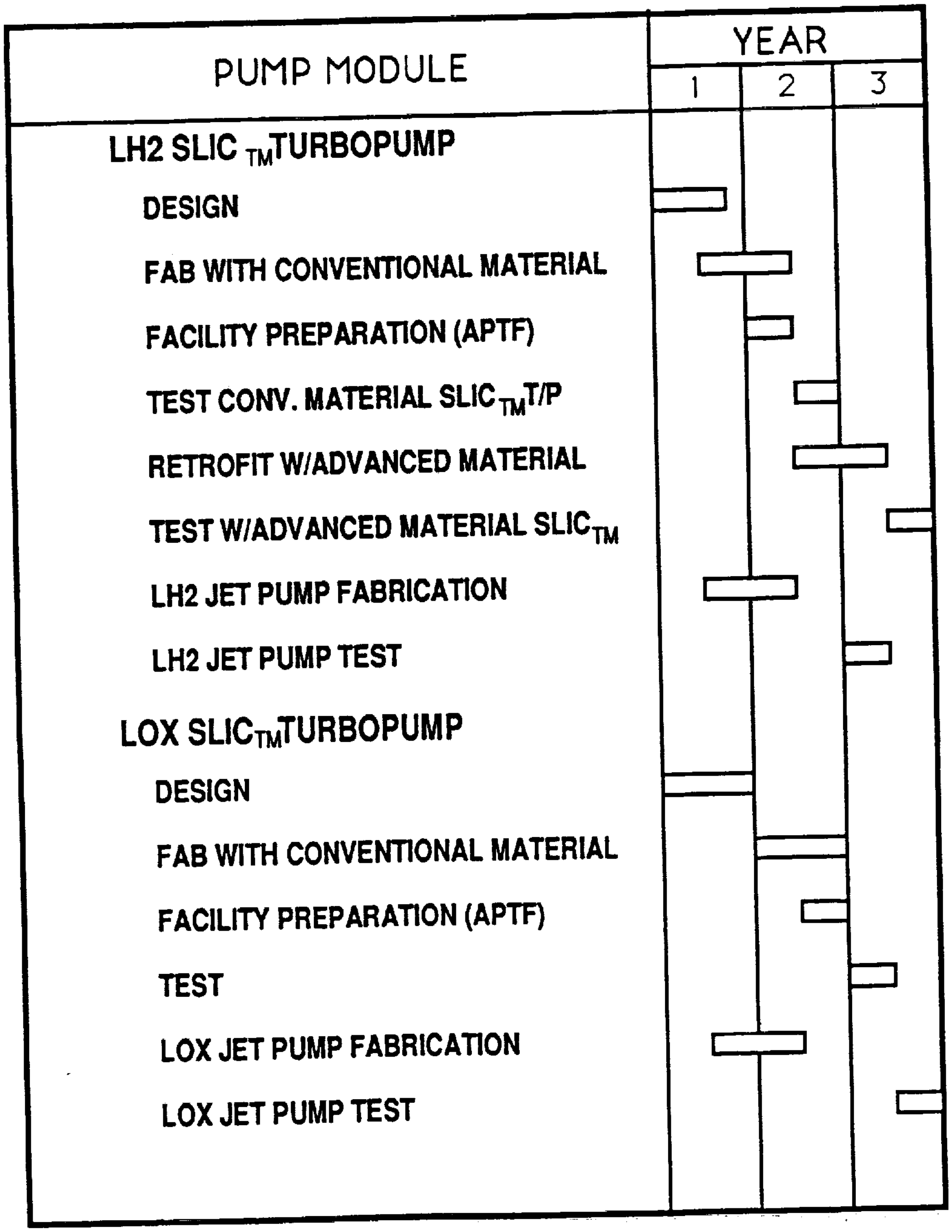


**Figure 6.2-3 Jet Boost Pump/SLIC Turbopump Technology Schedule**

### 6.2.4 INTEGRATED PROPULSION MODULE TEST BED TECHNOLOGY DEVELOPMENT PLAN.

The culminating technology tying together the elements of STPOES technology development tasks and providing a framework for other component technologies is an integrated propulsion module (IPM) technology development plan. The concept can be proven along with its operational advantages. The STPOES test bed development plan is phased to build on sequential technology and experience gains. The test bed plan demonstrates operational efficiency advantages and includes demonstrations of an integrated two-fluid propulsion system, hybrid cycle propulsion, $SLIC_{TM}$ turbopumps, jet boost pumps, differential throttling, and an integral Reaction Control System (RCS).

**6.2.4.1 Objectives Of Development Plan.** The objective of this technology plan is to provide a system level validation of an integrated propulsion module, designed for operationally efficient execution of space transfer missions. Characterization of component behavior with system interactions and differential throttling for thrust vector control is included in this scope. System level operation provides a much more convincing demonstration of system operation since it not only provides real transient and steady state operating characteristics for the components, but also provides insight to the nature and magnitude of component-to-component interactions.

**6.2.4.2 Benefits.** Benefits of this technology development include validation of the feasibility of an integrated propulsion module. Features include deep throttling, two-axis differential throttling, high chamber pressure, and integrated RCS operation. The test bed system will also be able to prove the capability to perform with only two propellant commodities demonstrating vast simplification over existing upper stages. Another benefit is the demonstration of the $SLIC_{TM}$ pump in a complete engine system operating environment.

Testing of this hybrid engine cycle will aid in the understanding of engine start and shutdown characteristics, steady state operation sensitivities and performance, and component durability trends. This data then can be used to anchor existing analytical codes which can then be used for engine design analyses for future operational systems with a high level of confidence. Additionally, the traits of operating an integrated modular propulsion system with multiple thrust chambers, ring-manifolds, and tank-mounted turbopumps will be developed and understood.

**6.2.4.3 Approach.** The Rocketdyne Integrated Propulsion Module Technology Demonstration Program discussed here uses concurrent or previous technology developments, leading to a full hybrid cycle test bed demonstration. This test bed approach offers maximum flexibility in defining and reducing to practice the technologies for an Integrated Propulsion Module system. Advanced components are developed and then combined leading to validation of advanced technologies in a system environment.

This technology development could be completed over a five-year period. Enabling technologies (Oxidizer-rich preburner, $SLIC_{TM}$ pump) lead the effort and then are combined with the needed complement of components to comprise the integrated propulsion module system test bed. The IPM test bed configuration will incorporate four thrust chamber assemblies so that the system can be characterized in its proposed flight design. The real gain is total propulsion system technology development.

6.2.4.4 <u>Technology Status (and Rationale)</u>. Evaluation of the integrated propulsion module test bed technology results in a technology readiness level 2 assignment. Although some components of this system have evolved to higher readiness levels, the entire IPM testbed system is at the conceptual design stage.

6.2.4.5 <u>Relationships To Current/Ongoing Efforts</u>. Fabrication technologies being developed under the Air Force Phillips Laboratory "High Performance Thrust Cell" contract are applicable to components of this IPM test bed. Rocketdyne IR&D-funded efforts on $SLIC_{TM}$ turbopumps and oxidizer-rich preburners are related.

6.2.4.6 <u>Development Schedule</u>. The IPM test bed development schedule is shown in Figure 6.2-4.

6.2.4.7 <u>Facility Requirements</u>. Some components which will be integrated into the IPM testbed will first be tested at Rocketdyne's Advanced Propulsion Test Facility (APTF). The IPM test bed system testing activities will occur at a government facility, which can be determined by NASA. This effort assumes that all necessary propellants will be furnished by the government.

6.2.4.8 <u>ROM Costs</u>. The total estimated cost for the IPM test bed is $24 million.

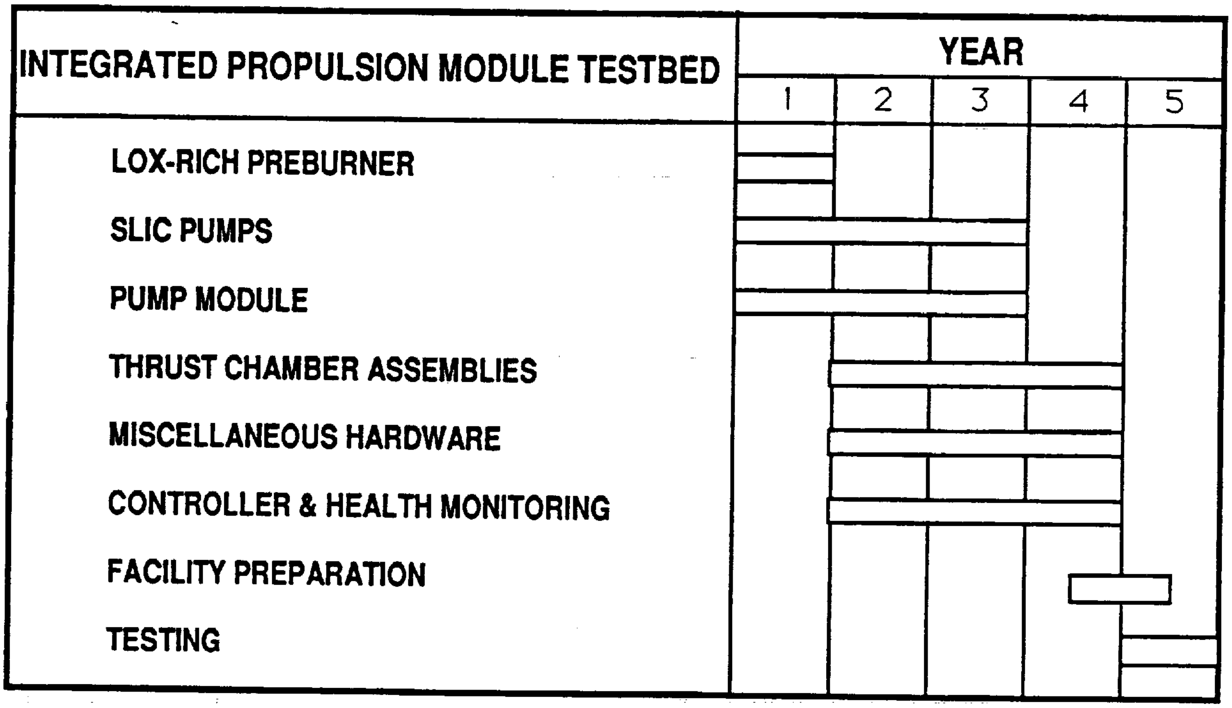


**Figure 6.2-4 IPM Test Bed Development Schedule**

# 7.0 RESULTS AND CONCLUSIONS

The Space Transfer Propulsion Operational Efficiency Study task studied, evaluated and identified design concepts and technologies which minimized launch and in-space operations and optimized in-space vehicle propulsion system operability. NASA defined a Lunar Lander mission/vehicle as the propulsion system to apply operability methodology and conceptualize an operable in-space propulsion system. The four design concepts that were developed were driven by operational considerations and each iteration provided a more operable concept. The final design iteration is highly operable and the supporting technologies are doable and would support an early year 2000 Lunar mission schedule. These operationally efficient designs revealed the necessary technologies to allow development of an operable Lunar lander concept.

Study task elements included acquiring operations databases from four current and past flight systems, initiating and defining a process to produce an in-space operations index, conceptualizing four operations driven lunar lander propulsion system designs and recommending technologies which require development in order to bring these operational designs to fruition.

Databases were generated relative to Launch, In-Space, and Management/Control operations. These databases are preliminary and can be significantly expanded. Sources used (government, industry, academic libraries and contacts) are cited and additional sources are noted, the pursuit of which was beyond funding the scope of the present effort. The experience gained in establishing the databases has led to the conclusion that a more effective process would be to conduct workshops with persons experienced in the subject area. Database information relative to operations is voluminous, however, the drawback has been the difficulty in obtaining the data. Data analysis was limited as most of the task time was spent accumulating information for the database volumes. The data gathering impediments were: 1) No single area where data is filed under the system category, 2) Data was in personal files, libraries, history files, repositories, on micro-fiche, or missing 3) Apollo era data was archived in regional storage facilities, and 4) Regional storage facilities required extensive travel. The workshop format could serve as the primary source of focus of operations issues with high importance. The documented data could then be obtained by the experts who are, or have been, working in the particular area.

Conceptual designs were devised which minimized operability concerns and issues for a Lunar Lander propulsion system. The concerns and issues list for the Lunar lander was generated from issues in the database, experience-based concerns (from KSC, Rocketdyne and Rockwell Space Division), and Lunar Lander requirements. Future database analysis and workshops on in-space propulsion could expand and modify the list. It was concluded that Lunar Lander operations concerns can be successfully addressed in the propulsion system design. The propulsion system designs included propellant tanks, propellant distribution and the necessary rocket engine components. Major operability enhancing features were a two fluid ($LOX/LH_2$) system, integrated designs including RCS, differential throttling for thrust vector control, zero NPSH pumps (no tank pressurization), turbopumps interfaced directly to propellant tanks, and no hydraulics, pneumatics, helium, hypergolics, monopropellants, gimbal systems or flex lines.

The Launch Operations Index (LOI) developed for boosters was used as the model for development of a In-Space Operations Index (ISOI). The LOI and ISOI studies have provided a road map for developing an operations index for propulsion systems. The methodology described is a first draft for an In-Space Operations Index (ISOI) approach. It is intended to stimulate thought by those experienced with in-space operations. This In-Space Operations Index is intended to be improved over the long term, through workshops, seminars, and in-space operations database additions. A similar approach is being used to develop the Launch Operations Index.

Design comparisons between the four Lunar Lander propulsion system concepts and the Centaur, S IV-B systems were completed. The immaturity of the In-Space Operations Index precluded propulsion system comparison against in-space concerns, however, a Launch Operations Index comparison was made. As all systems must be earth launched, use of the LOI has initial validity. The LOI percentages were all in the low 80's for the Lunar Lander conceptual designs. This compares with LOI percentages in the mid 30's for the Centaur and S IV-B. The NASA requirement to achieve an Operations Index greater than 0.9 was not achieved, indicating additional work focused toward achieving this goal should be pursued.

A computer-based LOI (Mac and IBM versions of EXCEL*) is available on 3.5 inch floppy disc. The user-friendly program provides operations guidance to the propulsion system designer by rating his selected concept(s) with respect to operability.

## 8.0 RECOMMENDATIONS

The OEPSS has made several contributions toward the goal of promoting operability for future propulsion system designs. However, there is more to be done in a number of areas. Activities (workshops, presentations, and other meetings) which implement concurrent engineering between planners, designers and operators should continue. Documentation and sharing of personal operational experience should be expanded to illuminate operational issues and provide databases for future use.

Four database volumes on in-space propulsion systems have been produced. The data in these volumes needs to be analyzed with a focus on issues based on experience and other operational issues should be culled. Additional pertinent database information should be added to these volumes as they are found. Subsequent propulsion systems concepts can be built on this foundation of experience.

The roadmap for an In-Space Operations Index (ISOI) generated by this study should continue until an functional ISOI has been completed. A methodology for combining indecies was suggested but alternatives and/or further implementation needs to be explored. The LOI can also be deepened to provide a means of assessing component operability. This index building process can be expanded, as a strategic tool, to other types of propulsion systems and to other operations on the ground and in space.

The LOI, ISOI and other indices, need to have wide distribution and usage if they are to be effective in enhancing propulsion sytem operability. Computer disc format and distribution would permit widespread use by designers and planners in making first cut evaluations of operability. The disc generated by this study is a valuable starting point and should be expanded as the indices are developed.

Conceptual designs with greatly enhanced operations features have been produced. As in-space propulsion requirements become firm, the designs need to be deepened to assure that operability features are included during the Conceptual, Preliminary, and Detailed design phases. Experience has shown that it is next to impossible to add operability enhancing features in later phases. The operationally efficient concepts generated in this study were intended to stimulate ideas at the component, subsystem, and system levels. These concepts should be evaluated and revised or superseded, as necessary for improvement. Alternate concepts should be generated to provide fresh approaches to an optimum propulsion system.

Technologies recommended herein, as well as by future studies, should be demonstrated as soon as possible to provide a firm foundation for subsequent development efforts. An overall plan should be implemented so that synergistic technologies can be implemented together. A test bed is mandatory for demonstrating system technology maturation. This test bed would provide convincing technology demonstration in the system environment.

Continued advancement in the operational efficiency area is mandatory if routine in-space missions are to be achieved. These efforts should include analysis, design, technology development, and group communications among those involved in design, operations and programmatics.

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