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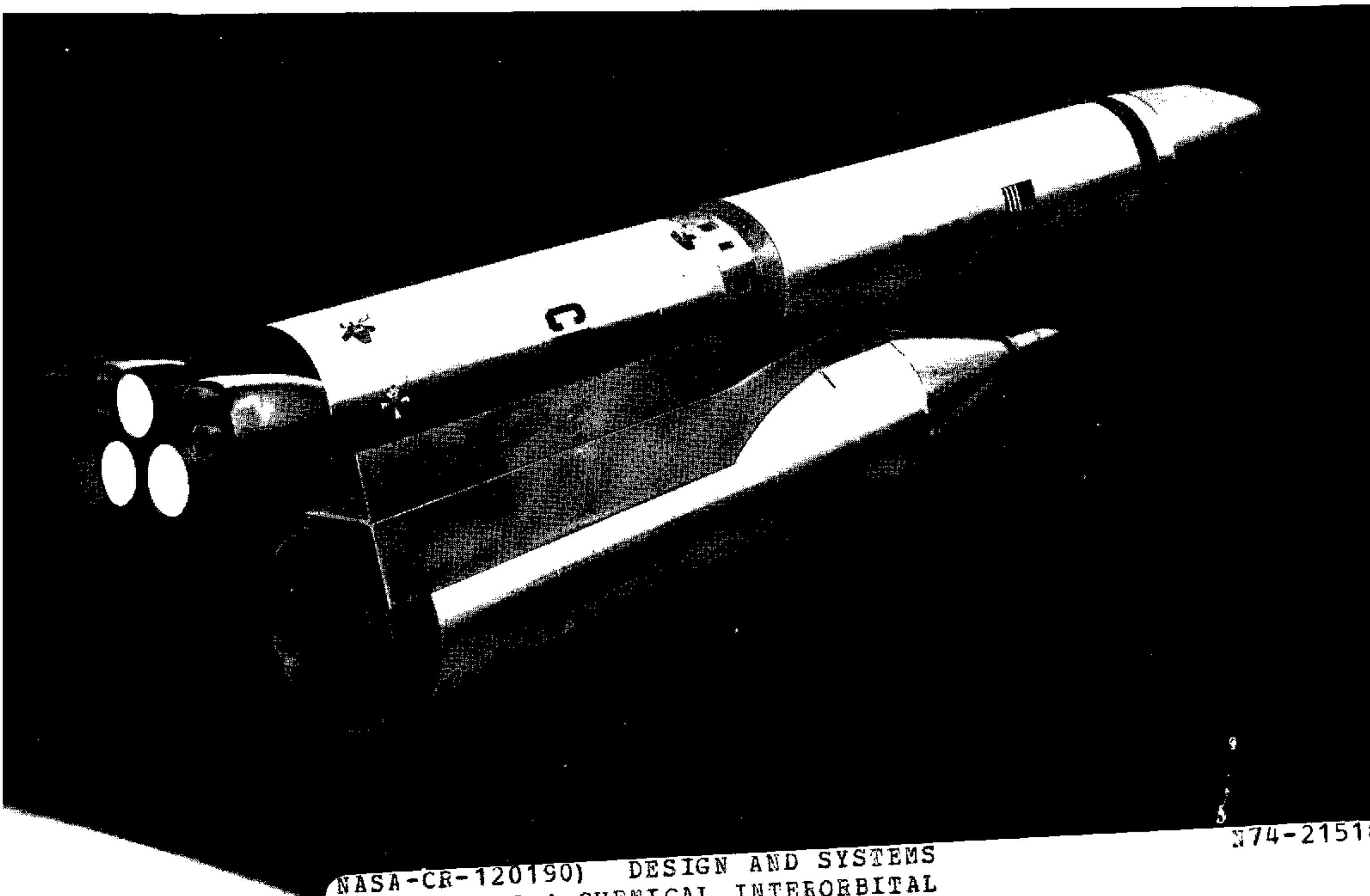
FINAL REPORT

Design and Systems Analysis of a Chemical Interorbital Shuttle



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VOLUME I

EXECUTIVE SUMMARY

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Prepared for NASA/MSFC Under Contract NAS8-27670

Space Division
North American Rockwell

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SD-72-SA-0042-1

FINAL REPORT

DESIGN AND SYSTEMS ANALYSIS OF A CHEMICAL INTERORBITAL SHUTTLE

VOLUME 1
EXECUTIVE SUMMARY

MAY, 1972 DPD 264; DR NO. MA-04

Prepared for NASA/MSFC Under Contract NAS8-27670

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TECHNICAL REPORT INDEX/ABSTRACT

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ABSTRACT

This report describes an interorbital shuttle that can be utilized to carry payloads between low earth orbit (180 n mi, 37.6°) and lunar or geosynchronous orbits, and also to interplanetary trajectories. After each mission the stage returns to its earth parking orbit where it delivers the inbound payloads, and where it is maintained and refueled for the subsequent missions. The stage can also be utilized to carry large payloads (150 to 200 KLBS) to the Space Station orbit (270 n mi, 55°) when it is used as a second or parallel burn stage to the space shuttle booster. The report volumes describe the mission and systems analysis, as well as the results of fairly detailed structural, mechanical and propulsion, and avionics subsystems analysis and design. A development plan and cost estimates are also included.

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FOREWORD

The final report of the "Design and Systems Analysis of a Chemical Interorbital Shuttle," Phase A Study, is submitted by the Space Division of the North American Rockwell Corporation to the National Aeronautics and Space Administration, Marshall Space Flight Center, Huntsville, Alabama, in compliance with Data Procurement Document No. 264, DR No. MA-04 requirements of NASA/MSFC Contract NAS8-27670.

The study was under the direction of Mr. S. P. Saucier, the NASA Contracting Officer's Representative. W. Nissim of North American Rockwell's Space Division was the study manager. This study was performed by personnel in the Space Systems and Applications organization. The contract period was from July 1, 1971 through May 15, 1972. Results of the study are documented in five volumes as listed below.

SD72-SA-0042-1 Executive Summary

SD72-SA-0042-2 Preliminary Design
- Part 1 Selected Concept
- Part 2 Design Requirements & Criteria

SD72-SA-0042-3 Design Analysis and Trade Studies
- Part 1 Mission and Systems Analysis
- Part 2 Subsystem Analysis
- Part 3 Conceptual Design

SD72-SA-0042-4 Project Planning Data
- Part 1 Project Planning Requirements
- Part 2 Schedules, Milestones and Networks
- Part 3 Support Research & Technology

SD72-SA-0042-5 Cost Estimates

A summary of the principal findings of the study is presented in this volume (SD72-0042-1).

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I. INTRODUCTION

Continued manned exploration has been identified as one of the specific objectives included in a balanced space program for this nation in the future. However, present techniques of using expendable launch vehicles for space transportation would entail very substantial costs. Consequently, new transportation capabilities need to be developed within the concepts of an integrated space program, focusing on lower-cost systems using commonality and reusability. By commonality is meant that major space systems, modified or newly developed, should be capable of adaption to multi-purpose use. By reusability is meant the repeated utilization of vehicles and spacecraft rather than expending each item in the performance of a single mission.

The space transportation requirements may be broadly defined as consisting of delivery (and in some cases recovery) of small, medium, and large payloads to low earth orbit and to geosynchronous orbits, lunar orbits, or to interplanetary transfer trajectories. Figure 1 illustrates these requirements, as well as the various space transportation system elements which meet, either singly or combined, the total mission/payload performance requirements.

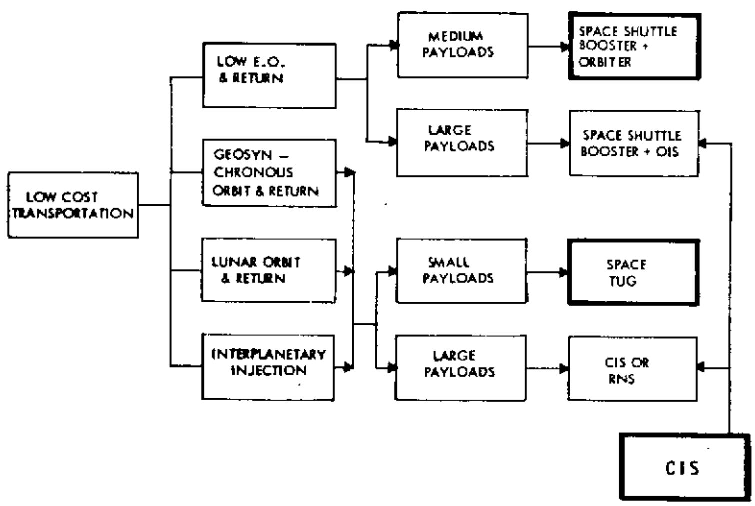


Figure 1. Space Transportation System Elements

The Chemical Interorbital Shuttle (CIS) stage meets two of the ISPP system element functions. As an orbital injection stage (OIS), the CIS stage can carry large single payloads (150k to 200k pounds) to low earth orbit. In this function, it performs as a second stage to the shuttle booster. The high performance of this stage (which replaces the shuttle orbiter for this mission) results from the elimination of re-entry and flyback capability and, therefore, reusability.

When used as a chemical interorbital shuttle (CIS), it can deliver large manned or unmanned payloads to geosynchronous or lunar orbits or to inter-planetary trajectories. In this function, the CIS stage is based in low earth orbit and is resupplied with propellants and payloads by the space shuttle. Therefore, each CIS stage is utilized to perform once as a non-expendable OIS and numerous times as a CIS.

The introduction of the CIS allows the space transportation requirements defined earlier to be met with three basic elements: namely the space shuttle, the space tug and the chemical interorbital shuttle.

Several NASA-funded studies during the last two years have investigated the design feasibility of several of these transportation system elements (space shuttle, OIS, CIS, tug). The CIS studies developed stage concepts that were modifications to the Saturn S-II and S-IVB stages. While the feasibility of the approach was demonstrated, it was recommended that an unconstrained study be conducted in order to develop an optimum configuration design. This recommendation is reflected in the objectives of this study.

A large technical base for subsystems optimization had already been accomplished in a prior study covering the S-II derived CIS. Thus, a greater fraction of total study effort could be spent in responding to changes in the space shuttle concept, since the CIS design is sensitive to the booster and orbiter design and to their operational characteristics. This technical base was also expanded in the areas of mission and operations analyses where new mission modes and lunar program payload concepts were defined. The availability of a program development plan and cost estimates for a single stage CIS allowed (during Phase III of the study) the investigation of a competitive 1-1/2 stage concept. The 1-1/2 CIS stage concept is based on the use of a drop tank for containing the translunar injection maneuver propellants. This allowed the CIS stage to be designed with a propellant capacity as required for the post TLI maneuvers. The special virtue of this concept is that the stage design is relatively insensitive to the drop tank capacity, and therefore, does not become obsolete if the drop tank capacity changes.

The current study has developed an optimization approach and established a design concept for a CIS stage which performs both OIS and CIS missions.

The technical feasibility of the CIS vehicle has been established by accomplishing a rather detailed design definition. The principal features of a development program have been identified, and costs were estimated for development, production, and operations. The costing exercise corresponded to that performed under the S-II derived CIS study (Contract NAS7-200, CO 2021), in order to allow comparisons to be made between a single and a 1-1/2 stage CIS concept.

The 1-1/2 stage concept, by virtue of its efficient utilization of the orbiter propulsion system, the drop tank, and its interfaces with the booster, has merit inasmuch as it provides to this nation the means for complementing the performance and operational capabilities of the space shuttle to allow the economic implementation of a meaningful space program.

II. STUDY OBJECTIVES

The objectives of this study were to determine the feasibility of a reusable chemical propulsion stage for future manned and unmanned missions beyond low earth orbit and also as an economical system for delivery of large payloads to earth orbit when used as an alternate second stage on the reusable space shuttle booster.

The missions are defined as follows:

Class I - Lunar/Geosynchronous Orbit Shuttle Missions

Class II - High Lift Capability to Earth Orbit

Class III - Unmanned Planetary Missions

Class I and II missions only were considered for baseline design analysis; Class III missions were considered only as potential applications of the baseline design.

III. RELATIONSHIP TO OTHER NASA EFFORTS

Data from nine other NASA studies were required during the performance of this study.

The CIS design constraints for the Class II mission were based on the Space Shuttle Phase B study (Contracts NAS9-10960 and NAS9-11160). Also orbiter performance, payload support, and manipulator characteristics in the Phase B study influenced the definition of propellant logistics, orbital assembly and maintenance, and operational costs; and orbital drop tank design and cost characteristics influenced the definition of the 1-1/2 stage CIS and the concept evaluation.

Mission modes, lunar program analysis, subsystem trades, and program cost-estimate data from the S-II Stage Interorbital Shuttle Capability Analysis (Contract NAS7-200, Change Order 2021) contributed a substantial technical base for this study.

The definition of an optimum approach for a two-stage mission from the S-IVB Chemical Interorbital Shuttle Capability Study (Contract NAS7-101, Task Authorization 9) was used as one of the alternatives for CIS concept evaluation. The In-Space Propellant Logistics and Safety Study (Contract NAS8-27692) propellant refueling concept (linear acceleration), propellant transfer times, and the conceptual design of the resupply module were coordinated with this CIS study.

The tug definition from the Space Tug Point Design Study (Contract NAS7-200, Study Authorization 2190) was utilized in the mission analysis. Also, subsystem data were utilized where applicable (i.e., tug engines were used in the CIS APS system).

The CIS lunar program and payload definition was based on the results of the Lunar Base Synthesis Study (Contract NAS8-26145) and of the Phase A Feasibility and Definition Study of an Orbiting Lunar Station (NAS9-10924).

The definition of orbital operations for the CIS were coordinated with the Orbital Operations Study (NAS9-12068). The thermal prediction methodology of the Cryo Storage Thermal Improvement Study (Contract NAS7-200, Study Authorization 2049) was used to develop CIS boiloff estimates.

IV. METHOD OF APPROACH AND PRINCIPAL GUIDELINES

METHOD OF APPROACH

Since the CIS stage would be utilized once as an OIS and ten times as a CIS, the candidate CIS stage configurations were sized to perform a selected CIS mission. The stage design would meet the requirements for both the OIS and CIS missions. The total DDT&E, production, and operations cost for a baseline CIS utilization program was used to select the most desirable configuration on the basis of minimum cost. An additional evaluation parameter was the payload-to-orbit (270 n mi, 55 degrees) performance capability of the CIS candidate concept when used as an OIS stage. An acceptable OIS performance was defined as 150K pounds or higher.

The study was conducted in four phases as illustrated in Figure 2.

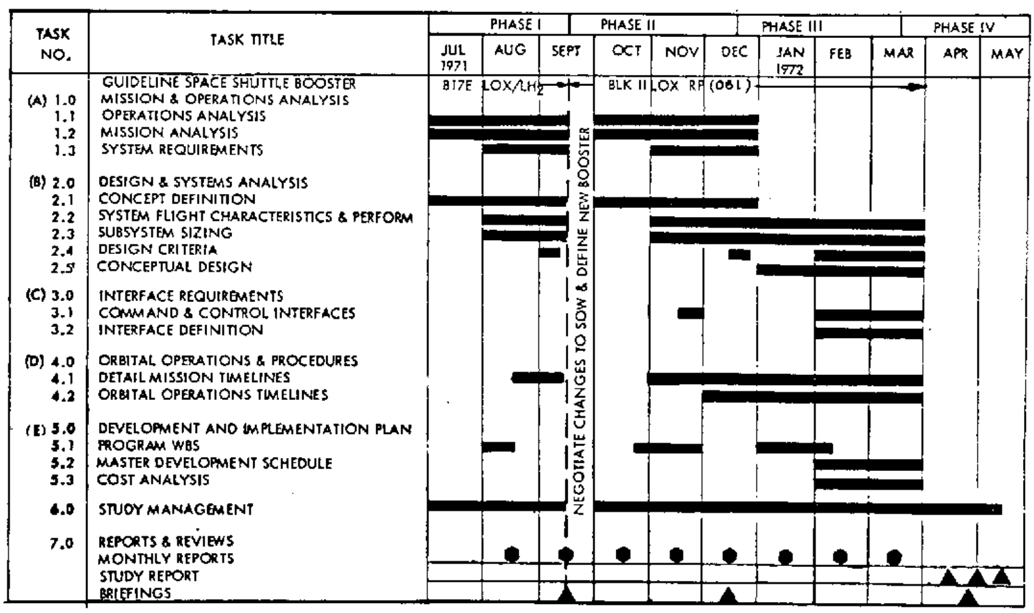


Figure 2. CIS Study Schedule

The first phase, lasting 2-1/2 months, consisted of establishing program and traffic models, performing parametric analysis and developing concepts for mission modes, system operations, and stage configurations. Preliminary analysis of the launch ascent flight characteristics and performance were performed to determine that the LOX/LH_2 flyback shuttle booster staging constraints were not violated. Phase 1 results consisted of technical data supporting the selection of two configurations for deeper analysis during the second phase of the study. At the end of Phase 1 the Shuttle Booster configuration had undergone considerable change and it was deemed desirable to repeat Phase 1 tasks based on the LOX/RP flyback booster.

Phase II of the study, lasting three months, identified the single-stage concept as an optimum CIS on the basis of preliminary space transportation cost evaluations. Since the dimensions and characteristics of the selected concept were very similar to the results of the S-II derived CIS study for which development planning and cost estimates were available, the 1-1/2 stage concept was selected for conceptual design during the third phase.

Phase III, in which the conceptual design was developed, lasted 3.5 months. During this phase, stage and subsystem design was completed, drawings were prepared depicting configuration details, installation of subsystems, and key features such as interfaces with the space shuttle booster and payload and guidance sensors and antenna installations. Interface specifications and design criteria documentation were completed. Orbital operational concepts were finalized and documented. A development and implementation plan and cost estimates were developed.

The fourth phase, lasting two months, was devoted to the preparation of the final briefing and the final report.

PRINCIPAL GUIDELINES

The CIS is man-rated, i.e., it meets all structural, material, environmental, and quality standards required for manned application. The baseline operational lifetime is three years or ten uses, whichever occurs first, and is capable of earth orbital maintenance and quiescency in space between missions for periods up to 180 days. All subsystems except the primary structure and pressure vessels are designed to fail operational after the failure of the most critical component and to fail safe for crew survival after the second failure. Electronic systems are designed to fail operational after the second failure of a critical component.

The reusable space shuttle booster configurations include the LOX/LH_2 flyback booster (B-17-E) for Phase I of the study and the LOX/RP flyback booster (061) for Phases II and III. Effects of the latest booster configuration on CIS stage design have been evaluated. Staging during earth launch missions are at conditions in which the space shuttle booster heating and flyback conditions are no more severe than anticipated for the nominal space shuttle booster/orbiter configuration. Designs that introduce an incompatibility between the space shuttle booster and orbiter were not considered, and the load carrying capability of booster primary structure was not exceeded for the baseline space shuttle booster configuration.

The primary propulsion rocket engines considered is the space shuttle high performance engine as identified in MSFC Document 13 M 15000B, Space Shuttle Vehicle/Engine 550K (SL) Interface Control Document, dated February 18, 1971.

The CIS was designed for maximum on-board control, using on-board and ground capabilities as appropriate to maximize operational flexibility and to minimize ground mission operations consistently with low cost. The avionics systems are independent of the payload; however, for manned operations, the crew has the capability for active control and provides both stage and stage/payload (combined) attitude control. The CIS is capable of control while in orbit from the space shuttle, space station, ground stations, or others, and does not present a hazard to other orbital elements and payloads. By using ground facilities and other aids when appropriate, the CIS can rendezvous, dock, and conduct maneuvers with other program elements (space station, space tug, crew modules, etc.).

All costs are normalized to GFY 1972. Common use of space shuttle technology, components, and systems is considered to reduce development costs. The flight test hardware and operations cost for the flight test program are charged to the non-recurring cost. Advanced components technology was considered where improvements are clearly advantageous and capable of development, including ground qualification prior to 1978. Lunar mission rates of 1, 2, 4, and 6 per year for a total duration of 10 years are considered for costing purposes.

The non-modular space station is the heaviest boost-to-orbit payload identified in earlier studies. It weighed 177k pounds and measured 33 feet in diameter and 111 feet in length. This payload constrained the minimum diameter of the stage during the early phases of the study. For Phase III, it was decided that the high-lift (Class II mission) payload diameter would not constrain the CIS stage optimization; and on this basis, the Phase III configuration is shown mated to a 22-foot diameter payload.

V. BASIC STUDY DATA GENERATED AND SIGNIFICANT RESULTS

The principal tasks in the study included mission and operations analysis, design and systems analysis, interface requirements, orbital operations and procedures, and a development and implementation plan. Important study outputs include:

1. Definition of an optimum lunar payload
2. CIS stage concepts and evaluation
3. Selection of CIS concept for configuration design (Phase III)
4. 1-1/2 stage configuration design (structural, thermal, propulsion and mechanical systems, avionics, interfaces, weights)
5. 1-1/2 stage performance and dynamics
6. Orbital operations
7. Development program
8. Cost estimates

As the study developed it became evident that there were some easily recognizable major study drivers. These included: (1) the requirement for developing a CIS stage which resulted in minimum overall space program costs, (2) the design influences of the space shuttle overall concept, (3) the performance required to accomplish Class I (CIS) missions, and (4) the performance required for Class II (OIS) missions. The optimum CIS stage would in turn impact the payloads in the areas of dimensional and weight constraints and interface compatibility. It would additionally create some technology requirements.

DEFINITION OF BASELINE LUNAR PAYLOAD

When the CIS missions were reviewed in order to select a baseline mission and traffic model, it was readily apparent that a ten year manned lunar exploration program incorporating an orbiting lunar station (OLS) and a lunar surface base (LSB) was the best defined, suitable program. The results of the OLS and LSB studies performed in 1971 were utilized to define the baseline CIS program and performance requirements.

An annual lunar payload requirement of approximately 600 k pounds to support the OLS and LSB programs was identified in the S-II CIS study and confirmed in Phase I of this study. Identification of the orbital propellant requirements as the overriding cost variable was also identified in the S-II CIS study. These facts permitted the early identification of the cost trend for combinations of payload weight and annual mission requirements as measured in total propellant requirements. This relationship is shown in Figure 3. The propellant requirement for six flights of 100 k pounds payload each is approximately 2-1/2 times that for two flights with a 300 k pound payload and a trend is clearly established. Detail examination of the lunar payload requirements in conjunction with other factors such as: (1) the requirement for crew rotation at 168-day intervals or less, (2) the cyclic nature of maximum payload lunar mission opportunities (every 54 days), (3) the maximum weight of a single payload element (OLS - 157k lb), and (4) compatibility with an OIS payload (~177k lb), resulted in the selection of a 320 k pound outbound and 16.8 k pound inbound lunar payload requirement for the CIS.

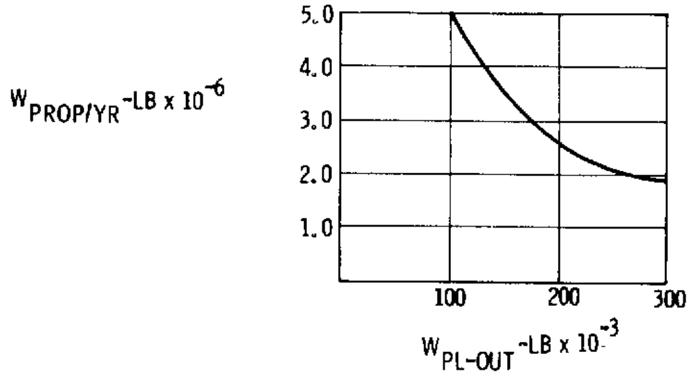


Figure 3. Yearly Propellant Requirements

CIS STAGE CONCEPTS AND EVALUATION

Candidate CIS configurations during Phase I of the study were developed on a parametric basis. After the lunar payload definition was completed early in Phase II, the stages were sized to perform the 320 k pounds outbound/16.8 K pounds inbound lunar mission. Propellant requirements per mission and for the total ten-year lunar program were developed. Additionally the OIS (Class II) mission performance of each candidate concept were evaluated. Thus the stage that resulted in minimum program cost and which also exhibited adequate OIS performance became the selected concept, see Figure 4.

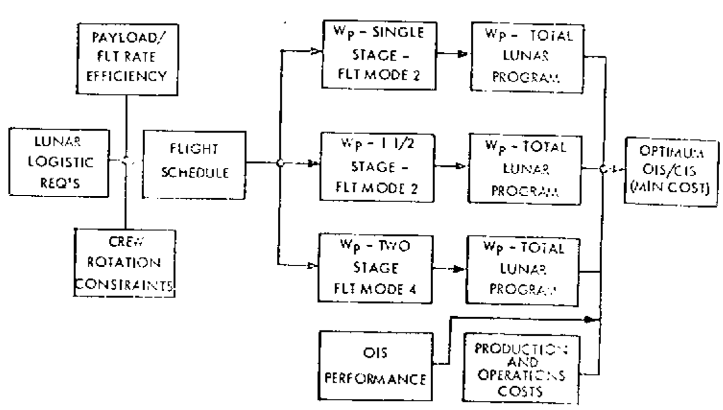


Figure 4. CIS Concept Evaluation

The sizing (propellant capacity) of CIS stages depends on the flight mode selected. Since a rather diverse variety of mission modes are feasible in delivering a large payload to and from the moon an understanding of these modes and their relationship to single, dual and 1-1/2 stage configurations is required. See Figure 5.

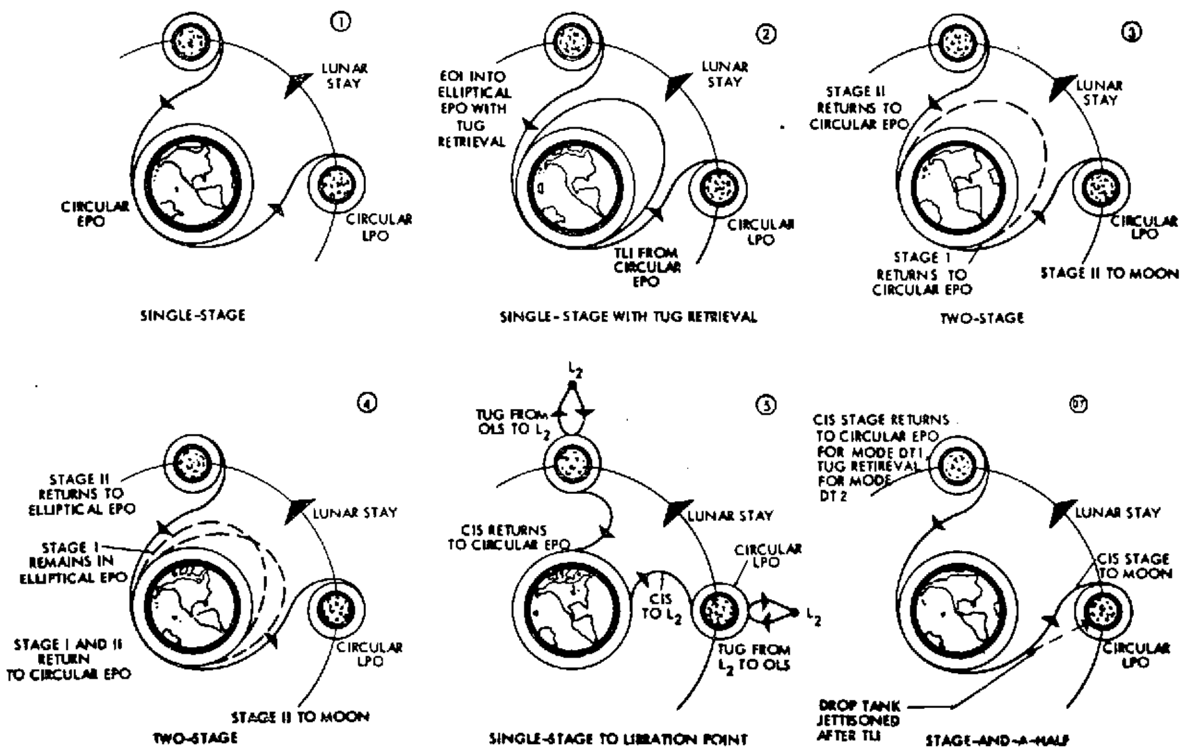


Figure 5. Mission Modes

Mode 1 - A single-stage CIS departs from a low circular earth orbit and arrives in a polar lunar orbit. After a proper time interval, a return to earth and coplanar deceleration into the original earth parking orbit is performed.

Mode 2 - This mode is similar to Mode 1 except that return is made to a highly elliptical earth orbit, thus reducing the energy requirements for the CIS. Retrieval of the CIS stage and inbound payload from the elliptical orbit is the task of an earth-based tug.

Mode 3 - A two-stage CIS kicks off from earth parking orbit under the impetus of Stage 1. Stage II completes the translunar injection and the remainder of the mission in a manner similar to Mode 1. Stage I returns independently to the earth assembly orbit.

Mode 4 - A two-stage CIS operates similarly to Mode 3 except that Stage 1 returns to a high elliptic retrieval orbit prior to return of Stage II and after remating stage 1 retrieves the whole two stage CIS back down to the original earth parking orbit.

Mode DT(1) or (2) - A 1-1/2-stage CIS employing a shuttle drop tank containing sufficient propellants for earth departure operates in either Mode 1 or Mode 2 after the tank has been jettisoned following completion of trans-lunar injection.

Mode 5 - Utilization of the lunar libration point behind the moon as a cargo transfer station between CIS and lunar delivery tug has been considered. Single stage modes 1 and 2 can be utilized in conjunction with Mode 5.

Except for the more efficient (but more complex) libration point mission, the Modes 2 and DT exhibit the least propellant requirements for the lunar mission.

SINGLE STAGE CONCEPTS (PHASES I AND II)

Single stage CIS concepts (Figure 6) developed during study Phases I and II were based on the utilization of Flight Mode 2 (tug retrieval). Phase I concepts were based on the utilization of a LOX/LH_2 booster with a staging velocity of 8500 fps. The cylindrical stage was sized for a propellant capability of 835k pounds.

During Phase II, the booster selected was a LOX/RP flyby configuration which had a lower staging velocity, approximately 8000 fps. The propellant requirement of the CIS stage increased to 937k pounds.

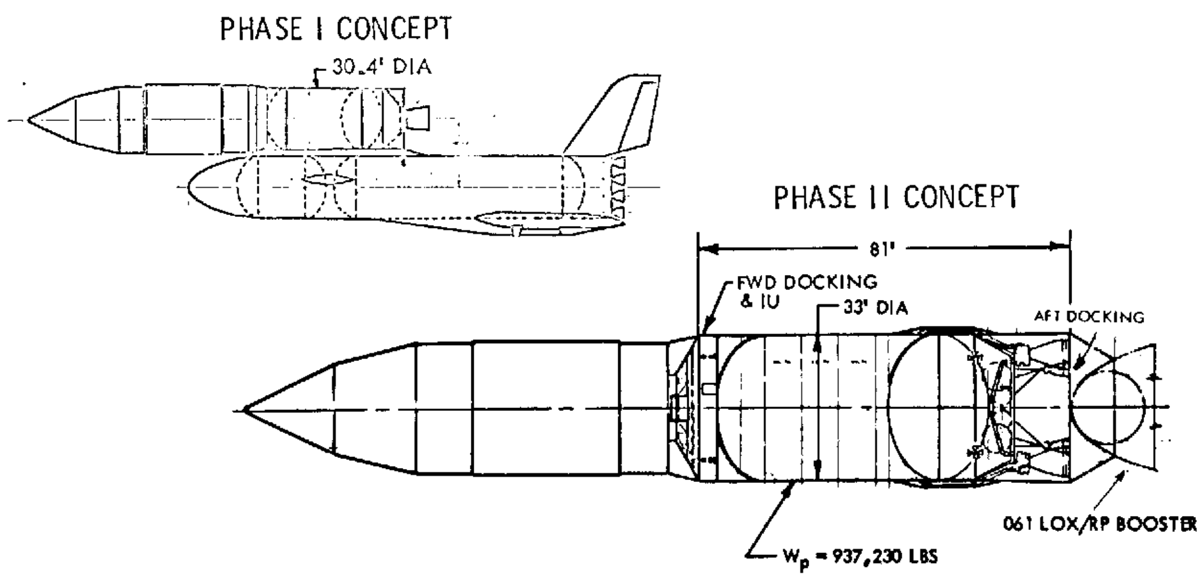


Figure 6. Phase I and II Single Stage Concepts

TWO-STAGE CONCEPTS (PHASES I AND II)

Two-stage concepts (see Figure 7) were also investigated during Phases I and II of the study; and the propellant requirements for the stages exhibit a dependency on the booster selected, in the same fashion as the single stages.

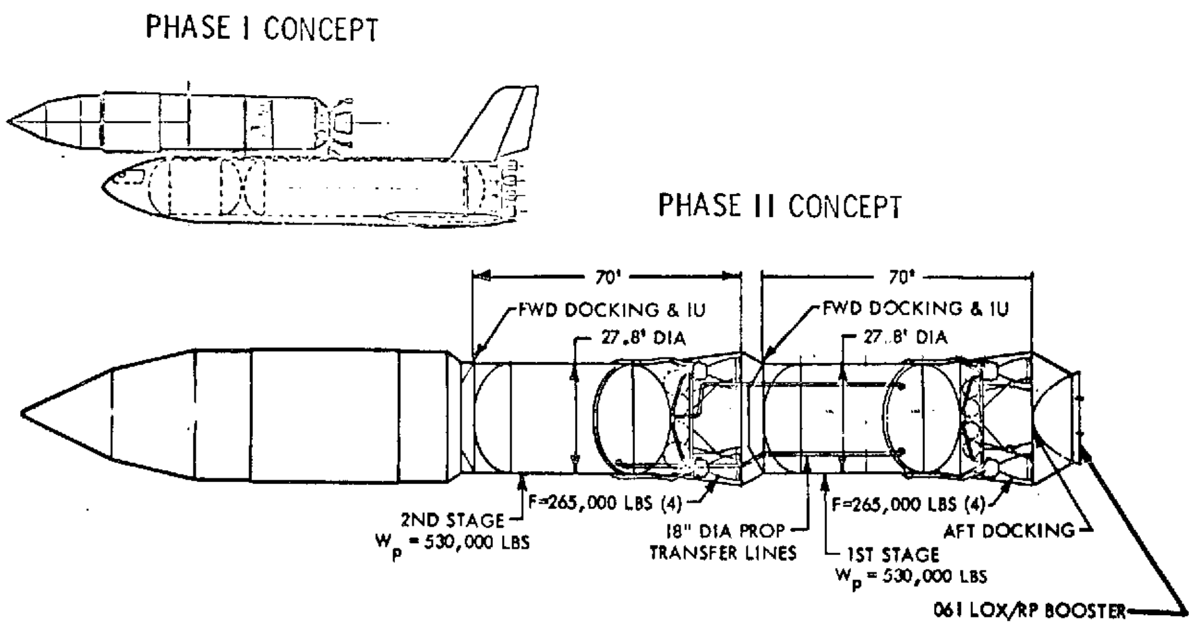


Figure 7. Phase I and II Two-Stage Concepts

A propellant loading of 450k pounds per stage were required when boosted by the LOX/LH_2 flyby booster. For the LOX/RP booster, the propellant loading per stage increased to 530k pounds. Two alternative mission modes (3 and 4) could be utilized; however, little difference in efficiency were found between them.

1-1/2 STAGE CONCEPT (PHASE II)

At the beginning of Phase II, the space shuttle concept had changed considerably, in that a drop tank orbiter was being utilized instead of a totally reusable orbiter. This change in approach was evaluated for the CIS stage, and on this basis the 1-1/2 stage concept was generated. The utilization of the drop tank and of the orbiter propulsion were potential cost saving features because of commonality with the space shuttle program. The configuration was developed on the basis of flying it in mission mode DT-2, in which the drop tank is jettisoned after completion of the trans-lunar injection maneuver. On return to the earth, the stage injects into an earth elliptical orbit from which a space tug retrieves and brings to nominal earth parking orbit (180 n mi, 37.6 degrees). The propellant requirements for the stage itself are quite small, being in the order of 180k pounds. See Figure 8.

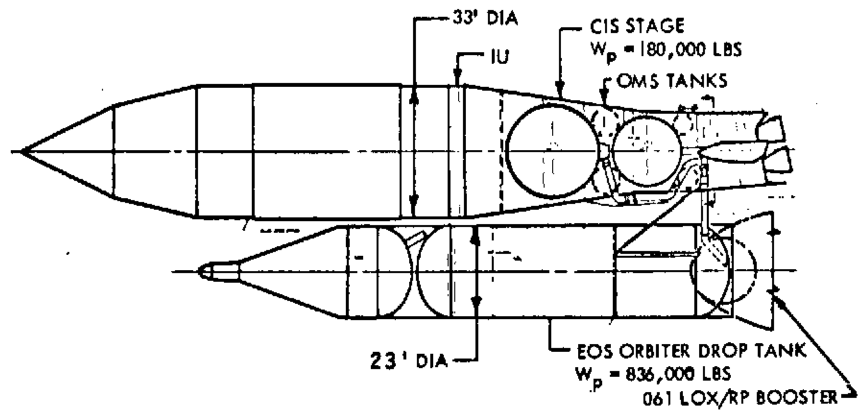


Figure 8. One-and-a-Half Stage Concept

SELECTION OF CIS CONCEPT FOR STUDY DURING PHASE III

At the conclusion of Phase II of this study, the propellant requirements for the different flight modes and their respective configurations were developed, along with the estimated overall program costs including maintenance, orbital operations, and stage development and production costs. The minimum cost stage was the single stage flying in Mode 2. The next best is the 1-1/2 stage, which requires less propellant but which, because of the jettisoning of a drop tank on each trip, costs four percent more. The next most attractive configuration is the 1-1/2 stage flying in a DT-1 mode (Figure 9). The propellant requirements (Figure 10) are a little higher, and the overall program costs are 14 percent higher than the Mode 2. The other alternatives, two stages flying Modes 3 or 4R and a single stage flying Mode 1 require more propellant and cost about 23 percent more than the Mode 2.

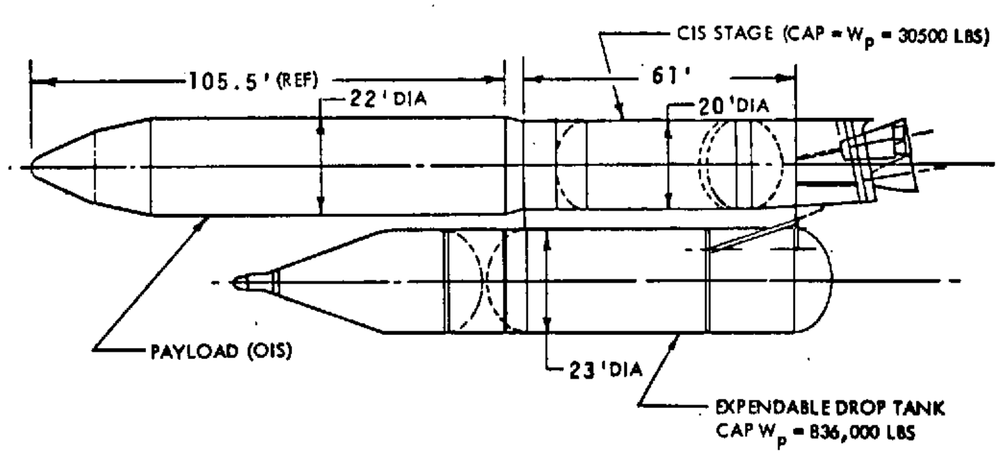


Figure 9. Concept Selected for Phase III

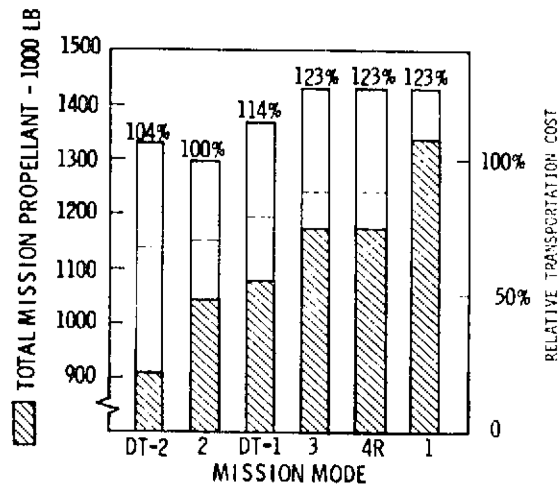


Figure 10. Total Mission Propellant for Various Mission Modes

The conclusions and recommendations for a concept selection for Phase III were as follows:

1. The single stage - Mission Mode 2 - exhibits the lowest program cost, meets Class I mission requirements, and exhibits adequate Class II mission performance.

2. The 1-1/2 stage concept flown in Mode 1 increases program costs 14 percent, but does not depend on space tug availability.

3. The two stage concept has low performance for Class II missions and results in 23 percent higher program costs for Class I missions.

4. The single stage CIS is very similar in dimension and arrangement to the S-II derived CIS, for which design and cost was developed recently.

5. The detailed design and analysis of the 1-1/2 stage concept during Phase III would allow a more accurate comparison between the single stage and the 1-1/2 stage concepts.

Consequently, NASA directed that during the remainder of the study, the design of 1-1/2 stage concept would be developed.

1-1/2 STAGE CONFIGURATION

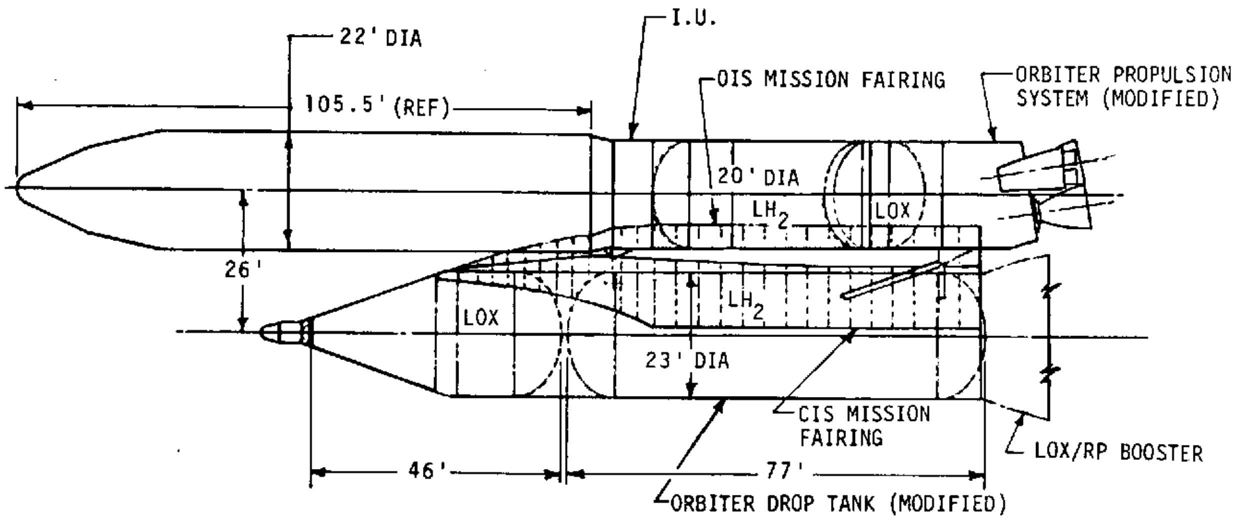


Figure 11. CIS Vehicle Concept

The selected concept (Figure 11) is a 20-foot diameter, 88.5-foot long stage which includes the propulsion section of the orbiter at the aft end and an intelligence unit at the forward end. The primary configurational components consist of the payload, the drop tank and fairing and the stage itself. The structural elements of the stage consist of the IU or avionics compartment at the forward end of the stage, attached to the forward skirt. The hydrogen tank is forward, the LOX tank aft, each a separate entity joined together with a bolting ring with the bulkheads in a nested position. The modified orbiter propulsion section is attached to the aft skirt structure. It mates with the payload at the forward end and with a drop tank in parallel fashion, identical to the shuttle orbiter/drop tank interface. Fittings attaching the stage to the drop tank are located at the forward end of the avionics compartment and at the aft end of the aft skirt structure.

The CIS stage tankage was sized for a propellant capacity of 305,000 pounds. Suitable thermal and meteoroid protection is provided to maintain propellant boil-off losses at 3.5 pounds per hour.

For the OIS mission, the foam-insulated expendable drop tank is jettisoned prior to arrival in the earth parking orbit. After two of the main engines are removed from the CIS stage, the stage is refueled and subsequently mated with an insulated (multi-layer insulation) expendable drop tank, for performing the CIS mission. After the translunar injection (TLI) phase of the CIS mission, the expendable tank is jettisoned and the remainder of the mission is performed with propellant from the 20-foot diameter CIS stage. Some of the design features of the structure and other subsystems used in the stage are described later in this summary.

The shuttle orbiter expendable drop tank and propulsion system were modified by:

1. Addition of thermal and meteoroid protection to the drop tank to minimize propellant boiloff during the CIS missions.

2. Increasing the distance between the shuttle orbiter propulsion system and the expendable drop tank to provide clearance between the 22-foot-diameter OIS mission payload and the expendable drop tank. Increasing the length of the LOX and LH_2 feedlines, and other system lines between the propulsion system and expendable drop tank.

3. Replacing the two hypergolic orbital maneuvering system (OMS) engines located on either side of the propulsion system, with four auxiliary propulsion system (APS) 10,000-pound thrust LOX/LH_2 engines (two on each side of the propulsion system).

4. Providing reusable structural, fluid and electrical disconnect systems between the CIS stage and the expendable drop tank.

5. Redesign of existing and addition of new propellant control, conditioning, and management systems to meet the CIS and OIS mission requirements.

The propellant tanks have 1.5:1 oblate spheroid-type bulkheads. The aft LH_2 tank bulkhead and forward LOX tank bulkhead are nested together with minimum clearance requirements in order to minimize the overall stage length and weight.

A short interstage is required between the aft end of the cylindrical tank structure and the modified circular cross-sectional dimensions of the orbiter propulsion system, because the overall stage length (732 inches) was predicated upon a requirement to utilize the existing expendable tank support points.

A 15-degree conical payload adapter is provided on the forward end of the stage to provide an interface for the 22-foot-diameter payload.

It was necessary to utilize fairings to prevent high acoustic loads on the CIS and drop tank insulated walls in the area where the drop tank and CIS stage interface. A high propellant boiloff would develop from the great number of posts required to support the insulation protection panels, if the fairing were not used. Addition of the fairing proved to be a good design solution.

STRUCTURES

Criteria

The design criteria (Figure 12) used in the design of the CIS stage was extracted from the study plan and supplemented with data obtained from detailed analysis. The dominant external loads develop during the Class II mission in which the payload and the stage are subjected to acceleration and drag loads. The mission life objective of 3 years or 10 uses in conjunction with the ullage pressure (25 psig - LH_2 tank; and 27 psig for the LOX tank) and the fracture mechanics guideline (NASA SP-8040) impacts the tank and structure design. The meteoroid protection environment criteria (NASA TMX-53957) impacted the design of the meteoroid and aeroshear barrier.

SAFETY FACTORS

	MANNED	UNMANNED	VIBRATION -
F. S. (ULT) F. S. (YIELD)	1.40 1.10	1.25 1.05	1.1 (SPACE SHUTTLE BOOSTER BURN ONLY)

TEMPERATURE CONDITIONS

ITEM	MAX TEMP (°F)	MIN TEMP (°F)
BODY SHELL	+ 160	-100
LH_2 TANK	+ 110	-423
LOX TANK	+ 140	-297
PROPULSION UNIT	+ 490	-100

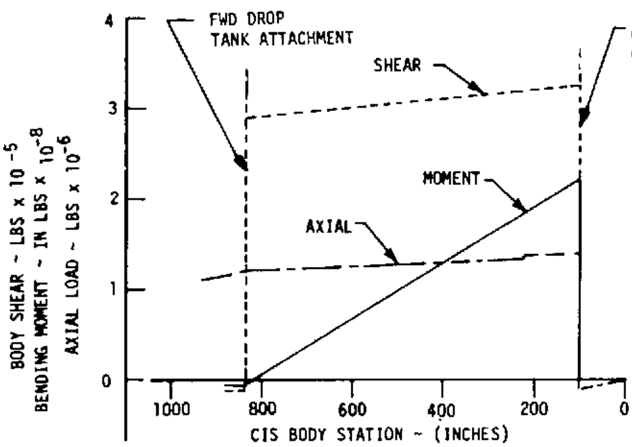
LOAD FACTORS

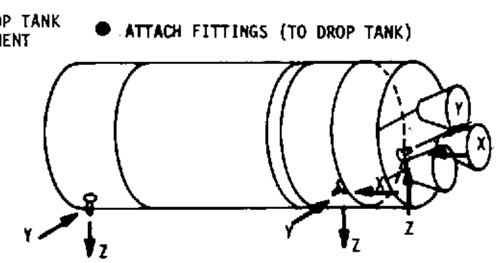
CONDITION	CIS ACCELERATIONS (G'S)		
	X	Y	Z
OIS LAUNCH FIRST STAGE BURN	-3.0	± 1.0	± 1.0
OIS LAUNCH CIS BURN	-4.0	± 0.6	± 0.6
CIS SHUTTLE OPERATION	-0.6	± 0.2	± 0.2

Figure 12. CIS Design Criteria

Structural Loading

The critical design axial load, bending moment, body shear, and Space Shuttle drop tank attach fitting loads are shown in Figure 13. The loads are limit and result from the baseline OIS mission at the initial 3.0 g condition. These loads using a rigid body analysis, serve as input data to a structural analysis computer program which solves for the internal body loads.

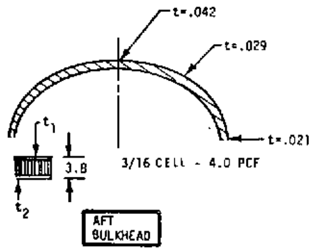




MAX FITTING LOADS (KIPS)				
FWD CIS		AFT CIS		
Y	Z	X	Y	Z
85.7	308	693	85.7	163

Figure 13. CIS Loads

LH_2 Tank



SKIN THICKNESS (NOM) INCHES			
LOCATION	STATION		
	③	④	⑤
Ⓐ	.095	.072	.067
Ⓓ	.090	.070	.067
Ⓔ	.076	.067	.067
Ⓒ	.067	.067	.067
Ⓑ	.067	.067	.067

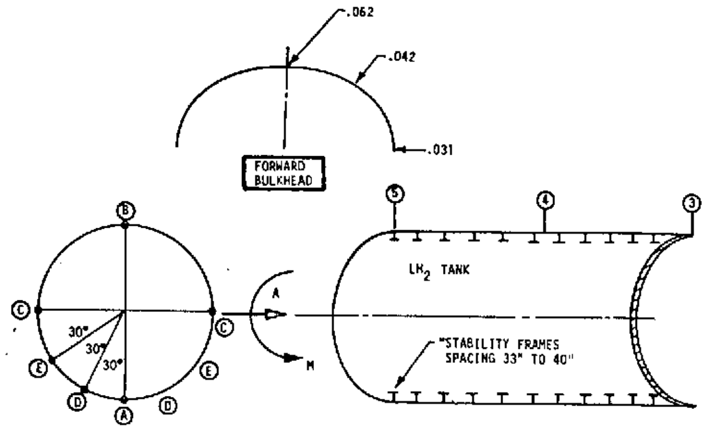
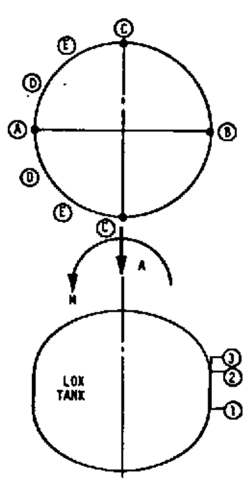
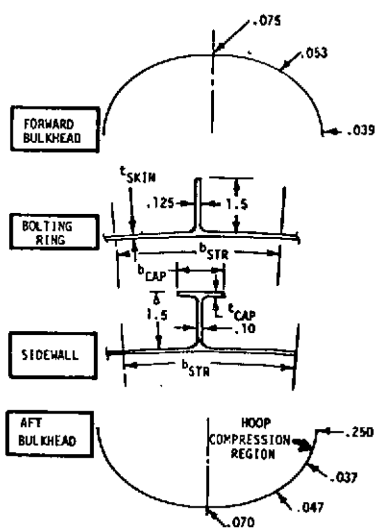


Figure 14. LH_2 Tank

The main LH_2 tank is a cylindrical 2014-T651 aluminum alloy tank 240 inches in diameter, with modified ellipsoid bulkheads as shown in Figure 14. The bulkheads are oblate spheroids with an aspect ratio of 1.5 to 1. The end bulkheads consist of preformed gores and a circular central section all butt-welded together. Two facing sheets (membrane bulkheads) are used with 3.8 inch thick heat resistant phenolic (HRP) core to form a compression bulkhead. The main cylindrical section is made up of machined plate stock butt-welded together to form a cylinder which is then chem-milled to the thickness shown on the chart between weldlands. The end bulkheads are chem-milled to the thickness shown on the chart between weldlands.

LOX Tank





MATERIALS:
LOX TANK: 2014-T6 AL
BOLTING RING: 5 AL-2.5 SN TI

LOCATION	b_{STR}	b_{CAP}	t_{CAP}	t_{SKIN} (NOM) ① TO ②	t_{SKIN} (NOM) ② TO ③
Ⓐ	4.0	1.5	.170	.095	.110
Ⓓ	4.0	1.5	.125	.090	.103
Ⓔ	4.0	1.0	.100	.076	.092
Ⓒ	4.0	1.0	.083	.072	.072
Ⓒ TO Ⓑ	5.3	1.0	.083	.072	.093

Figure 15. LOX Tank

The main LOX tank is a short cylindrical, 2014-T651 aluminum alloy tank 240 inches in diameter, with modified ellipsoid bulkheads as shown in Figure 15. The bulkheads are oblate spheroids with an aspect ratio of 1.5 to 1. The bolting ring is a short cylindrical, 5AL-2.5 Sn Titanium alloy shell 240 inches in diameter, which connects the LOX tank to the LH_2 tank. The sidewall and bolting ring are machined from plate stock and each welded into a cylinder.

Forward Skirt/IU

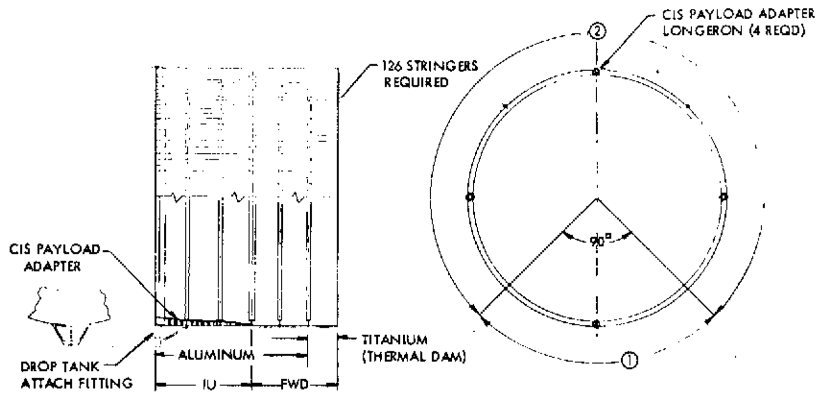


Figure 16. Forward Skirt/IU

The forward skirt/IU structure is a cylindrical shell, 240 inches in diameter and 167 inches in length (Figure 16). Aluminum external hat-shaped stringers are spaced 6 inches apart. Aluminum skin stringer construction is used except for the bay adjacent to the LH_2 tank which is titanium in order to form a thermal dam. The internal frames, also made of aluminum, are of I shaped, or channel shaped construction. Slightly smaller dimensions are required for the titanium section, based on equivalent stiffness.

Aft Skirt

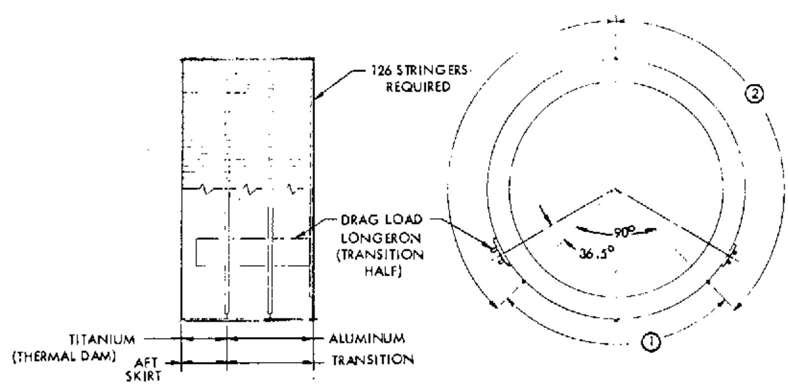


Figure 17. Aft Skirt

The aft skirt/transition section structure is a cylindrical shell 120 inches in diameter as shown in Figure 17. There are 126 external equally-spaced hot-shaped stringers. All frames are constructed of aluminum, as are stringers and skin except for the bay adjacent to tank structure, which is titanium in order to form a thermal dam. Two large tapered longerons, with local skin doubler, (the forward portion of which are located on this structure) distribute the drag loads to the shell. These drag loads are applied at the two Space Shuttle drop tank attach fittings on the propulsion unit.

Propulsion Unit

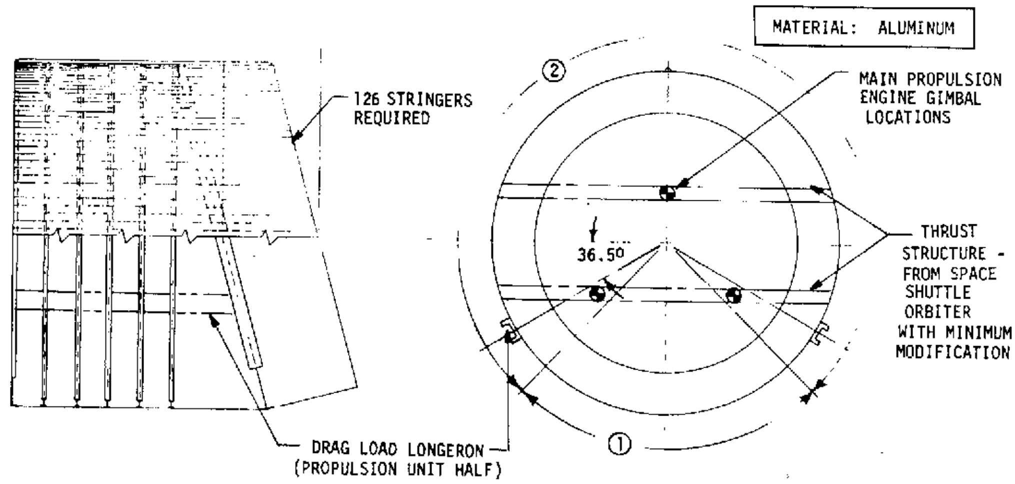


Figure 18. Propulsion Unit

The propulsion unit (Figure 18) is a cylindrical shell 120 inches in diameter with its aft end being inclined at 10 degrees. Skin, stringers, frames, and longerons are constructed of aluminum. The engine fairing is constructed of polyimide honeycomb sandwich construction.

Attach Fittings

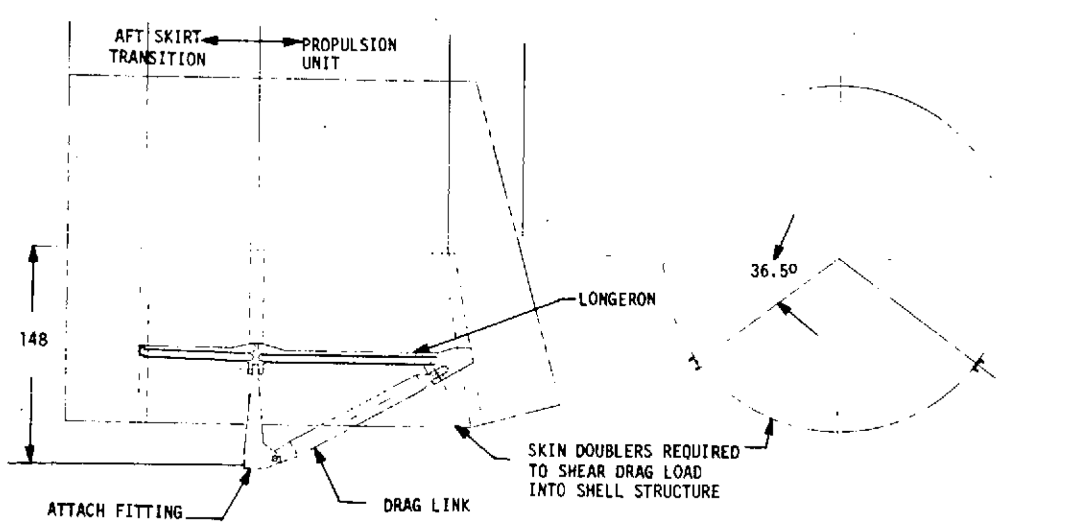


Figure 19. Attach Fittings

There are three major drop tank attach fittings on the CIS (Figure 19). The forward mount consists of a single yoke fitting attached to the IU shell structure. There are two rear mounts symmetrically located on the propulsion unit. Each mount consists of a drag tube tied to the frame, a cross-shaped member tied to the longeron, and a side load tube connected to the CIS at its bottom centerline.

Meteoroid Barrier

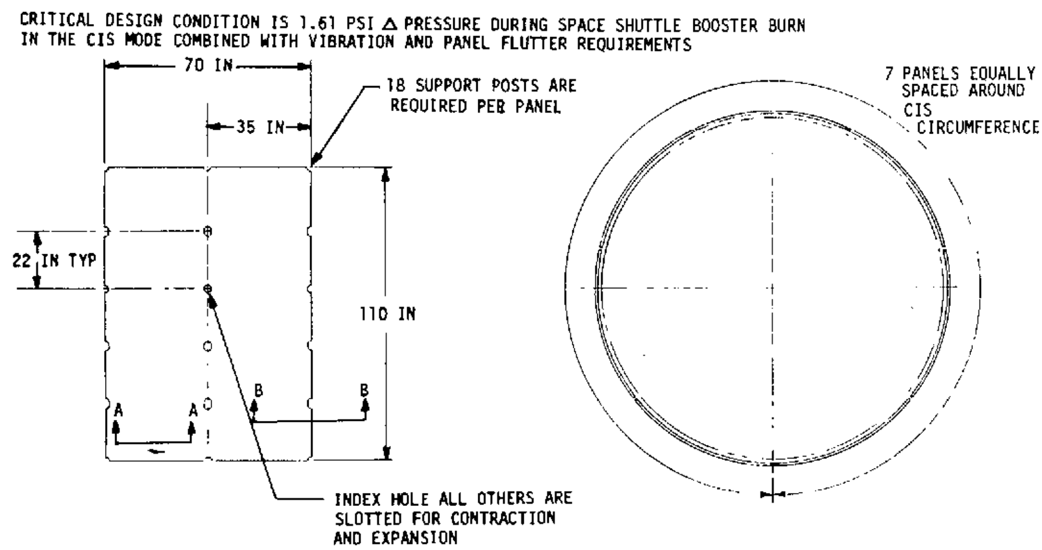
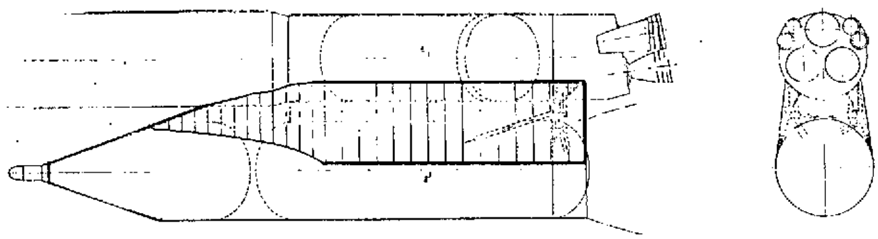


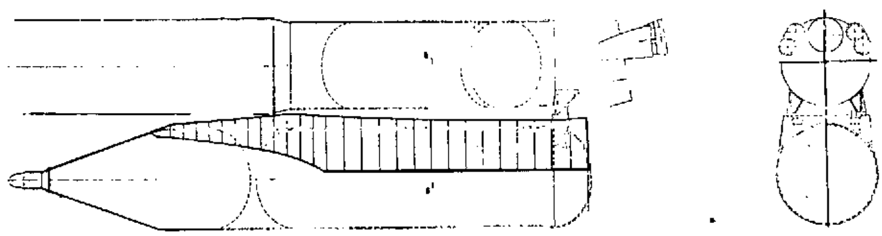
Figure 20. Meteoroid Barrier

The design of the meteoroid barrier (Figure 20) was optimized using standard procedures and equations for honeycomb sandwich construction. The study considered various facing sheet materials, core depths, support post patterns, and panel dimensions. Ground, flight, and in-orbit differential pressures, aeroshears, aerodynamic vibration and flutter criteria, and thermal environment were used in the optimization. The resulting barrier configuration is shown in Figure 20. Honeycomb sandwich construction utilizing glass polymide facing sheets and 3/4 inch core depth was selected. Selection of the support post pattern (24" x 35") was the result of aerodynamic flutter requirements considering that the panel must not flutter with one post failed. The probability of no penetration for a period of 1 year is 0.99.

Fairings



CLASS II MISSION FAIRING



CLASS I MISSION FAIRING

Figure 21. CIS Fairings

Large fairings are required to prevent aerodynamic flow between the CIS stage and the drop tank during Space Shuttle booster burn in the OIS mode. The optimization study and criteria used to define the honeycomb fairing panels and support beams is the same as that described for the previous chart. The fairings are supported off the drop tank and are jettisoned with the tank. Two types of fairings are required, as shown in Figure 21.

THERMAL PROTECTION

The thermal control problem in the CIS consists principally in minimizing the propellant boiloff resulting from heat inputs to the stage sidewall and bulkheads. The sources of heat are the solar flux and earth and lunar orbit albedo, and during propulsion maneuvers from the engines. During the OIS missions (Class II), there is considerable heat generated when the stage is boosted through the atmosphere, especially in the areas where the stage interfaces with the drop tank.

The design solutions include the utilization of high performance insulation, heat dams to minimize heat flow between the LOX and LH_2 tanks and heat protection applied in specific areas where aerodynamic heat creates high temperatures.

High Performance Insulation

A purged multi-layer insulation (MLI) system was selected as the baseline for insulating the LH_2 and LOX tanks and associated feedlines on both the CIS stage and the insulated drop tank. See Figure 22.

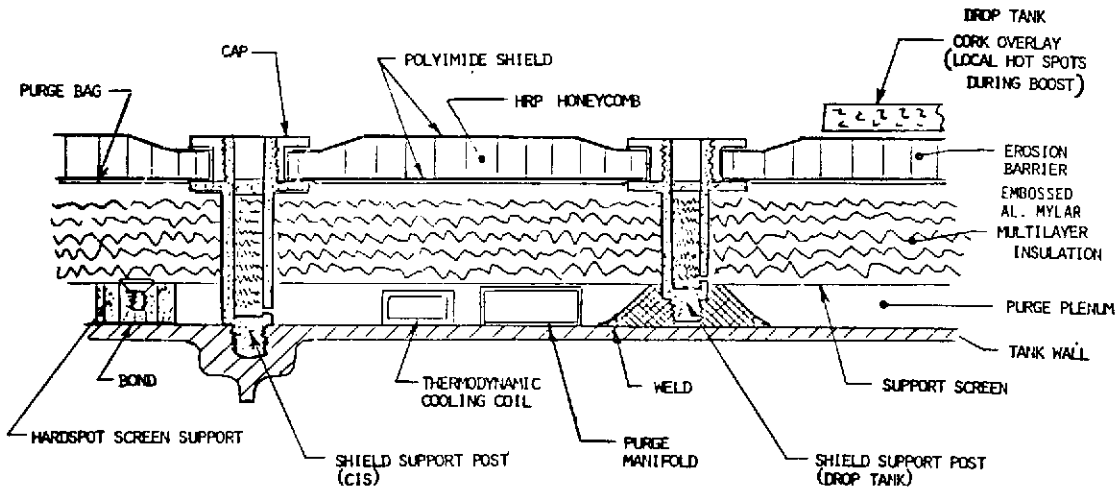


Figure 22. CIS Stage and Drop Tank Thermal Protection

Mylar was selected for the shield film in all areas except the fairing where Kapton was selected due to its ability to withstand the higher aerodynamic temperatures during launch ascent. The shields will be 0.25 mil thick and will be aluminized on one side. They will be embossed to provide proper separation between adjacent shields without use of insulative separators.

Tests at NR have shown an upper temperature limit of 140 F for Mylar when it is subjected to compressive loads approaching 0.108 PSI. An upper limit of 200 F has been demonstrated in the quiescent, no load condition. Kapton is used for temperatures above these limits.

The total MLI thickness is 2 inches on all surfaces except shielded bulkheads and exterior fairing surfaces. A 0.5-inch purge plenum between the tank and MLI affords even distribution of preconditioning purge gas flow. The MLI is supported on aluminum wire mesh screen. The outer shield, which serves as a purge container, is a 5 mil non-embossed aluminized shield.

The MLI is protected in earth environment and during launch ascent by a structural composite barrier fabricated from polyimide shields bonded to 3/4-inch fiberglass honeycomb core. Barriers are supported by hollow fiberglass posts which are threaded into tapped holes on CIS or welded hard spots on the drop tank. The hollow core of the post is filled with MLI to limit heat loss. The drop tank cone, as well as local hot spot areas, is protected by cork overlay.

Ground Purge

The MLI is protected from intruding atmospheric contamination (such as moisture, dust, and chemicals) by a ground supplied purge system. This system flows dry nitrogen or missile grade air into the CIS and drop tank insulation at a rate that will maintain positive internal pressure of approximately 0.1 psig.

Prior to the start of cryogen loading, the nitrogen and other condensable gases are flushed from the purged areas with helium. The helium purge is continued during ground cryogenic operations to maintain 0.10 psig pressure and is discontinued by umbilical disconnect just prior to lift-off.

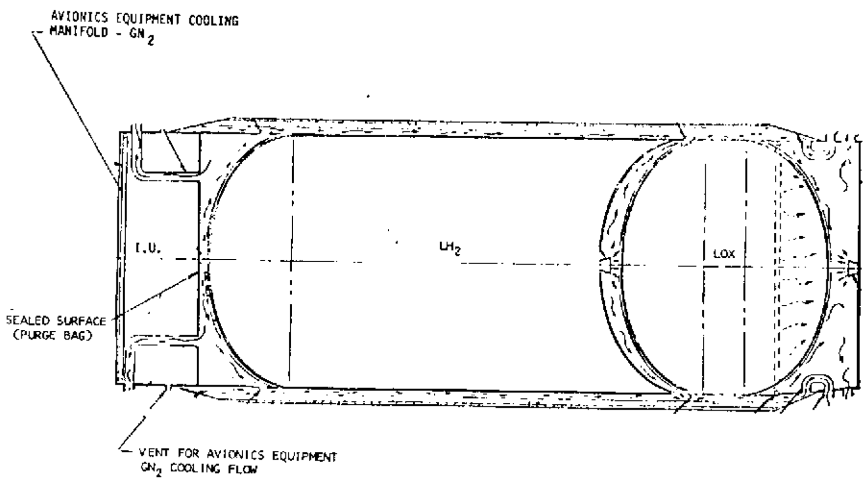


Figure 23. CIS Purge System

The CIS has four separate purged circuits: forward bulkhead, sidewall, nested bulkhead, and aft bulkhead (Figure 23). These circuits isolate areas of possible leakage and provide contamination isolation. Purge is introduced into the forward, nested, and aft bulkhead areas through diffusers and is discharged through specific outlet valves. In subsequent detail design of this system, instrumentation will be installed at these outlets for hazardous gas detection. The sidewall purge is introduced through a manifold at the top of the stage or through ports from the forward bulkhead area. It flows downward through the plenum and MLI. It is ducted through the aft skirt to the vent valves.

The functions of the purge systems on the drop tank are the same as on the CIS, but the flow pattern is different. Purge is introduced through a manifold at the top of the cone; it flows downward and around the structure, exiting into the fairing. The sidewall purge is introduced from a vertical manifold. It flows around the stage and exits into the fairing void. Purge is introduced through a diffuser into the intertank area, exiting through specific discharge valves into the fairing void.

Thermodynamic Vent System

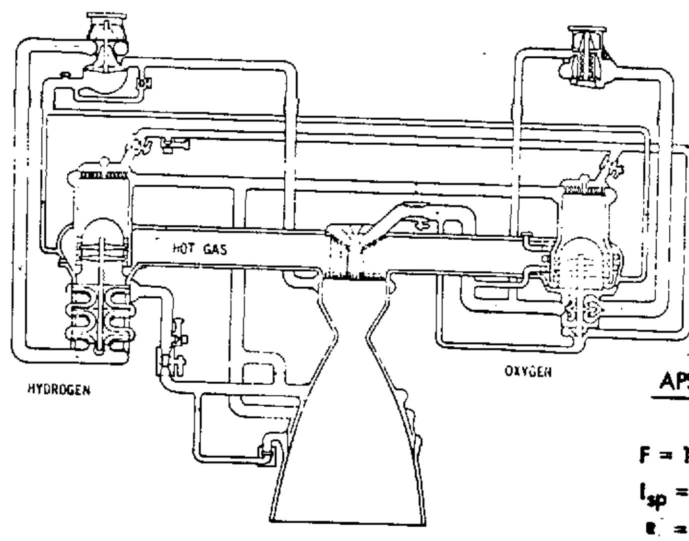
An important part of the thermal protection system are the cooling coils used in connection with the thermodynamic vent system. Since this system is primarily utilized for propellant conditioning it is described in the propulsion section.

PROPULSION

The propulsion subsystems defined for the CIS stage include main, auxiliary and attitude control propulsion plus the ancillary functions such as: pressurization; thrust vector control; propellant pressurization, orientation, conditioning, venting, transfer and management.

Main Propulsion (MPS) and Auxiliary Propulsion (APS)

MPS AND APS
ENGINE SCHEMATIC



MPS

F = 470 K LBS
I_{sp} = 454 SEC
ϵ = 90:1
EMR (NOM) = 6.0:1
EMR (RANGE) = 5.5 TO 6.5
THROTTLING RANGE = 2.0:1
P_c = 3000 PSIA
WEIGHT = 5720 LBS

APS

F = 10 K LBS
I_{sp} = 470 SEC
ϵ = 400:1
EMR (NOM) = 6.0:1
EMR (RANGE) = 5.5 TO 6.5
THROTTLING RANGE = 5.0:1
P_c = 1800 PSIA
WEIGHT = 213 LBS

Figure 24. CIS Propulsion Engines Characteristics

The engines to be used in both the MPS and the APS are high chamber pressure staged combustion LOX/LH_2 engines (Figure 24). The 470K engine is the same engine planned for usage on the Shuttle orbiter. The 10K engine is the same engine concept used in the recently completed tug point design study.

The MPS engines presently have the capability of being throttled to 50%. In the OIS mode, the three main engines will require only 15% reduction in thrust towards the end of boost. Throttling will not be required for the single main engine during TLI.

The APS with an I_{sp} of 470 seconds is the most efficient of the three propulsion systems (MPS I_{sp} = 454 seconds and ACPS I_{sp} = 391 seconds) for all maneuvers requiring an impulse greater than 0.136 x 10^6 lb-sec. However, the high impulse requirements of the OIS boost to orbit and the CIS TLI maneuvers result in prohibitively long APS burn times and associated high gravity losses. Thus, the high thrust MPS will be used for these two maneuvers.

Analysis of APS start transient and shutdown losses indicates that for impulse requirements under 0.136 x 10^6 lb-sec the ACPS is more efficient and will therefore be used for all low impulse maneuvers.

In order to satisfy the high thrust requirement of the OIS boost to orbit maneuver, three 470K engines are required. However, a single 470K engine provides sufficient thrust for the CIS TLI maneuver. Two engines are removed after arrival in earth parking orbit. Analysis of this concept showed that a savings of 58,000 pounds of propellant can be realized by using only one engine for TLI.

Since use of only one engine does not allow stage roll control, the APS (four 10K engines) will be operated in conjunction with the single engine MPS. Besides providing the required roll control, this MPS/APS parallel burn results in a propellant savings of 2,000 pounds since the APS is more efficient than the MPS.

To provide propellant conditioning during the OIS mode, systems, similar to those used on the shuttle orbiter will be employed. The LH_2 side uses pumps reverse recirculation and the LOX side uses forward natural convection recirculation with helium injection assist. These flows will be initiated prior to liftoff and remain in operation through boost until the OIS engines are started.

To provide propellant conditioning during the CIS mode, the LH_2 side will use the same system employed during the OIS mode. However, due to the low "g" environment in orbit, the LOX side will use an overboard bleed system in place of the natural convection system. The selection of these systems will minimize propellant losses without any 470K engine design changes.

Attitude Control Propulsion System (ACPS)

The ACPS provides impulse for attitude control, vehicle orientation, propellant settling, near rendezvous maneuvering, small velocity changes and docking. In addition to prepressurization of the propellant tanks, the ACPS supplies GOX and GH_2 for fuel cell reactants.

The study guidelines affecting the ACPS are essentially the same as were used in the preceding SII-CIS study. However, the subsystem design selected was required to utilize LH_2 and LOX, be capable of performing active docking, and be modular to facilitate in flight maintenance. The FO/FS requirement has been construed to mean the ACPS must be FO/FO to enable safe return of the CIS after the second failure. The ACPS was designed to satisfy these requirements. In addition, the ACPS has the capability of holding attitude with any thruster failed.

The ACPS propellants are supplied by a turbopump system which utilizes 0 NPSP pumps and refillable capillary propellant tanks located in the main propellant tanks. The turbopumps withdraw propellants from the capillary tanks, increase the propellant pressure, and force the propellants through heat exchangers to convert the liquids to gases for storage in accumulators. The stored gas is then withdrawn upon demand to perform the various ACPS functions. A thermodynamic vent system is utilized to chill the turbopumps and feedlines.

Three propellant conditioning assemblies (PCA) and component redundancy were designed into the ACPS to assure FO/FO reliability. Each PCA consists of a GOX and a GH_2 propellant conditioning system (PCS) each of which contain a turbopump and heat exchanger operated by gas generators.

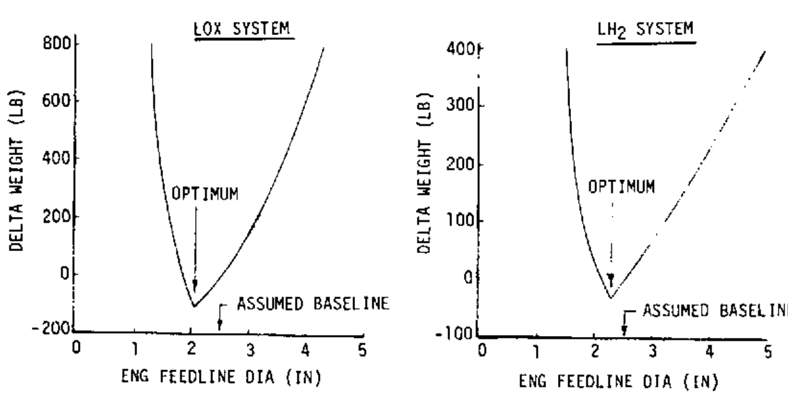
Propellant Feed, Vent and Pressurization

The propellant feed, vent, and pressurization subsystem is required to satisfy the operating requirements of the MPS and APS engines, and the ACPS turbopumps. Also, propellant tank venting requirements during ground, powered boost, and on-orbit operations are satisfied by this subsystem.

	Start and Run NPSP
MPS Engines	LOX: 8 psi LH_2: 2 psi
APS Engines	LOX: 1 psi LH_2: 0.5 psi
ACPS Turbopumps	LOX: 0 psi LH_2: 0 psi

Table 1. NPSP Requirements

The various engine start and run NPSP requirements are shown in Table 1. The ACPS turbopumps are shown with zero NPSP requirements to emphasize the fact that the ACPS subsystem requires no pressurization.



SELECTED SIZE

FEEDLINE	LOX	LH_2
ENGINE	2-1/4	2-1/4
MANIFOLD	3-1/4	3-1/4

Figure 25. Feedline Optimization

The propellant feedline diameter for the APS engines was optimized based on total vehicle weight (Figure 25). The results can be seen for both LOX and LH_2 subsystems on the respective graphs. The optimum feedline diameters are 2.05 inches and 2.25 inches for the LOX and LH_2 feedlines respectively. Therefore, to maintain compatibility among both propellant feed subsystems, engine feedline diameters of 2.25 inches were selected. This reflects a manifold size of 3.25 inches, based on constant velocity considerations.

considerations.

MPS propellants are transferred from the drop tank to the MPS engines. Pneumatic actuated prevalves, one located in each engine feedline, are utilized for drop tank isolation. The APS propellants are transferred from the CIS tanks via dual manifolds and engine feedlines. Two motor operated prevalves located in each engine feedline serve to isolate the CIS propellant tanks. Thermal isolators are located in the APS engine feedlines to reduce propellant boiloff losses.

The drop tanks are vented during propellant loading prior to liftoff. The drop tanks are not vented after liftoff unless a failure of either the insulation or pressurization subsystems results in tank over-pressurization. The CIS propellant tanks are vented during propellant loading in the same manner as the drop tanks. In addition, the CIS tanks are vented after the TLI burn to facilitate propellant transfer from the drop tanks. Any other CIS tank venting is not anticipated unless a failure of either the insulation or pressurization subsystems causes tank over-pressurization. The ACPS tanks are vented at random times throughout the lunar mission to refill these tanks from the CIS propellant tanks.

The drop tanks and CIS tanks are prepressurized prior to liftoff from ground supplied helium. Prior to either MPS or APS engine firings, except for the initial OIS mode firings, the tanks will be prepressurized from the ACPS vaporized propellant supply. Pressurization gases during MPS or APS engine firings are tapped-off of the respective engines. The ACPS propellant tanks do not require pressurization due to zero NPSH inducers on the ACPS turbopumps.

Orientation

In the weightless environment of a coasting spacecraft the orientation of liquid propellants in the propellant tanks can be impossible to predict and difficult to measure. Certain specific orientations are required in some tanks for some functions. The propellant, if not in the required location, must be acquired. Two basic means of propellant acquisition are used: artificial gravity produced by accelerating the spacecraft and capillary devices. When refilling a propellant tank in space basic problems are refill time and loss of propellant out of the vent line. When transferring propellant out of the tank to an engine a basic problem is feedout of bubble free propellant with minimum residual.

The orbital refill liquid and vent lines are one inch diameter from the refill disconnects to the point where they connect with the tank feedout lines. The ground fill lines are 8" diameter and the main engine feed lines are 8", LO_2 and 17", LH_2. If the tanks are filled in orbit through these lines childown and boiloff loss would be about 6000 lb LH_2 and 25,000 lb LO_2. By using 1" diameter orbital refill lines the boiloff and childown loss is reduced to 20 lb LH_2 and 400 lb LO_2.

The CIS tank contains instrumentation to determine if the propellants are satisfactorily located for operation of the APS engines. If the depth of propellant above the outlet is 0.5 feet or more, childown of the APS engines can be started. When childown of the engines is completed, if the propellant depth above the outlet is two feet or more, the engines can be started. If less than the required depth is sensed, 2 +x ACPS engines must be used to obtain the required depth.

The propellants are required to be at the feedout end of the CIS tanks in order to refill the ACPS tanks. This requirement is met by refilling the ACPS tanks during APS burn periods.

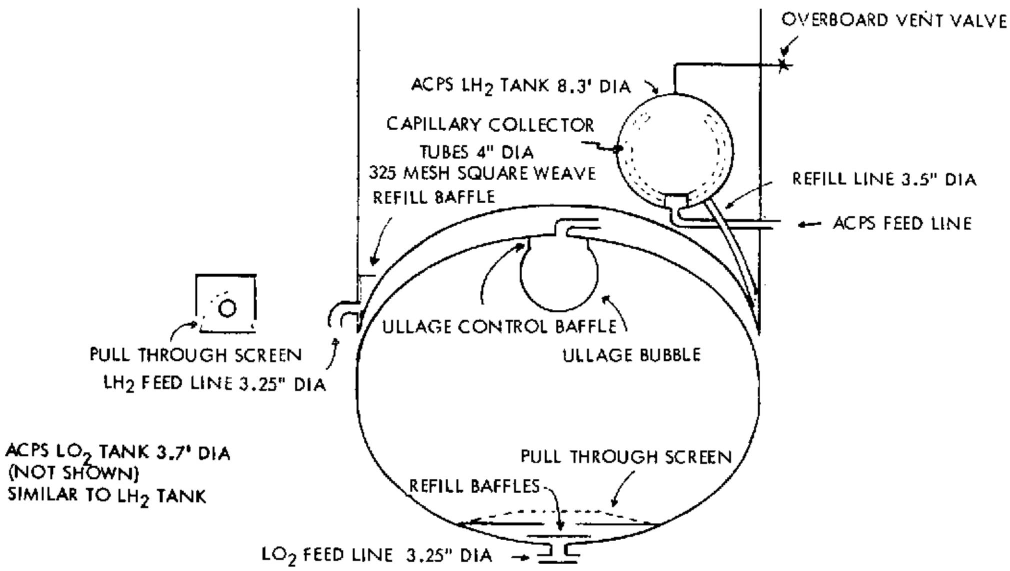


Figure 26. Feedout Concept

Pull through screens are used in the CIS tanks to ensure feedout of bubble free propellant down to the level of the screen (Figure 26). The residuals below the pull through screen are 97 lb LH_2 and 3000 lb LO_2.

Refueling in Orbit

Refueling of the CIS prior to initiation of a lunar, geosynchronous or interplanetary transfer mission is accomplished by a propellant logistics module which is delivered by the shuttle orbiter.

After the CIS tanks are full the drop tank is brought up, attached and filled. Based on the available criteria for orbital propellant tank refill it was determined that a logistics tank load of propellant could be transferred to the drop tank in nine hours and the propellant remain collected at the inlet end of the tank. If a transfer time less than about 6.5 hours is desired, inlet baffles will be required. The inlet baffles considered substantially increase the residual. While the residual could be transferred to the CIS a more complex system of capillary collectors would be required and higher flow rates would make design of the transfer provisions more difficult.

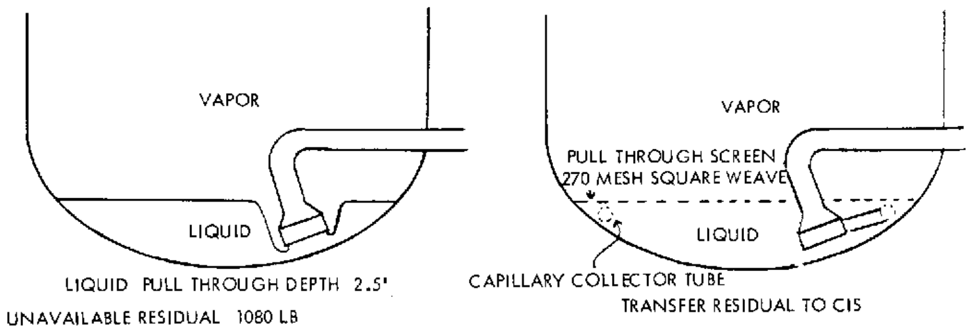


Figure 27. Drop Tank Pull Through

When feeding out of the drop tank to the main engine the gas-liquid interface will become distorted as shown on the left side of Figure 27 and gas will enter the feedout line when the liquid depth is 2.0 feet. In order to avoid this occurrence the low level sensor was set at about 2.5 feet. The resulting residual is 1080 lb. By using a pull through screen shown at the right, liquid can safely be fed out down to the 2.0-foot level. The residual, 870 lb, is retained by the screen and can be transferred to the CIS tank. A similar arrangement is used in the drop tank LO_2 tank to retain and transfer 14,200 lb of LO_2 to the CIS.

Thermodynamic Vent

The combined thermodynamic vent and cooling coil system is a novel approach to the reduction in overall propellant losses in the CIS and drop tank. The fluid (gas or liquid) is vented from the top of the hydrogen tank. The gas goes through a control orifice or valve and then through a metal tube attached to the outside of the hydrogen tank. Since the pressure in the tube will be less than the pressure in the saturated tank, any liquid in the tube will vaporize with the heat of vaporization coming from the fluid and the outside tank wall.

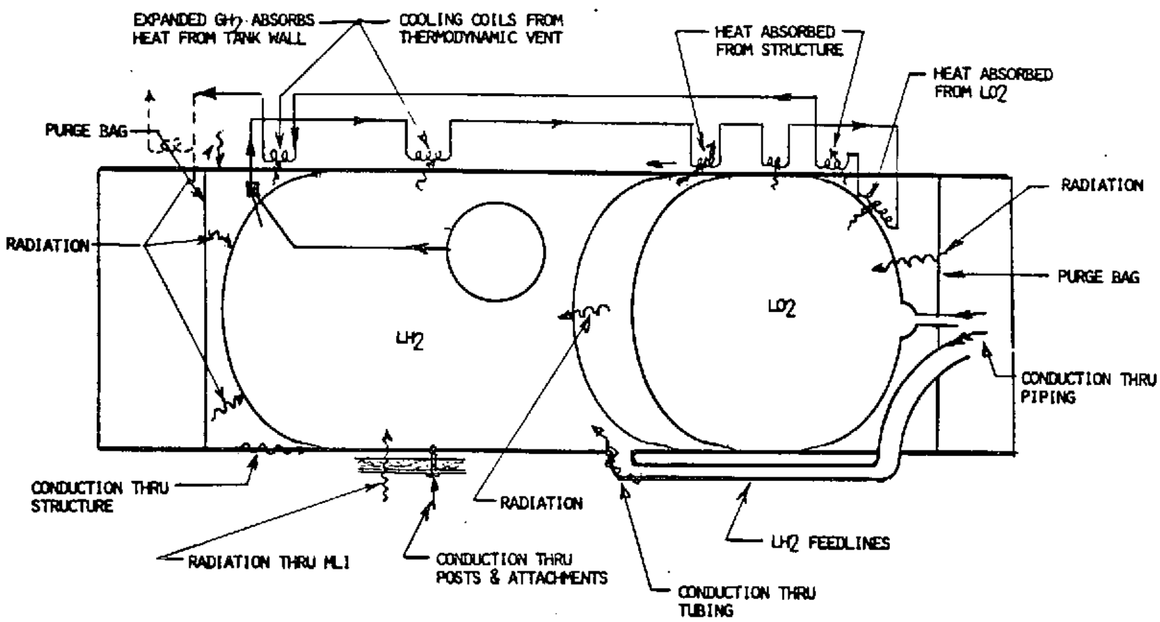


Figure 28. Thermodynamic Vent and Cooling Coil

The position of the fluid in the tank is not controllable under all conditions. For this reason, the system is recommended with venting of gas from the top, but with a system that can accept liquid. The gas (liquid) goes through coils on the side of the hydrogen tank and only gas at a slightly reduced pressure would leave the hydrogen tank wall. This completes the normal thermodynamic vent system/cycle.

The gas is then passed through two coils fastened to the inside of the bolting ring between the LOX and the hydrogen tank. These coils act as a reverse flow heat exchanger for the removal of heat from the metal as the heat flows into the hydrogen. This reduces the boiloff of hydrogen and at the same time can be used to cool the oxygen and prevent oxygen boiloff. The gas, after leaving the oxygen tank coils, is used to cool the aft skirt so as to prevent extra heat from entering the LOX tank and then cools the forward skirt and the instrumentation compartment.

For each 2 pounds per hour of hydrogen (the suggested minimum rate) that is vented, a total of 384 BTU/hr is removed from the hydrogen tank, 630 BTU/hr is removed from the LOX tank, 500 BTU/hr are available for removal of structural heat and 1340 BTU/hr are available for other structural or instrumentation compartment cooling.

The thermodynamic venting hardware weight is 376 pounds for the CIS and 412 pounds for the drop tank. Analyzing the extra system weight versus the economies in propellant over a lunar program of 20 missions in ten years discloses that there is a net total savings of 372,600 pounds of propellant. In terms of shuttle flights, this is equivalent to seven flights, thus saving $70 million.

The thermodynamic balance for the CIS stage during a translunar coast is shown in Figure 29.

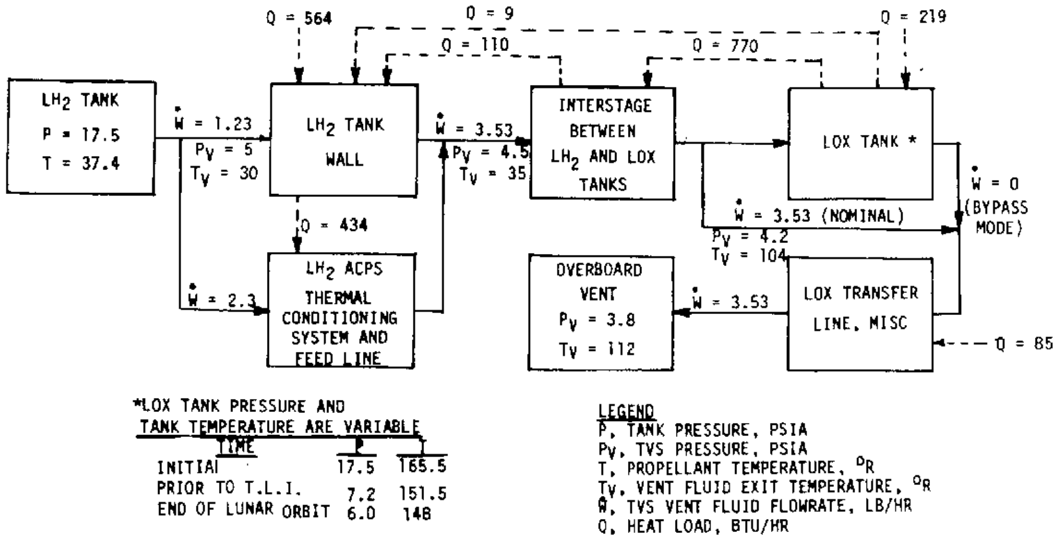


Figure 29. Thermodynamic Balance

Electrical Power Source

The electrical power source generates electrical energy for all CIS subsystems. For space applications several different devices may be used to supply electrical power. The most common power sources currently in use are fuel cells, primary batteries, and solar arrays. For certain special applications, radioisotope thermoelectric generators are used (RTG's).

Peak powers occur during engine burn and are for short durations. The power source is normally sized for the longest sustained average power and peak powers are obtained by topping with secondary batteries. The longest sustained average power is for translunar coast. In order to allow for power growth (payloads, refurbishment), a design level of 2.5 KWe is selected for power source sizing.

Five different power sources were evaluated based on solar arrays, fuel cells or a combination of the two. Table 2 summarizes electrical power source weights and cost for three years of operation based on a first mission time of 225 days and four subsequent 164-day missions with quiescent periods in between. Times include refurbishment in earth orbit and round trips to the moon.

Two different fuel cells are considered, one based on the NASA Lewis advanced shuttle fuel cell technology program and the other based on the current shuttle design. The qualification life for the shuttle fuel cell is 2000 hours with a development lifetime of 5000 hours. A cost increment of $6 million is estimated for development from 2000 hr certification to 5000 hr service life. Fuel cell reactant weight is based upon 4390 KW hrs per mission, one third being required for orbit dark periods. Main engine propellant penalty is based on 2-1/2 times consumable weight and 5 times inert weight.

POWER SOURCE / ITEM	SOLAR ARRAY / BATT	FUEL CELLS	FUEL CELLS	SOLAR ARRAY / FUEL CELL	SOLAR ARRAY / FUEL CELL
FUEL CELL TYPE		ADV. SH	SHUTTLE	ADV. SH	SHUTTLE
WEIGHTS, LBS					
OP. POWER SOURCE	1,980	571	976	709	967
RESUPPLY	400	258	2,424 (4)	0	402 (4)
SUBTOTAL	2,380	829	3,400	709	1,369
CONSUMABLES					
FUEL CELL REACTANT (1)	0	18,275	18,275	6,090	6,090
MAIN ENGINE PROPELLANT PENALTY (3)	49,500	59,960	70,085	37,950	39,400
ATTITUDE CONT. PROP. PENALTY	(2)	0	0	(2)	(2)
SUBTOTAL	49,500	78,235	88,350	39,040	45,490
TOTAL WEIGHT, 3 YEARS	51,880	79,064	91,750	39,749	46,859
COST $ MILLIONS					
NON-RECURRING	8.76	14.64	3.50 (4)	17.31	6.11 (4)
RECURRING FIRST UNIT	7.31	1.12	1.50	2.05	2.26
SUB SUBTOTAL	16.07	15.76	5.00	19.36	8.37
RESUPPLY (6)	.17	1.02	5.10	0	.85
SUBTOTAL	16.14	16.78	10.10	19.36	9.32
LAUNCH COST AT $200/LB (5)	10.37	15.81	18.35	7.94	9.35
TOTAL 3-YEAR COST	26.51	32.59	28.45	27.30	18.67

(1) NO ALLOWANCE FOR BY-PRODUCT WATER CREDIT
(2) NOT DETERMINED, BREAKEVEN WITH ADV, SHUTTLE VARIES 0.76 TO 1.47 LB/HR
(3) 5 TIMES INERT WEIGHT AND 2-1/2 TIMES REACTANT WEIGHT PER MISSION
(4) ASSUMES SHUTTLE PROGRAM DEVELOPS 5000 HR FUEL CELL
(5) BASED ON 3 YR MISSION WEIGHTS
(6) NO ALLOWANCE FOR INSTALLATION CHARGES AND SHIPPING STRUCTURE

Table 2. Power Source Evaluation

The fuel cells selected for the CIS are based upon the NASA Lewis advanced Space Shuttle fuel cell development program. This is scheduled to be a multi-year effort (1976-68 technology) and is approximately one year old at this time. One of the goals of this program is a fuel cell operating life of 10,000 hrs. This may be compared with proposed shuttle fuel cell 2,000 hr qualification lifetime and 5,000 hr development lifetime. The shuttle fuel cell was rejected on the basis of weight and lifetime. The NASA Lewis technology developed will be incorporated into a 7 KW breadboard fuel cell design for system evaluation. Technology development must yield a state-of-the-art that can be scaled down from a 7 KW size to a 2.5 KW fuel cell. Availability of the fuel cell selected for the CIS depends on the continuation of NASA support of this technology.

The primary driver for fuel cell selection is the readily available reactants from the CIS. The fuel cells consumption for an entire mission is about 0.4% of the propellant required by the CIS propulsion subsystem.

Thrust Vector Control (TVC)

The selected configuration for the hydraulic TVC system to position and gimbal the CIS engines, consists of two separate hydraulic fluid loops. One loop consisting of pumps and reservoir provides fluid power to the actuators on the four APS engines and to the main engine used for translunar injection. The other loop with its own pump and reservoir provides fluid power to the actuators on the two main engines which are removed in orbit and returned to earth.

Main engine actuators are supplied with hydraulic power from the engine driven pumps during engine firing. Prior to engine start, one of the GH_2 driven pumps is activated to provide initial power. The APS engine actuators are supplied from one of two electrically driven pumps.

Propellant Management

All propellant gaging is based on settling of propellants with a small acceleration (10^{-5}g) in order to attain 1% accuracy. Primary effect of propellant utilization (PU) is in terms of increased mission reliability and crew safety in the event of propellant losses due to malfunction, meteoroid impact or insulation damage. PU is not provided for the drop tank, since its residuals are transferred to the CIS stage prior to drop tank jettison.

AVIONICS

The CIS mission presents some challenging requirements to the avionics subsystems designer. Some of the tasks that the avionics subsystem must perform as well as interfacing requirements are indicated in Figure 30. Principal avionics design drivers were the autonomy and the fail operational/fail operational (FO/FO) requirements.

The subsystem selected provides FO/FO capabilities by means of triple redundant components, and exhibits autonomous operation except for navigation during translunar transfers where it required some ground support.

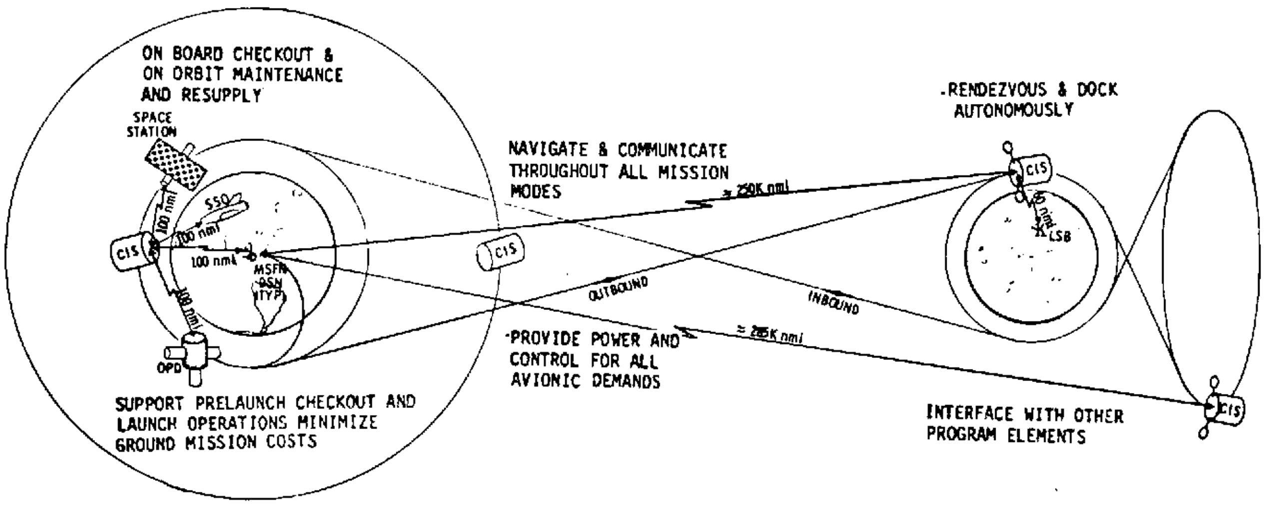


Figure 30 - CIS Avionics Tasks

The basic subsystems (Figure 31) established were power, control, and data management; guidance, navigation and flight control, communications, rendezvous and docking, propellant management, data acquisition and formatting, electrical controls, and avionic equipment installations.

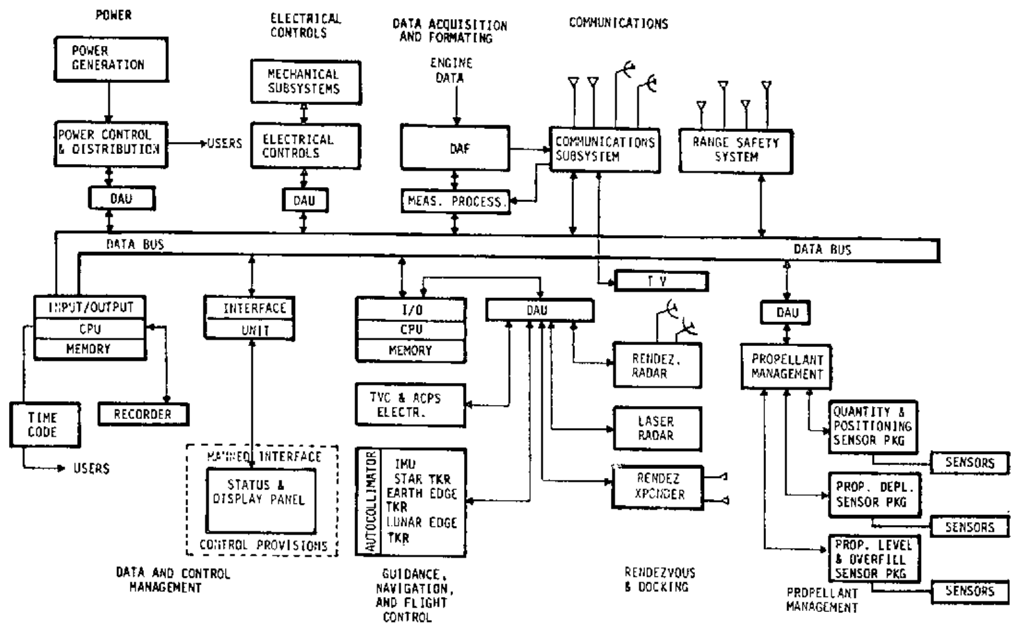


Figure 31 - CIS Avionics Block Diagram

Since autonomy and redundancy were imposed on the avionic subsystems, a highly flexible, integrated, computer-controlled subsystem was required. In conjunction with numerous sensors and dedicated computational capability for navigation, the central data bus services all subsystems elements.

Adopting this approach provided the capability to sequence commands to monitor responses and to reconfigure automatically to meet mission contingency requirements. Also provided is the capability to accept up-link data which can modify the on-board, stored, predetermined software programs.

Electrical Power Profile

The electrical power profile (Figure 32) is based on the estimate of power at launch and continues through CIS main engine burn to earth orbit. The profile also includes an estimate of the power for the Translunar Insertion (TLI) events. The LH_2 recirculation motor driven pumps will be required to condition the main engine prior to TLI burn. The TVC (Thrust Vector Control) 5KW motor driven hydraulic pump will be required for gimballing the four APS engines during the TLI burn. Referring to the power profile, only the main events are shown which are required for operation of the three main engines during boost and one main and four APS engines during TLI.

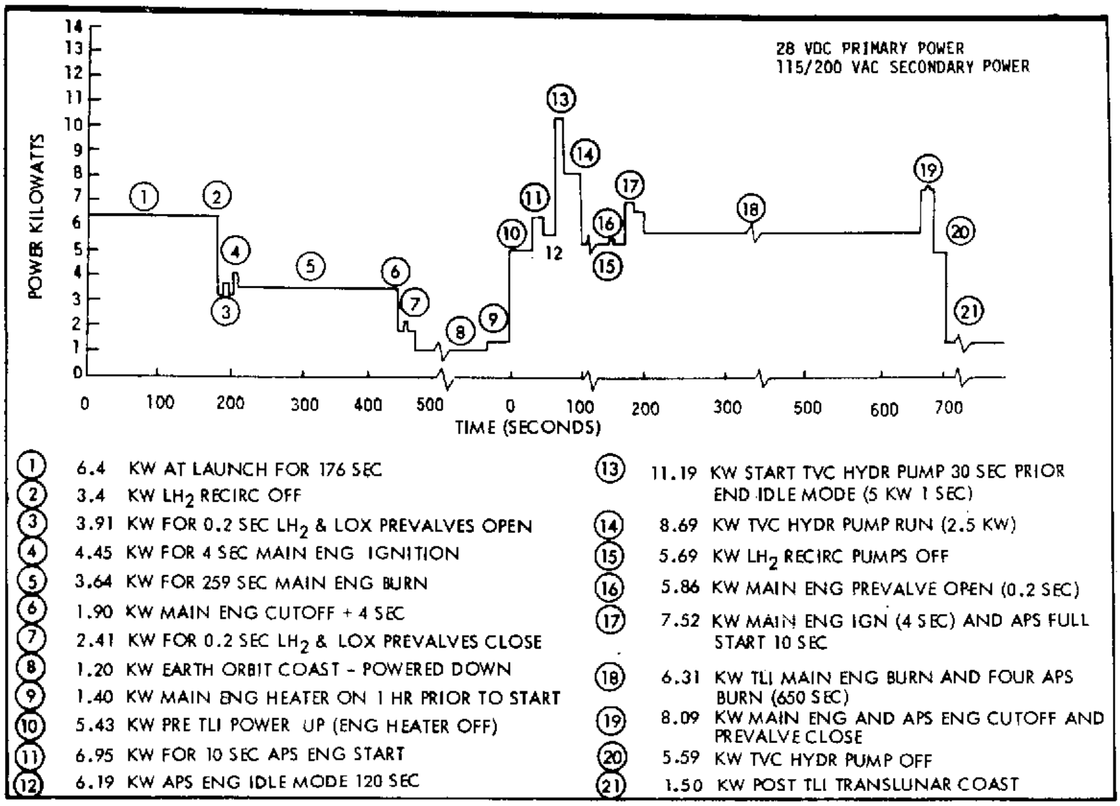


Figure 32 - Electrical Power Profile

Peak power will be approximately 11.2 KW during starting of the 5 KW TVC motor driven hydraulic pump used to gimbal the APS engines. The CIS electrical load profile during TLI burn incorporates the power requirements for four APS engines during start, idle mode, full thrust operation and engine shutdown transient.

Electrical Power and Control Subsystem

The Electrical Power and Control Subsystem distributes, sequences and controls the electrical energy necessary for the CIS vehicle to accomplish a 45-day mission (10 day pre, 25 lunar, 10 day post) and up to 180 days of earth orbital quiescent coast with logistical resupply of consumables and opportunities for on-orbit maintenance.

Primary power	28 VDC fuel cell
Secondary power	115/200 VAC, 3 Ø, 400 Hz static inverter
Average	1.5 kilowatt
Peak	11 kilowatt
Mission energy (45 day)	1647 kilowatt hours

Three 2.5 KW fuel cells will be utilized to provide 28 VDC primary power. Three 5 KW solid state static inverters will provide the 115/200 VAC, 3 Ø 400 hertz power which is required to drive electrical motors, main engines, LH_2 recirculation pumps, hydraulic pumps and AC motor driven valves.

Electrical power will be distributed by utilizing three main power buses, three power transfer switches, various sub-buses, and power switches. Cross-strapping of busses will be used during ground checkout, on-orbit checkout, and component fault isolation. During normal operation, each fuel cell and associated bus will be isolated and independent of each other.

Nickel cadmium 28-VDC batteries with associated chargers will be provided for emergency operation and peak loading as required. The batteries will be connected to the three computers in such a manner that in the case where primary 28-VDC is removed from a computer, the battery will automatically come on line to provide power to that computer. This will allow disconnection of a malfunctioned fuel cell and cross strapping of the buses and dc loads to the remaining two fuel cells.

A combination single point and chassis grounding system will be used for the CIS vehicle. Structures return will be used for solenoid operated valves and single point ground will be utilized for low level measurements where noise pickup may pose a problem.

Data and Control Management Subsystem

The Data and Control Management subsystem provides the CIS with the intelligence coupled with the vehicular integration and control capability necessary to enable the CIS to perform as an autonomous entity. The DCS subsystem configuration baselined for the CIS vehicle consists of three major elements: (1) computer complex (central control and GN&FC), (2) data bus with signal distribution and acquisition components, and (3) flight software. The central computer complex provides the centralized vehicle intelligence for control of vehicle subsystems and associated data. The data bus element provides the vehicle communications link between the computers, and the vehicle subsystems and external interfaces. The signal distribution and acquisition components provide the means of transmitting data within the DCM subsystem and to or from external elements as required. The software element includes all computer programs required for checkout, vehicle control, mission sequencing, and configuration management of the CIS vehicle.

Central Control Computer

The central control computer consists of triple redundant, stored-program general-purpose, digital computers designed to meet real-time high speed applications. Each computer consists of a central processing unit with dedicated memory and input/output controller. These computers perform the task of integration and overall control of the CIS vehicle. These tasks consist of the action and operations associated with:

Vehicle sequence and control
Redundancy Management
On-board Checkout
Data handling except for GN&FC sensors

Dedicated Guidance, Navigation and Flight Control Computers

The GN&FC computers are general purpose digital computers designed to meet the computational requirement of the CIS in the performance of the functions of Guidance, Navigation and Flight Control (GN&FC). The GN&FC computers are configured in a functional path orientation. Each functional path consists of a GN&FC computer, GN&FC sensors, attitude and thrust vector control driver electronics and the data bus elements required to integrate the segments of the functional path.

Serial Digital Intravehicular Command/Response Transmission

To accommodate the data traffic between the DCM computer complex and the other CIS subsystems, the line sharing technique commonly referred to as a data bus system was adopted. The data bus is comprised of a data transmission media, interface units, data acquisition and distribution units, and a status and control panel. For the CIS the data transmission media is a twin axial, twisted shielded jacked cable configured to operate as a balanced pair terminated at each end.

Interfacing Elements

To provide for the coupling of data on and off the transmission media, interface units are provided (Figure 33). The interface unit (IU) provides a serial digital input/output interface with the data bus and a parallel digital input/output interface for data to/from data acquisition units or other peripherals.

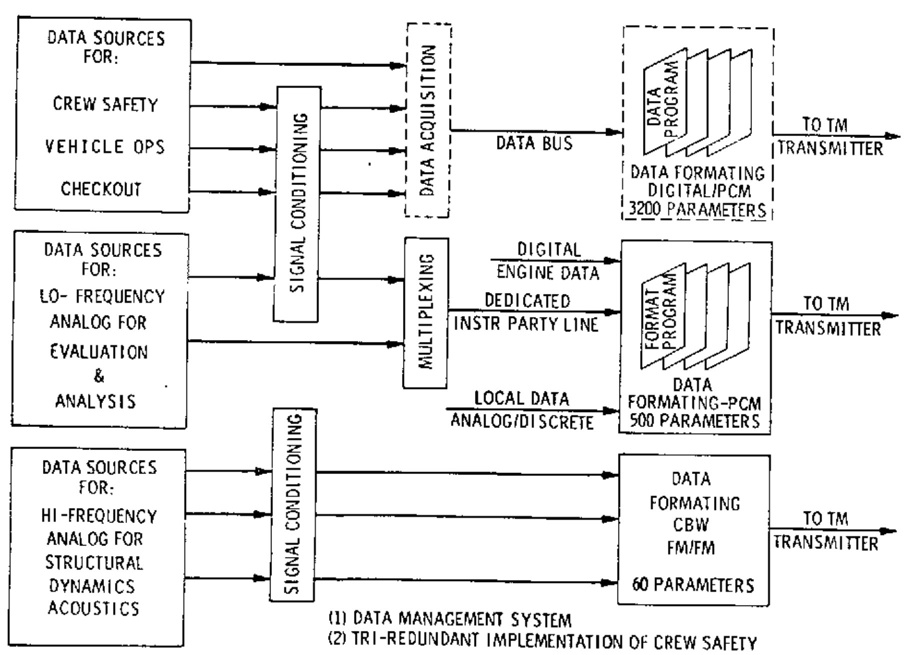


Figure 33 - Data Management Interface Unit

Data acquisition units (DAU) are provided as a data bus element to sample analog, discrete, and serial digital signals from the vehicle subsystems and processing these data for subsequent transfer to the control computer on command. The DAU also provides a means of outputing discrete, analog and serial digital signals to the vehicle subsystems under direct control of the computer controlling that functional path.

The measurement processor is another type of data acquisition and formatting device implemented as a data bus element. This unit is utilized to provide the means of acquiring and formatting subsystem parameters for outputing by the communication subsystem. This unit also provides the means of interfacing the up-link data with the data bus for access by the central control computer.

Recording and Storage

In addition to the interfaces provided for the vehicle and systems, the central control computers are provided a direct interface with recording elements which provide for the storage of the following:

Real time events data history
Malfunction history
Non-real time programs
Selected vehicle parameters

Guidance, Navigation and Flight Control Subsystem

The GN&FC, together with appropriate interfacing subsystem elements (DCM, main engines, APS, ACPS), and available ground-based aids (MSFN), provides the capability to determine the position, velocity and inertial attitude of the CIS from launch through all mission phases (lunar orbit and return). It will also provide steering commands and attitude control from booster separation through all mission phases. These phases include thrusting modes (main engines and APS) and non-thrusting modes (attitude maneuvering and holds using the ACPS).

GN&FC operations can be divided into five distinct types of phases. First is the boost and LEO insertion phase. This phase requires only the inertial reference system to control vehicle attitude and maintain the vehicle in the proper boost trajectory. The second phase includes both lunar and earth orbit operations which impose similar requirements on the GN&FC. During these orbit periods, the active instruments include the horizon edge tracker for determining the geocenter, the star tracker for accurate inertial attitude determination, the IMU for maintaining short-term knowledge of relative attitude and for measuring velocity changes, the autocollimator for measuring relative alignment in two axes between the IMU/star tracker mounting base and the horizon tracker base, and the gimbal servo amplifiers and ACPS control drivers.

The third type of phase (TLI and TEI), like the first phase, is a thrusting phase and requires only the IMU for control. During portions of the trans-lunar and transearth coast phases, the IMU and a horizon tracker will be required for navigation updates. During the balance of the coast period, the IMU data will be updated by star tracking and by the MSFN. MSFN tracking and update are recommended due to inaccuracies inherent in measuring the range and direction to the moon at great distances. The last type of phase - LOI and EOI - is similar to other thrusting phases in that control of the vehicle requires only the IMU for reference purposes during main engine braking periods.

Rendezvous and Docking Subsystem

The rendezvous and docking subsystem (R&DS), in conjunction with other subsystem elements, provides the CIS with the capability to locate a selected target lying within 400 n mi and to perform autonomous maneuvers to permit stationkeeping and/or docking operations. In addition, the subsystem provides the capability for man to perform visually-assisted docking operations. An L-band transponder is included in the R&DS to facilitate the location of the CIS by other vehicles.

Functional operations are divided into three distinct phases with each phase a function of range. The first involves the detection of the target at a maximum range of 400 n mi. The microwave radar system provides the DCM with necessary data to control the approach of the CIS to within approximately 1000 feet of the target. The second phase requires accurate range and rate control as the CIS begins to move nearer to the target. The laser radar system provides various datum to the DCM for controlling the final phase of the approach to the target - down to 100 feet. At this point the television subsystem is provided to allow inspection of the target prior to attempted docking. The laser radar subsystem provides data necessary to perform the docking function autonomously. The last phase involves docking with a target that, due to predictable or unforeseen circumstances, requires that the CIS be the active element. The television and lighting subsystems, along with the laser radar subsystem, is provided to allow docking with man-in-the-loop control.

Communications Subsystem

Mission requirements dictate the need for reliable communications at various distances such as low-earth orbit, geosynchronous and lunar orbits. Communications compatibility with CIS external interfaces such as the MSFN, Space Shuttle Orbiter, Space Station, Space Tug, etc., is required throughout all mission phases. Commonality factors with external interfaces fixed parameters such as frequency ranges, bandwidths and data rates. Video transmission capability to transmit imagery data from low-earth and lunar orbits during CIS/Payload docking operations will be provided to permit man-in-the-loop control. Communications during low-earth orbit will be limited to line-of-sight conditions since continuous ground coverage is not a study requirement. Range communications beyond 10,000 n mi from earth will be available continuously.

To insure range safety throughout the initial launch phase of CIS missions, an onboard Range Safety System will be provided in compliance with the Eastern Test Range requirements.

Propellant Management Subsystem

The Propellant Management Subsystem (PM) provides the capability to measure the quantity of propellants in the cryogenic tanks and provide capability to monitor the propellant at low levels to effect engine cutoff or prevent start of engine at a predetermined level. Accurate monitoring and measurements can be achieved with the selected subsystem during ground operations and during thrusting modes when propellants are settled to the tank bottoms.

During non-thrusting zero gravity modes, the DCM subsystem will provide the capability to calculate boiloff losses, hence will maintain accurate propellant quantities. The DCM subsystem will utilize the vent valve position measurements as well as tank pressures and temperatures to calculate losses. The data in the computer program are to be updated during each thrusting operation.

Data Acquisition and Formatting Subsystem

The data acquisition and formatting subsystem encompasses the electrical/electronic hardware and interconnecting electrical wiring networks required for vehicle subsystems operation/control data acquisition and data formatting. The electronics formatting of data will be accomplished to the extent that data signals are compatible with the proposed RF transmission/communication links and on-board data management system.

The data acquisition and formatting system has the following performance characteristics:

*Measurements capacity	4000
Maximum frequency response	8KHz
Number of down link outputs	5
% measurements under software control	95%
Accuracy of system	3%

*Measurement capacity may be varied by adding or deleting subsystem buildup blocks such as data acquisition units, computer core memory, remote multiplexers, data formatting central units and CBW frequency multiplexers.

Avionic Equipment Installation

The avionics equipment will be located in an area dedicated to the fuel cells, and avionics equipment installation. This area is located in the forward end of the CIS stage. The area is approximately 20 feet in diameter and 7 feet in length. It is an all aluminum structure with 2 intermittent frames 6 inches in height. The equipment environment within this area will be controlled to a minimum of 40 F and a maximum of 70 F during orbit.

Antennas and other field of view units are located in the avionics module. They are placed strategically around the circumference of the vehicle. Antennas will be flush mounted. Special design considerations are made for certain units to attain the proper field of view. The horizon trackers, and the star trackers are to be installed on panels which will deploy while on orbit. The parabolic and microwave antennas which are furlable and steerable will be stowed in tubes prior to use. When in use they will be deployed and remain so during the entire mission. The laser radar shall be located in the forward portion of the avionics area to allow a forward view.

WEIGHTS

The weights of the CIS stage when utilized for Class I (CIS) of Class II (OIS) functions are summarized in Table 3.

Table 3 - CIS Weight Summary

	CLASS II	CLASS I
STRUCTURE	26569	27369
INDUCED ENVIRONMENT PROTECTION	3677	3677
PROPULSION (MAIN AND AUXILIARY)	28685	16210
EQUIPMENT AND ELECTRONIC SYSTEMS	4738	4738
CONTINGENCY (3%)	1931	1606
DRY STAGE	65600	53600
RESIDUAL PROPELLANTS AND PRESSURANTS	5141	5141
USABLE PROPELLANTS	20029	301709
GROSS WEIGHT	90770	360450
DROP TANK - DRY	64616	69300
RESIDUAL PROPELLANTS AND PRESSURANTS	10803	10803
MAINSTAGE PROPELLANTS (INCL RESERVES)	812855	664340
LAUNCH WEIGHT	979044	1104893
MASS FRACTION	0.85	0.874

MISSION ANALYSIS AND PERFORMANCE

CIS Performance for Class I Function

The design lunar mission (Class I) is based on delivering a maximum of 320,000 pounds to low lunar polar orbit and returning to earth with 16,800 pounds in Mode DT-1. Departure is from a CIS assembly orbit (180 n mi x 55°). A selected departure epoch in January 1985 resulted in a design mission enabling return at the first opportunity (lunar stay about six days) and a 24-hour launch window (Figure 34). This resulted in a CIS vehicle requiring approximately 302,000 pounds propellant capacity, and in turn, requiring about 650,000 pounds of propellant in the drop tank for use during the departure to the moon. The drop tank is jettisoned after TLI. If an earth retrieval tug is available, Mode DT-2 may be flown with a requirement of about 221,000 pounds propellant in the CIS stage and 588,000 pounds in the drop tank.

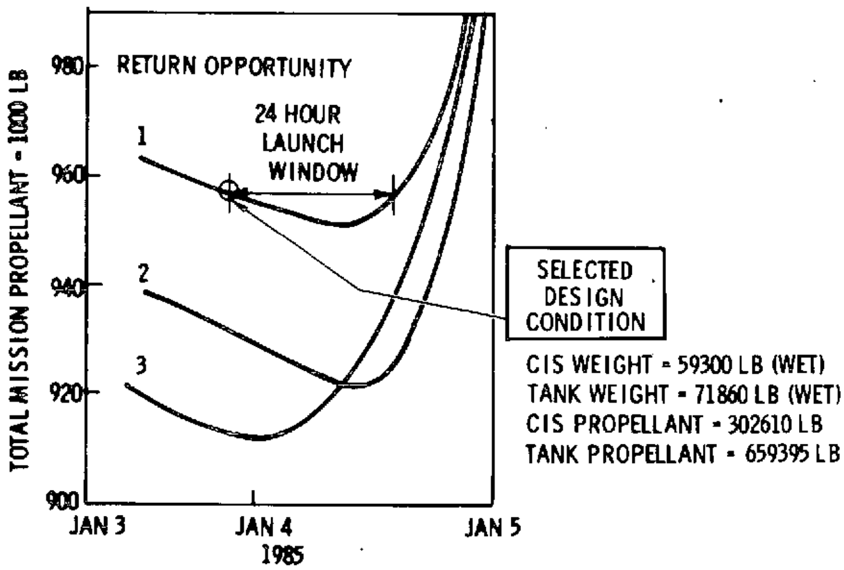


Figure 34. Orbital Departure Window
One and One-Half Stage CIS Mode DT-1

The reduction in necessary CIS stage and total mission propellants for reduced out-bound payloads is shown in Figure 35 by the solid line curves. If a reasonable scaling law is assumed, a CIS vehicle designed to match the lower payload requirements is shown by the dashed curves. Since the CIS stage is relatively small, and the major portion of mission propellants is carried in the fixed-size drop tank, it is evident that only slight pnealties accrue from designing for large payloads. For example, if the nominal payload were reduced from 320K pounds to 120K pounds (a 62.5 percent reduction), only 5.9 percent reduction in propellants could be realized with a stage specifically designed for the smaller payload in comparison to the nominal stage sized for the larger payload. This indicates that if a reasonable lunar exploration program is

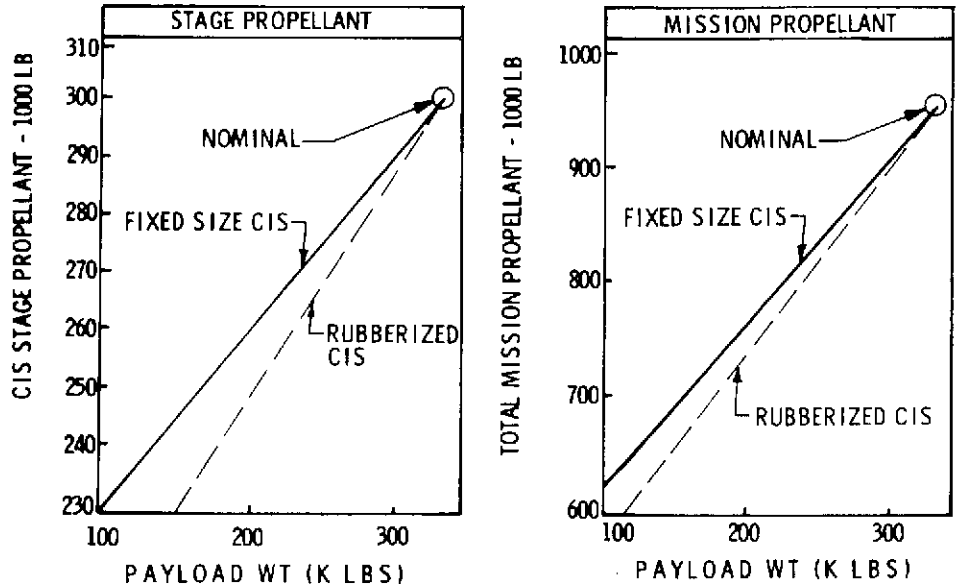


Figure 35. Payload Off-Load Penalty

envisioned for the future, it would be wise to design the CIS stage for the larger payload, even though it would be used offloaded for the near-term missions.

Examination of the effects of varying certain vehicle and mission parameters for a fixed CIS stage weight (Table 4) provides a set of performance partial derivatives that may be used for evaluation of the mission sensitivity to these parameters.

The size of the CIS concept enables a single 470,000-pound thrust main engine to suffice for earth departure while the remainder of the CIS mission is powered by the four-engine APS system. Thus, a high mission reliability and abort capability are possible due to the redundancy of the multi-engine propulsion system. Malfunction of one engine would tend to increase gravity lossess during burn near the earth or moon but not otherwise seriously compromise the mission.

Abort of the CIS mission (Figure 36) from lunar orbit is feasible at any time if a crew module of 15,000 pounds weight is assumed to be the return payload. A finite residual propellant may be present under the worst return condition. Abort return is in the plane of the moon's earth orbit to minimize total ΔV; thus a retrieval of the CIS and its passengers may be required from orbits of 18 to 29 degrees inclination, depending on the return date.

Table 4. CIS Performance Derivatives

FIXED CIS STAGE INERT WEIGHT

MODE DT-1

	CIS STAGE	TOTAL MISSION	
$\frac{\partial W_P}{\partial W_{P/L}}$	.316	1.476	LB/LB
$\frac{\partial W_P}{\partial W_{P/L}}$	2.18	4.13	LB/LB
$\frac{\partial W_P}{\partial I_{sp}}$	153 (MAIN) 3696 (APS)	2046 (MAIN) 7137 (APS)	LB/SEC LB/SEC
$\frac{\partial W_P}{\partial W_{CIS}}$	2.50	5.62	LB/LB
$\frac{\partial W_P}{\partial W_{TANK}}$	.086	1.055	LB/LB
$\frac{\partial W_P}{\partial \text{LOSS RATE}}$	540	1017	LB/LB/HR
$\frac{\partial W_P}{\partial \text{FPR}}$	8712	16480	LB/%

MODE DT-2

	CIS STAGE	TOTAL MISSION	
$\frac{\partial W_P}{\partial W_{P/L}}$	0.314	1.46	LB/LB
$\frac{\partial W_P}{\partial W_{PL}}$	1.60	3.02	LB/LB
$\frac{\partial W_P}{\partial I_{sp}}$	153 (MAIN) 2975 (APS)	1843 (MAIN) 5738 (APS)	LB/SEC LB/SEC
$\frac{\partial W_P}{\partial W_{CIS}}$	1.895	4.46	LB/LB
$\frac{\partial W_P}{\partial W_{TANK}}$	0.086	1.047	LB/LB
$\frac{\partial W_P}{\partial W_{\text{LOSS RATE}}}$	496	932	LB/LB/HR
$\frac{\partial W_P}{\partial \text{FPR}}$	4870	9182	LB/%

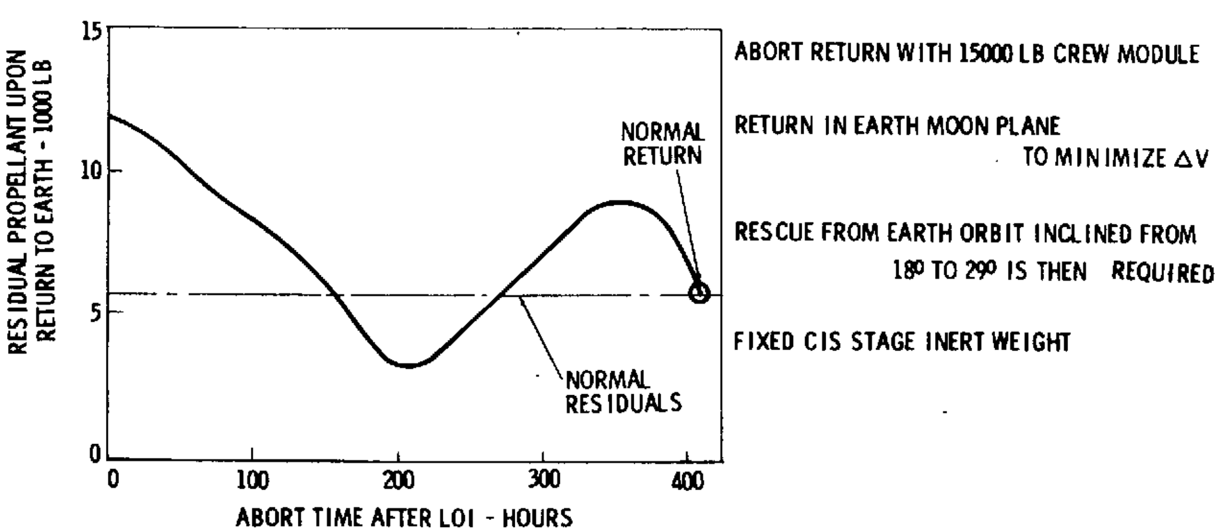


Figure 36. Abort Capability

Another type of Class I mission to equatorial geosynchronous may result in various combinations of outbound/return payloads. A total of 37.6 degrees plane change from the nominal CIS departure orbit renders this mission more stringent, from a performance standpoint, than the lunar mission

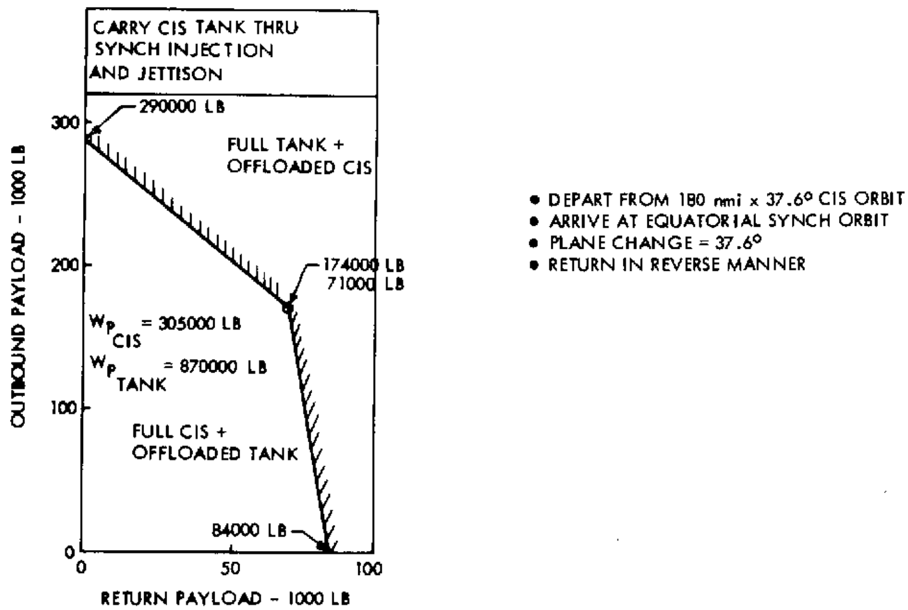


Figure 37. Synchronous Mission Performance

Performance - Class II Missions

In performing a lift to orbit in the OIS mode (Figure 39), the 969-061 LOX/RP flyback booster was specified for Phase III of the study. The capability of this booster plus the OIS firing three 470k pounds main engines and four 10k pounds APS engines resulted in a weight to a 270 n mi x 55 degree orbit of 250,900 pounds. This performance was limited by the maximum launch weight of the second stage of 1,150,000 pounds attributed to the nominal booster. Depending upon the option for mission abort recovery, a fail safe payload of 171,450 pounds may be carried if the OIS is assumed to be capable of attaining a 100-n mi circular orbit with one main engine out. A fail operational mission enabling a normal sequence of events to be followed with one engine out would limit the payload to 162,070 pounds. For a normal mission, a payload of 180,070 pounds could be carried. Normal payload to the 180-n mi CIS orbit would be 185,140 pounds. Jettisoning of the drop tank at 400,000-foot altitude was assumed in the above analysis.

Performance Class III Missions

Utilization of the CIS vehicle to inject an interplanetary payload into escape trajectories (Figure 38) is dependent upon the decision to return the CIS or the crop tank to earth or whether to fly a CIS alone without benefit of a drop tank.

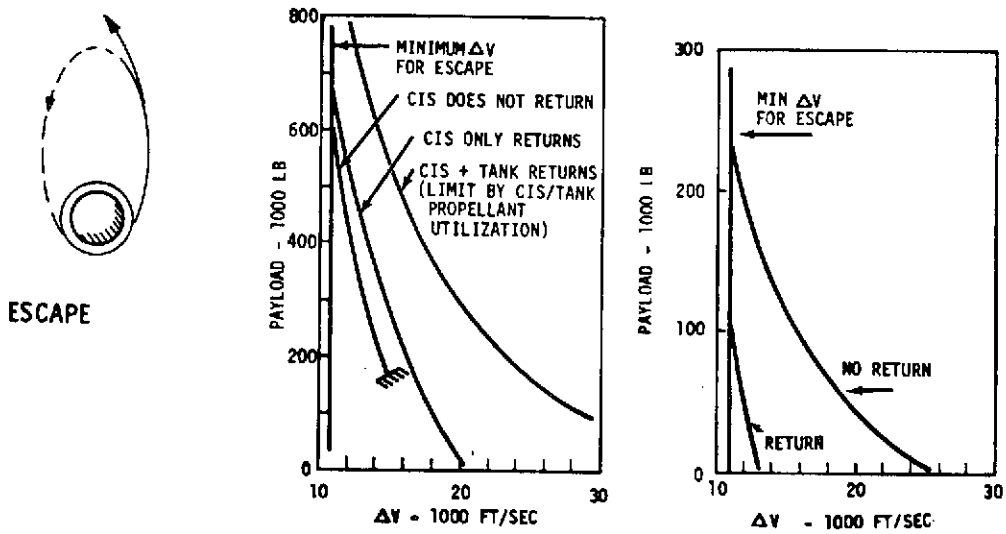


Figure 38. Interplanetary Mission Performance

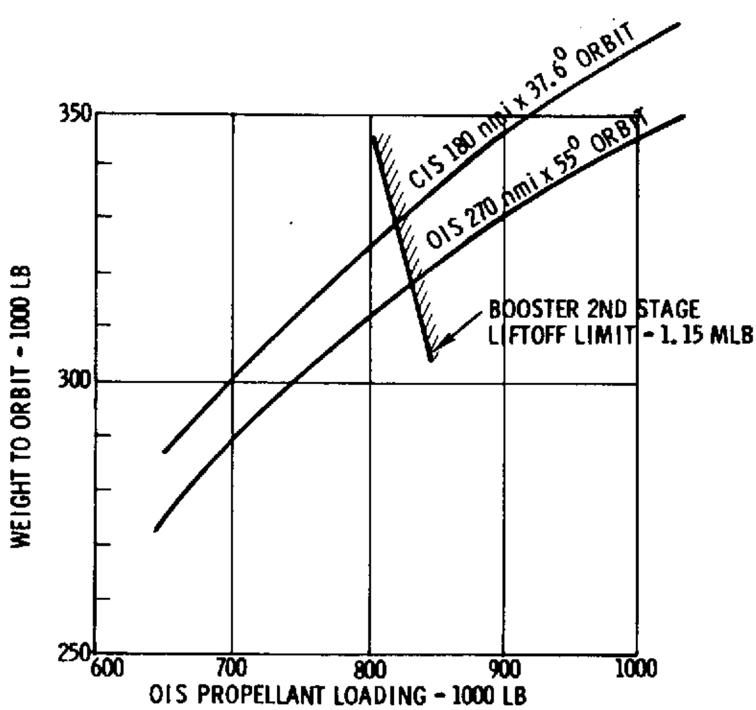
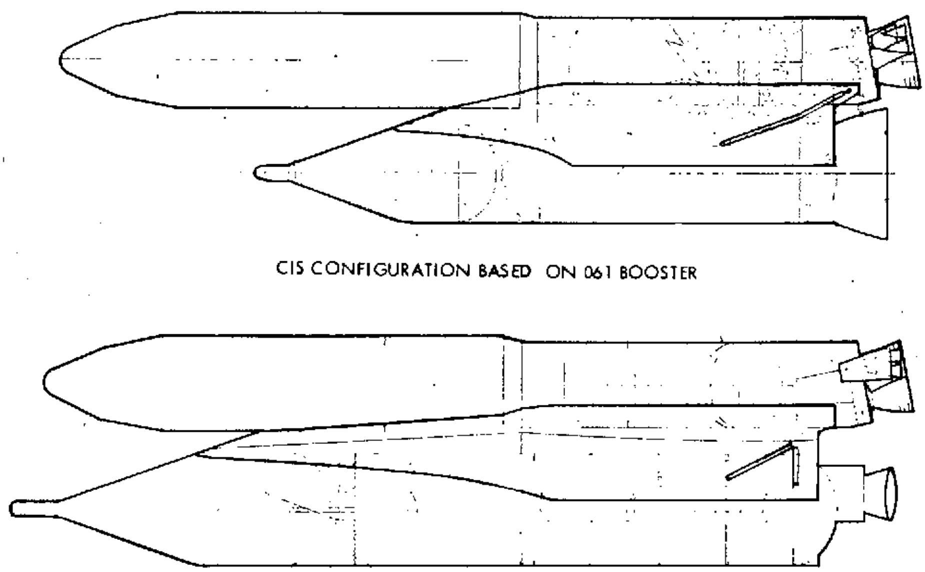


Figure 39. Class II (OIS) Performance

Effects of Latest Booster Configuration

The latest booster/orbiter shuttle configuration (Figure 40) differs principally from those studied heretofore in that the parallel burn concept is used in conjunction with solid boosters. The principal effect on the CIS configuration is that the drop tank is much larger for the latest booster. Effects on the CIS stage are minimal.



CIS CONFIGURATION BASED ON 061 BOOSTER

CIS CONFIGURATION BASED ON PARALLEL BURN-SRM BOOSTER

Figure 40. Booster Effects Comparison

The latest shuttle booster concept (parallel burn SRM's) envisions a very large drop tank with two 156-inch strap-on solid boosters and an orbiter with three 470k pound thrust main engines. Weight to OIS orbit maximizes at about 330k pounds for 1,700,000 pounds of second stage propellant (Figure 41). If the CIS/OIS stage mounted on the 1,770,000 pounds capacity drop tank flies at the optimum point, a payload of over 172,000 pounds may be placed in the 270 n mi orbit. Mounting the OIS propulsion system and other systems upon the drop tank itself would raise the payload to over 202k pounds.

Since the selected CIS concept utilizes the orbiter propulsion sytem, mounts to the drop tank in the same manner as the orbiter, and utilizes an off-loaded drop tank for only the earth departure portion of the CIS mission, it may be concluded that the CIS may operate satisfactorily as an OIS irregardless of the variation in booster/orbiter design. Compatibility will be assured as long as the orbiter concept embraces a separable drop tank for propellant supply.

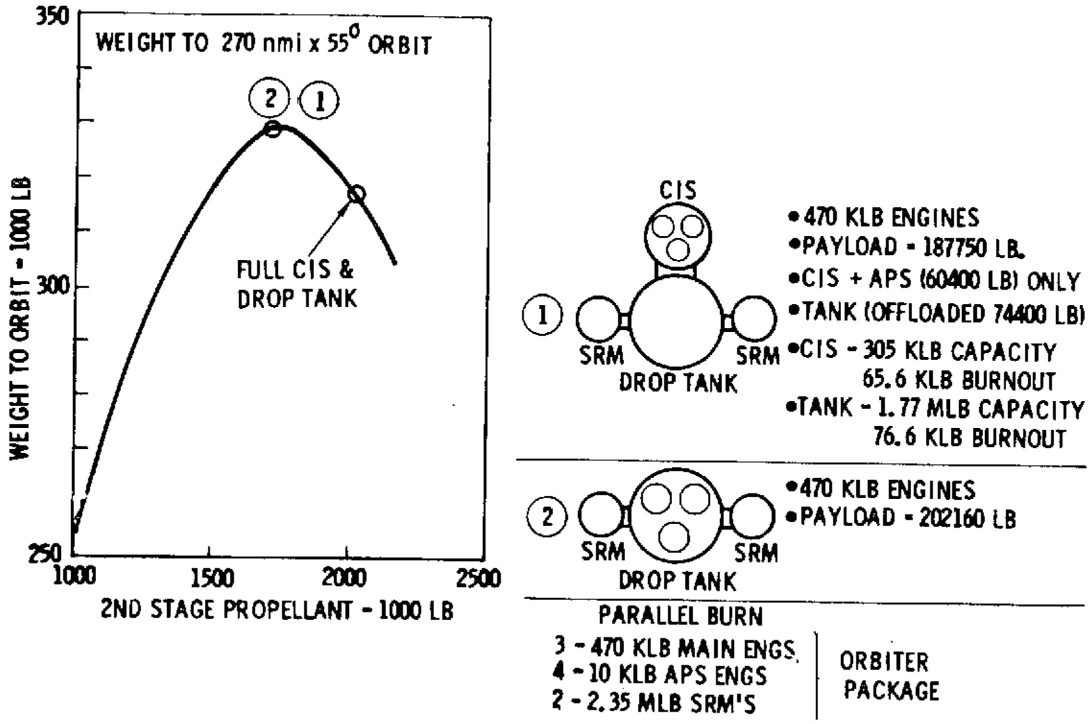


Figure 41. Latest Booster Effects on OIS Performance

OPERATIONS ANALYSIS

The operations analysis was undertaken (1) to establish a baseline program model, (2) to define concepts for performing orbital operations such as engine removal and drop tank replacement, and (3) to define the level of space shuttle activities necessary to support the CIS and lunar program logistics requirements.

The program model established for the study was one in which the primary CIS mission was in support of an extensive lunar exploration program consisting of an eight-man orbiting lunar station (OLS) and a concurrent twelve-man lunar surface base (LSB) program. The logistic requirements for these programs were utilized in establishing the CIS payload performance requirements at 320,000 lb outbound with 16,800 lb inbound. Of the total outbound payload for a typical mission approximately 70% is comprised of lunar tug propellants. The remainder consists of OLS and LSB resupplies and experiments. Detail rationale and derivation of this payload is included in Volume III, Part 1 of this report. Sizing of the CIS for this baseline payload resulted in a vehicle with 305,000 lb propellant capacity.

Orbital Operations

Removal of two of the three 470 K lb thrust CIS engines subsequent to placing a high payload in earth orbit (OIS) results in program savings in two ways. Recovery of the engines in itself amounts to approximately a $7.5 million savings in hardware costs alone. Even more important, however, the removal of the engines results in a net CIS propellant requirement of approximately 50,000 lb per CIS flight or one shuttle flight/CIS flight, thus effecting a dollar savings of approximately $200 million over the baseline 20 mission program. Removal of the engines is accomplished during subsequent shuttle flights whose prime purpose is the replenishment of propellants and utilizes the shuttle manipulator arms for removal and stowage of the engines.

Adoption of the drop tank flight mode results in the expenditure of a drop tank each CIS flight since this is inherent in this flight mode definition. It is, therefore, necessary to replace the drop tank prior to the next CIS mission. This replacement drop tank is one which has been specially modified for the CIS mission. The major modification consists of the addition of high performance, multi-layer insulation to limit boiloff during orbital operations; meteoroid protection for protection during orbital operations; and an aerodynamic fairing to protect the insulation during boost. This modified drop tank replaces a regular shuttle tank at the proper time in the CIS refueling cycle and is carried to the CIS orbit rather than be expended in an intermittent orbit as would a standard shuttle drop tank. Mating of the drop tank to the CIS is accomplished by utilizing the orbiter manipulator arms. In the case of an OIS-type mission, the standard shuttle tank is expended after achieving a 50 by 100 n mi orbit in standard shuttle fashion. Expending the drop tank is necessary to achieve the OIS performance goals. The modified drop tank is used for CIS-type missions and is expended upon the completion of the TLI maneuver.

Definition of the orbital operational requirements necessary to support the lunar program was conducted in order to achieve an insight into the demands to be placed on the space shuttle and the fleet size required. Definition of the optimum payload, in turn, resulted in the definition of the lunar support traffic model. The results indicated a requirement for 20 CIS flights to support the total lunar program. These 20 flights comprised flight intervals of 109, 164, and 218 days. During the initial phases of the program when only the OLS is in operation, a CIS flight is required every 218 days. The logistic requirements for this flight rate (25 shuttle flights) can be easily met with a fleet of three each booster and orbiters and two launch pads. A launch is accomplished every 6 days with the orbiter recovery two days after launch. A comfortable margin exists for any contingencies which may arise.

However, establishment of the LSB after approximately 3 years of OLS operation creates the requirement for two sequential flights falling 109 days after the preceding one. In this instance, the logistic requirement cannot be met with a fleet of three boosters and orbiters. (In fact, a fleet of 7 or 8 each with 4 launch pads would be required). One solution found for this situation might be through the use of an orbital propellant depot (OPD) as an intermediate receiver vehicle. This solution would be expensive, however, since it would require the development and deployment of the OPD. A second possible solution which is recommended since it obviates the need to develop an OPD is to launch the second CIS prior to the end of life of the first unit and thus take advantage of any opportunities for propellant replenishment when unit one is not available. The second CIS is launched directly to the earth parking orbit with excess propellant as its payload. This excess propellant reduces the space shuttle flight requirements by three, thus requiring only three orbiters and boosters.

The shuttle fleet size discussed here will support only the baseline lunar program. Support of the lunar program is marginal with no allowances for shuttle or pad downtimes other than the ground ruled turnaround times of 14 days for the orbiter and booster and 7 days for the pad. These two facts suggest that a larger shuttle fleet should be given serious consideration.

PROGRAM DEVELOPMENT SCHEDULE

The preliminary program development schedule, Figure 44, summarizes the major program milestones and activities necessary for the design, development, production, and test of an operational chemical interorbital shuttle (CIS). The schedule is in consonance with the engineering, procurement, manufacturing, and test schedules that were developed for this study and includes techniques developed by the contractor during previous programs, such as Apollo and Saturn S-II with changes appropriate to CIS requirements. The schedule presents an orderly evolution of events leading to and supporting a CIS program.

Major program phasing depicted in the schedule includes twelve months for Phase C (Design), prior to Phase D (Development/Operations) go-ahead. Included is a two-month review period prior to Phase D go-ahead. Phase D activities reflect in more detail the logical sequence of events leading to an operational posture approximately six years from Phase D go-ahead, which is shown as the zero point on the schedule time bar.

Manufacturing and checkout time spans reflected in the schedule are based on techniques developed by NR on previous programs and studies. This time represents the nominal manufacturing flow for a single shift operation. The time required to fabricate the major structures includes material procurement (first article), detail fabrication, welding, mating, assembly, systems installation, and checkout.

The purpose of the test program is to assist in developing design concepts and to qualify and certify CIS vehicles, support equipment, software, and personnel. The test program to support CIS development reflects the requirement for one structural test article, one auxiliary propulsion system/attitude control propulsion system (covered under component development and qualification), one flight configuration test vehicle (CIS-T) that will be static fired and flight tested, and static firing of the two production flight vehicles. The major tests include twelve months of structural testing, six months of dynamic testing, eighteen months of static firing, and twelve months of space flight testing.

A preliminary investigation reveals that existing goverment-owned/contractor-operated facilities will be adequate with modification so that new facilities are not required. The facility milestones reflected in the schedule (Figure 42) reflect the activation lead time in support of the program.

Test Program

The overall objective of the test program is to assist in developing design concepts and to qualify and certify Chemical Interorbital Shuttle (CIS) vehicles, support equipment, software, and personnel. The philosophy and criteria for testing is to utilize, to the maximum extent, the experience

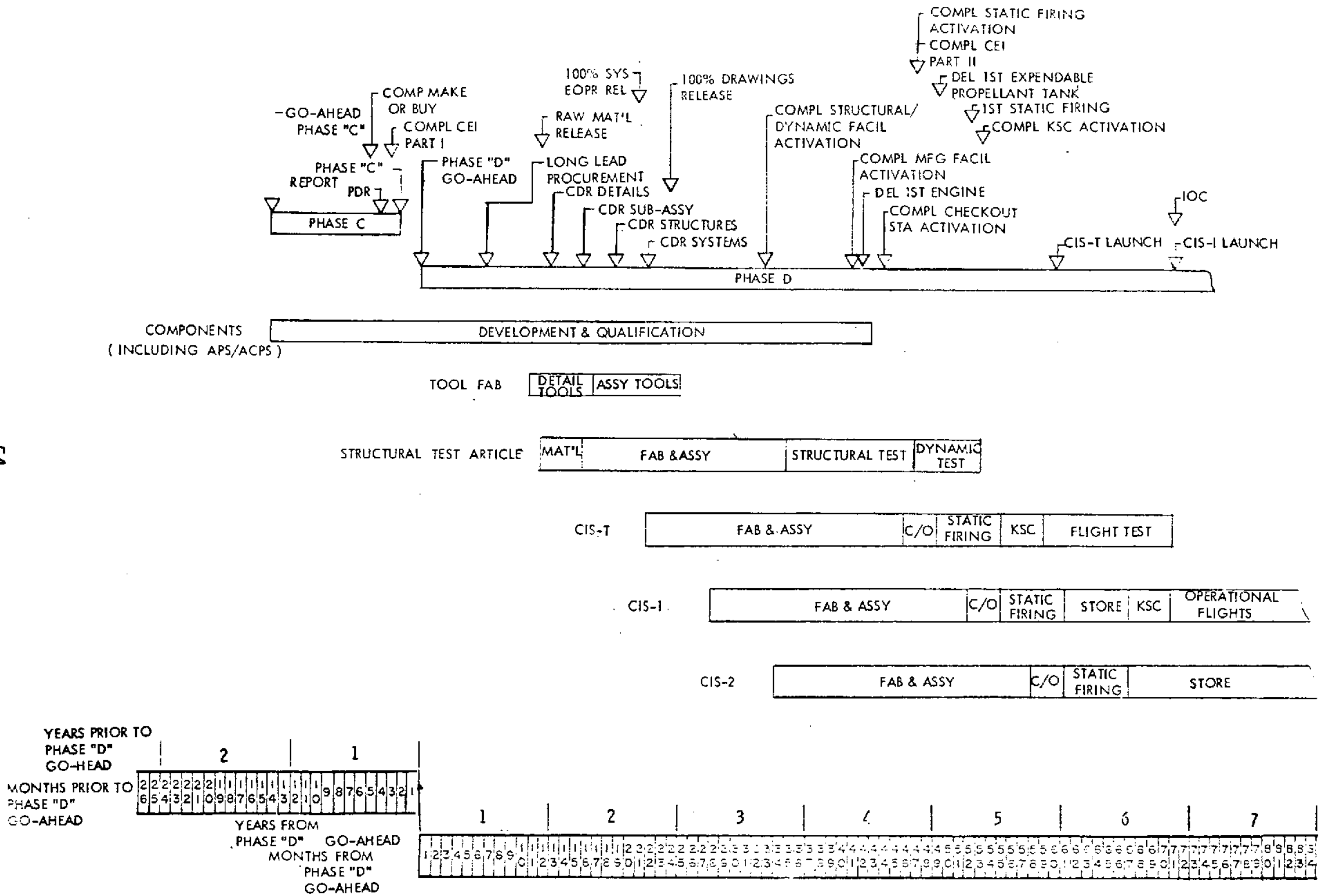


Figure 42. Preliminary CIS Program Development Schedule

Space Division North American Rockwell

gained on previous programs; expand that knowledge by analysis; combine qualification testing with analysis where practical; and minimize or eliminate any redundant testing in the checkout of deliverable hardware. This must be accomplished without compromising the performance capability of the CIS vehicle.

In view of the similarity of the CIS configuration to that of the Space Tug and Space Shuttle Orbiter vehicles, developmental testing is expected to be minimal, covering only those areas where modifications to existing designs are necessary due to CIS unique requirements, such as in-orbit maintenance capability. Similarly, the emphasis during qualification testing will be to verify delta changes to existing designs or operational environment and to certify any reactivated facilities or new tooling or manufacturing processes.

Subsystem development, qualification, and acceptance testing will be conducted at the component through subsystem level, utilizing existing test fixtures and facilities wherever possible. Design verification and certification of manufacturing processes for the primary structure and main tank assemblies will be accomplished utilizing one dedicated structural test article (Figure 43). System and combined system testing to integrate the total system and any critical interfaces with other systems will be accomplished with a single dedicated test vehicle (CIS-T). This vehicle will be of a complete flight configuration containing all flight systems. Current planning provides for cryogenic systems testing and full duration static firing of the CIS-T prior to launch. Each of the production vehicles will also be static fired. The current concept is to perform the static firings on the space shuttle orbiter modified S-IC test stand at MTF.

Final man-rating of the CIS is achieved upon conclusion of the CIS-T flight test program which involves a demonstration of the vehicle's ability to maneuver and dock with other vehicles, to remove engines and attach the expendable propellant tank, to accomplish in-orbit refueling and thermodynamic venting, and to propel unmanned payloads into geosynchronous and lunar orbital missions. After successful completion of all flight tests, it appears feasible to assign the CIS-T to unmanned operational missions for the remainder of its useful life. Before certifying the CIS-T for operational status, however, a maintenance crew would make an in-orbit inspection of the vehicle, perform any necessary maintenance functions, and in general, verify the maintainability of the CIS design. Depending upon the results of the overall test program, it is also conceivable that vehicle modification may be accomplished at this time and subsequently verified by further flight tests.

COST ESTIMATE

The total DDT&E, production, and operations costs for alternate quantities of one, two, four, and six CIS vehicles are shown in Table 5.

CIS PROGRAM STAGE AND ONE-HALF	(MILLION $)				
	DDT&E	PRODUCTION	OPERATIONS		TOTAL
			LAUNCH	MISSION	
1 VEHICLE PROGRAM	1,278.0	175.0	8.3	1,772.2	3,233.5
2 VEHICLE PROGRAM	1,278.0	315.0	21.6	3,545.5	5,160.1
3 VEHICLE PROGRAM	1,278.0	567.0	48.2	7,092.4	8,985.6
4 VEHICLE PROGRAM	1,278.0	799.6	74.8	10,637.9	12,790.3

Table 5 Total CIS Program Cost Estimate

A systematic cost estimating approach was used that injected four major areas of NR expertise: technical know-how, costing analysis, computer utilization, and management synthesis. The Work Breakdown Structure utilized is shown in Figure 43.

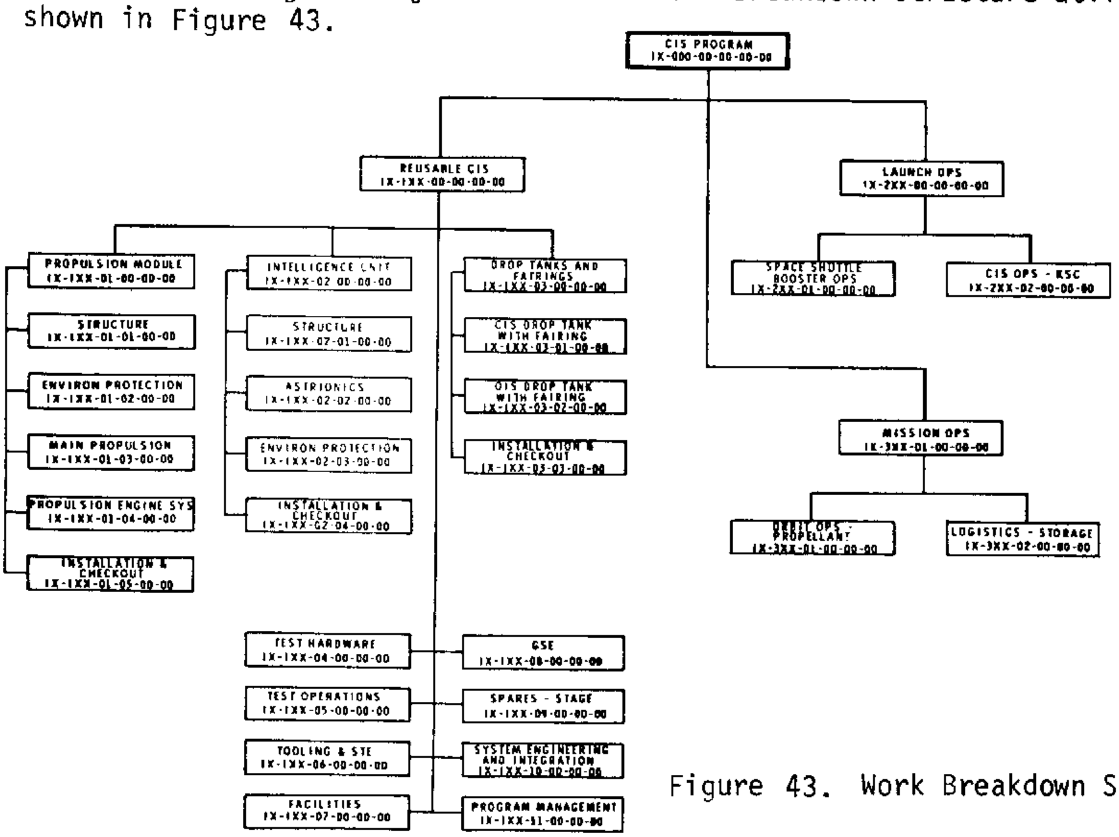


Figure 43. Work Breakdown Structure

The flow of information and inter-relationship of cost and technical data used in costing the CIS vehicle are shown in Figure 44. This cycle can and did repeat itself several times until a final cost estimate was prepared.

To initiate the cost development process, two primary methods were used: the "grass roots" (bottom up) and the parametric costing approach.

The grass roots approach consisted of three methods: functional, vendor, and analog estimates. Functional estimates are based on estimates made by the group or department expected to do the work. Vendor estimates reflect an input of cost from a vendor based on this knowledge or the technical requirements of his hardware in the CIS study. An analog estimate consists of identifying an existing analogous item that has been costed and using that cost for some new requirement.

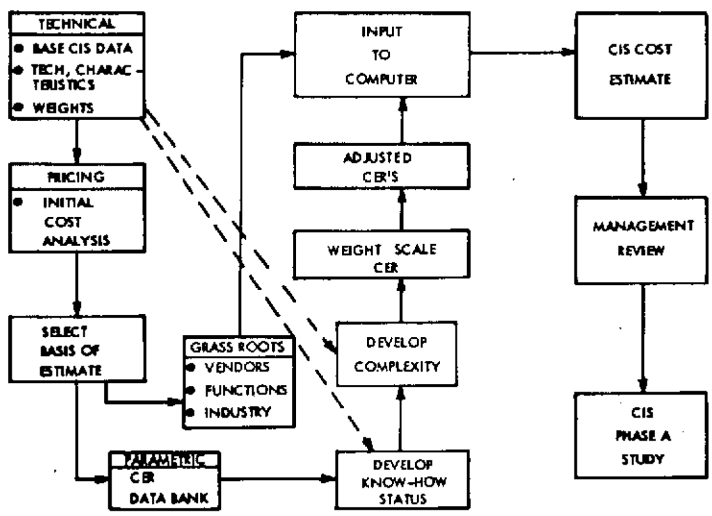


Figure 44. Cost Development

The parametric costing approach was responsible for the major portion of the CIS study costing. This was due to the limited time available and the increased confidence developed during the past several years in use of the parametric approach and cost estimating relationships (CER's).

The Space Division of North American Rockwell Corporation has demonstrated effective major aerospace program cost estimating techniques on the Apollo and Saturn S-II contracts. The experience gained on these programs have resulted in development of a data bank of valid actual program costs, cost models, cost analysis techniques, and CER's specifically related to space program costs. Since the CIS hardware requirements relate in many areas to the hardware already produced for Apollo and Saturn S-II programs, this cost data proved to be very applicable to the CIS vehicle costing.

First, the base CER's are selected (one for non-recurring and one for recurring cost) that most closely represent the subsystem to be costed. With the aid of technical representatives from Engineering and Manufacturing, the experience level for the new hardware is determined. The base CER's experience level is, of course, known. That experience level shift from base to new hardware is translated to a percentage which is applied to the base non-recurring CER in order to adjust that CER for differences in the state-of-the-art between the base and new hardware at inspection of the development phases of the respective programs.

The next major step was development of the subsystem complexity factors as they relate to the base CER. Again, Engineering and Manufacturing personnel are represented at the time this selection is made. The complexity factors, expressed in percentages, were used to modify both non-recurring and recurring CER's. They are a measure of the intrinsic features of a subsystem and reflect the differences in effort needed to develop or produce that subsystem when compared to the subsystems on which the CER cost data were based.

The final adjustment to the base CER is weight scaling. This adjustment is made to modify the base CER's for the weight difference (in pounds) between the base CER hardware and the CIS subsystem hardware. Historical aerospace data indicate that a corresponding change in weight does not dictate a straight line change in cost. Extrapolations have been made from various programs of similar hardware and different subsystem weights and plotted to develop a slope for weight scaling the base CER to the new hardware weight. This adjustment is made to both non-recurring and recurring CER's using the slope applicable to the specific subsystems.

Upon definitization of the basic CER adjustment data, the information was input to the NR Parametric Pricing Computer Program; the results were analyzed and recycled until all anomalies were resolved; and a CIS vehicle cost estimate was prepared for management review, approval, and inclusion in the CIS study final report.

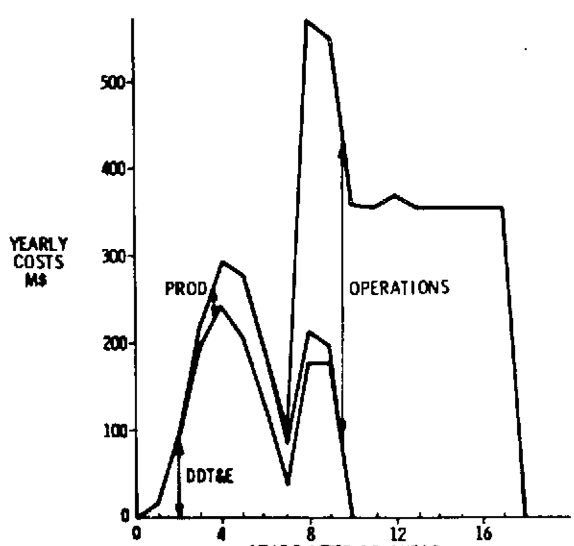


Figure 45. CIS Program Cost Components

Figure 45 shows the yearly costs of development and incrementally the costs of production and operations. The operational costs (cost of space shuttle flights for propellant logistics) clearly dominate the CIS program costs.

NOTE: First flight in Year 8.

CONCLUSIONS

The study has established that the 1-1/2 stage CIS concept is feasible and represents an extremely economical way to meet the requirements of high-energy missions for a space program that incorporates a meaningful lunar exploration plan. A principal feature of its design is that it very effectively utilizes the shuttle technology in that it incorporates sections of the orbiter, such as the propulsion system and the drop tank and, further, interfaces with these elements in the same manner that the Orbiter does. The CIS stage design is relatively insensitive to major perturbations in the drop tank sizing. Some interesting design features which were developed that are worthy of special mention include the utilization of a thermodynamic venting system to minimize the boiloff losses to an extremely low value. This is a critical design requirement.

Another area includes the orbital operations of engine removal, drop tank mating, and refueling. Operations have been planned in such a manner that the propellant logistics mission also performs the duty of transporting to orbit the CIS insulated drop tanks, as well as the spares required to perform in-orbit maintenance.

The development program effort indicated that it is possible to fly after five years from Phase D go-ahead. The cost analysis indicates fairly reasonable costs for such a high performance stage.

VI. STUDY LIMITATIONS

The study required that a feasible reusable concept be developed to perform equally well as an orbital insertion stage (OIS) for high lift capability to low earth (space station) orbit and as an interorbital shuttle (CIS) for future missions beyond low earth orbit.

Requirements to utilize the space shuttle concept as a baseline for first-stage boost combined with study schedule and budgetary limitations, resulted in repetitive effort in Phases I and II to account for changes in the space shuttle booster, and a final OIS/CIS configuration not totally representative of the final space shuttle configurations. However, the final CIS configuration is adaptable to and relatively insensitive to the final space shuttle booster arrangement. Furthermore, time limitations did not permit a structural dynamic analysis or the examination of combinations of axial and lateral loading conditions for which specific structural elements might have been more critical.

It is believed that none of these limitations affected the general conclusions concerning the feasibility of the concept or the general validity of the subsystems analysis.

VII. IMPLICATIONS FOR RESEARCH

In general, the OIS/CIS stage design is based on established and accepted practices and procedures and state-of-the-art capability and demonstrated feasibility in systems design and analysis. However, the unique utilization of the vehicle for long-term space application establishes requirements for developmental activities to support this new utilization.

The following areas are considered prime candidates for research and development and require analytical investigations subsequently leading to hardware development or establishment of supporting techniques. The need for special technology development arises because current system/subsystem technologies may be inadequate, requirements for more advanced technologies have not been identified previously, or current technology efforts need re-orientation. Development in these areas may require specific effort separate from the main stream vehicle phased development program to insure timely availability of the necessary technology or equipment. An SR&T schedule is shown in Figure 46.

Fracture Mechanics Material Properties

The material properties required for fracture mechanics analysis are available for many materials, including 2219-T87 and 2014-T651 aluminums being considered for the CIS vehicle. However, when dealing with very thin materials (less than 0.2 inches) the commonly accepted values of K_{IC} and K_{TH} are no longer applicable. The thickness of material being considered for the CIS design are generally less than 0.1 inches. Therefore, material properties (e.g., K_{IC} and K_{TH}) must be developed for the materials and thickness ranges being considered for use in the CIS vehicle.

Advanced Composite Structure

Current technology with respect to advanced composites, which is adequate for the CIS fabrication, consists of hand layup techniques in conjunction with some mechanized equipment used for flat panel layups and filament winding of tubular structures similar to that used in aircraft design. Since the aircraft industry is the major user of composite materials and is not concerned with cryogenic temperatures, sufficient data for reliable design allowables at cryogenic temperatures are limited. Consequently most of the cryogenic mechanical property data being used for the CIS design are based either on engineering judgment or extrapolated from limited cryogenic test data.

Graphite composite honeycomb construction has been used in aircraft production for small panels wuch as wheelwell doors (4' by 5'), flaps (8' by 2'), and wing boxes (3' by 7'). A composite S-II center engine beam, 10 feet long, was recently constructed and tested, with results falling in the predicted range. However, no large composite structures have been built, and the verification of fabrication, inspection, and repair procedures for a large component such as the CIS skirt structure must be shown.

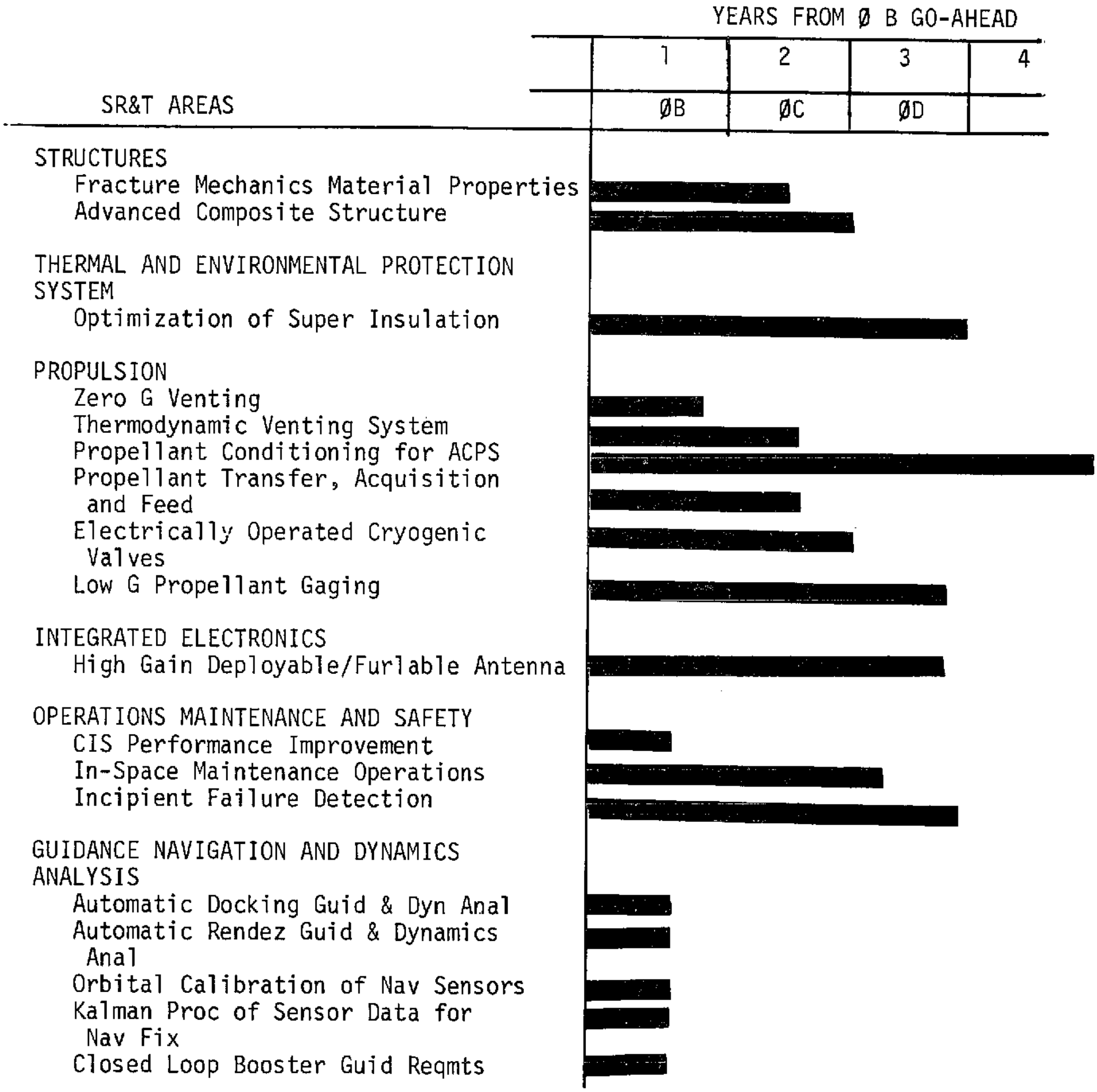


Figure 46. SR&T Schedule

Another required area of investigation is the verification of the micro-meteoroid protection of these composite materials in a space environment.

Optimization of Lightweight, Spacerless Super Insulation Material

Recent studies and test programs at NR have shown that lightweight, high performance insulation systems can be fabricated from commercially available embossed Mylar. These systems are highly competitive with other concepts in performance and appear to be lighter in weight.

Even better performance can be realized if the optical properties of the material can be enhanced and if the embossment pattern can be optimized.

Zero G Venting

The feasibility of the basic thermodynamic vent heat exchange system concept has been experimentally established. In addition, several technology studies provided additional data and current studies will soon provide more data.

Two systems were considered for control of the CIS internal thermodynamics. These are (1) a thermodynamic vent heat exchange system, and (2) direct overboard venting. The thermodynamic vent heat exchange system includes an expansion valve where LH_2 is throttled to a relatively low pressure and temperature. This provides the necessary temperature differential for extraction of heat from the propellant by an appropriate heat exchanger. Direct overboard venting using low g thrust during the venting operation as proposed for CIS has been experimentally confirmed during tests of an orbiting Saturn S-IVB LH_2 tank.

Thermodynamic Venting System

Reliable analytical procedures have not been fully developed for this form of venting. Although the feasibility of this vent concept, using compact heat exchangers and mixers, has been experimentally established at "1" g, it remains to be demonstrated at low gravity. Also, experimental demonstration of the wall mounted concept remains to be performed.

Propellant Conditioning System for Attitude Control Propulsion Subsystem

The Attitude Control Propulsion System (ACPS) contemplated for the CIS is a gaseous H_2/O_2 integrated system. The system, therefore, requires a propellant conditioning system to convert the liquid propellant to a high pressure gas which is burned in the ACPS engines, and is also used for fuel cell reactants and propellant tank pressurization. Several Government funded programs are now being conducted to develop components for a similar but larger size prototype propellant conditioning system originally conceived for the space shuttle.

Propellant Transfer, Acquisition and Feed

The transfer of cryogenic or storable propellants from a supply to a receiver tank in a zero or low g environment has not been accomplished. The use of capillary devices to acquire propellants in a zero or low gravity environment and provide the capability of the tank feeding out to a propulsion system also has not been accomplished. This study and other similar studies have identified advances in technology required to provide the confidence desired in design of an operational system.

Electrically Operated Cryogenic Valves

Electric motor driven actuators for cryogenic valves appear feasible; however, research, design and development are required to evaluate and optimize envelope, weight, output and operational speed characteristics. (Heretofore, cryogenic valves have generally utilized pneumatic actuators.) Further study is also required to evaluate motor operation, torque and speed characteristics, at various temperature levels such as ambient temperature for ground checkout, operation at intermediate temperatures between ambient and cryogenic levels, and also at cryogenic temperatures. Power consumption must also be optimized over this temperature range.

Low "g" Propellant Gaging Development

The candidate systems for CIS propellant gaging have included the analog "hot-wire" and analog capacitance type systems, operating under settling accelerations as low as 10^{-5} g. (True zero-g gaging is not considered realistic in light of the high accuracy requirements dictated by the high cost of propellants loaded in orbit.) Special sensor configurations and electronics have been designed to minimize the effects of capillarity on gaging accuracy. The most promising design appears to be a thin vertical wire for the analog "hot-wire" sensor or a pair of thin vertical wires for the capacitance probe. Dual current levels (alternately pulsed) would be used for the "hot-wire" system to compare its electrical resistance in the self-heating and no-heating modes. This "delta" resistance would be self-calibrating in flight, and independent of changes in leadwire resistance, electrical noise and system biases.

High Gain and Rendezvous Radar Deployable, Furlable Antenna Systems

Some independent studies, by Jet Propulsion Laboratories Outer Planets program, Lockheed and others have been conducted. However, no production hardware was produced. It is necessary to develop this unique application before full utilization of this concept can be established.

Antenna systems which require a large reflector which can be opened and closed, extended and retracted, rotated and gimballed have not been developed to the degree necessary for direct adaptation to the CIS vehicle. A study to evaluate and integrate all apsects of this problem needs to be conducted.

CIS Performance Improvement through Alternate Injection and Operational Techniques

Lunar and planetary missions have been accomplished by injection into trans-planetary/lunar trajectories by either direct injection from earth or earth orbit. Alternate injection modes, such as multi-orbit perigee burns, staging of reusable vehicles and expending propellant modules, have been postulated in lunar and planetary mission studies.

In-Space Maintenance Operations

Current space transportation systems are oriented toward performing maintenance operations on the ground. On-orbit maintenance of manned satellites is planned for such itmes as Skylab and Earth Orbit Space Station. In this instance the majority of maintenance operations is performed on equipment located in a shirt sleeve environment inside the station. In-space maintenance has been postulated for Space Transportation System elements but not to any great depth.

The CIS has a design life of up to three years in space with capability for maintenance in orbit. A research and development effort is required to define a cost effective approach to orbital maintenance taking into consideration application to other space program elements. Capability and limitations of extra-vehicular activity experienced on Apollo flights would form the basis for identifying maintenance tasks which could be performed by man. Space shuttle manipulator study activity would parallel the study activity for CIS which necessitates the use of remote manipulators because of human limitations of EVA.

Incipient Failure Detection

Missions of increasing duration impose operating reliability requirements which often approach the theoretical design limitations of existing equipment. It would be possible, however, to extend the period of dependable service usage by forecasting the impending failure of a component or subsystem. Although incipient failure detection (diagnostics) would benefit any program, the CIS would be ideally suited to such an approach employing available space elements. Statistical demonstrations of reliability could be less rigorous, and the reduction of test costs could help defray the costs of the detection equipment. A significant breakthrough is required, however, to identify component operating characteristics as components approach their point of failure.

Automatic Docking Guidance and Dynamic Analyses

The guidance, control and docking dynamic requirements for automatic docking between the CIS, payloads and other space vehicles have not been determined. The docking concepts, techniques and approaches used in the manned space flights may not be applicable to the CIS vehicles.

The above requirements must be established before preliminary design phase can proceed.

Automatic Rendezvous Guidance and Dynamic Analyses

Automatic rendezvous to a few hundred feet of accuracy has not been attempted. Guidance techniques applicable to manned rendezvous systems are well known but applicability to automatic implementing is uncertain. If guidance policy is changed in philosophy and format, the vehicle dynamics and responsiveness to commands will be different from the manned system case.

Orbital Calibration of Navigation Sensors

Prevailing manned Apollo flights did not require the on-board calibration of navigation sensors. Prime reasons are (1) the availability of ground tracking in furnishing navigation fix; (2) the use of a navigation base which furnishes a rigid mechanical reference to the star telescope and the inertial reference unit; and (3) the inertial reference unit is a gimballed system which facilitates ground calibration. Because of this status, orbital calibration has not been necessary and its technique development has been ignored. For the CIS vehicle, ground tracking may not be available, colocation of star tracker and horizon sensor is functionally non-compatible (one views the sky while the other views the earth), and gimballed inertial reference unit may not be used.

Kalman Processing of Sensor Data for Navigation Fix

Prevailing analyses treat, and computer programs are designed to deal with, the navigation fix determination problem in circular orbits and a simple error model. There is a need to develop similar techniques and processes for general orbits and more comprehensive error model.

Closed Loop Booster Guidance Requirements

Prior studies have examined booster control in the atmosphere while using open loop guidance. However, it is necessary to incorporate these results with the use of closed loop guidance during second stage ascent and to examine the feasibility of introducing closed loop guidance during first stage ascent.

VIII. SUGGESTED ADDITIONAL EFFORT

The study was sensitive to the space shuttle configuration and performance characteristics. Since considerable variation of the space shuttle concept occurred during the study period, it was necessary to adjust the CIS configuration and mission mode concepts to reflect the shuttle influences.

The final CIS configuration matched the drop tank shuttle concept and is relatively insensitive to the booster arrangement whether solid or liquid, or if series or parallel burn. It is also insensitive to the drop tank propellant capacity.

If the shuttle concept were to undergo basic approach changes, an updating of the CIS concept would be advisable. While a continuous tracking of the shuttle program does not appear justifiable at this time, it would be advisable to reexamine the CIS concept presented here as soon as the space shuttle program reaches a concept definition.

ABBREVIATIONS AND ACRONYMS

ACPS	Attitude control propulsion system
APS	Auxiliary propulsion system
CER	Cost estimating relationship
CIS	Chemical interorbital shuttle
CIS-T	Chemical interorbital shuttle - all systems test vehicle
DAU	Data acquisition unit
DCM	Data and control management
DDT&E	Design, development, test and engineering
DT	Drop tank
EO	Earth orbit
EOI	Earth orbital injection
EPO	Earth parking orbit
FO/FS	Fail operational/fail safe
GH_2	Gaseous hydrogen
GN&FC	Guidance, navigation and flight control
GOX	Gaseous oxygen
H_2O_2	Hydrogen peroxide
HRP	Heat resistant phenolic
IMU	Inertial measurement unit
IU	Intelligence unit
ISPP	Integrated space program plan
K_{TH}	Threshold stress intensity factor for sustained loads
Kw_e	Kilowatt electrical
K_{1C}	Critical stress intensity factor
LEO	Low earth orbit
LH_2	Liquid hydrogen
LO_2	Liquid oxygen
LOI	Lunar orbit insertion
LOX	Liquid oxygen
LSB	Lunar Surface Base
MLI	Multi-layer insulation
MPS	Main propulsion system
MSFN	Manned space flight network
MTF	Mississippi Test Facility

ABBREVIATIONS AND ACRONYMS (CONTINUED)

NPSH	Net positive suction head
NPSP	Net positive suction pressure
NR	North American Rockwell Corporation
OLS	Orbiting Lunar Station
PCA	Propellant conditioning assembly
R&DS	Rendezvous and docking subsystem
RNS	Reusable nuclear shuttle
RP	Rocket propellant
RTG	Radioisotope thermoelectric generator
SL	Sea level
SR&T	Supporting research and technology
TEI	Trans-earth injection
TLI	Trans-lunar injection
TVC	Thrust vector control
VAC	Volts alternating current
VDC	Volts direct current