SP 07R70-F

Contract NAS 5-11295

PLANETARY EXPLORER LIQUID PROPULSION STUDY

FINAL REPORT

February ,1971

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

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FINAL REPORT, 8 JUL. 1970 - 8 JAN. 1971

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Approved: John E. McCabe
J. McCabe
Study Program Manager

PREPARED FOR

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1. Report No. SP 07R70- F	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle PLANETARY EXPLORER LIQUID PROPULSION STUDY		5. Report Date February 1971
		6. Performing Organization Code
7. Author(s) John E. McCabe and Vincent J. Sansevero		8. Performing Organization Report No. SP 07R70- F
9. Performing Organization Name and Address Hamilton Standard Division of United Aircraft Corporation Windsor Locks, Connecticut 06096		10. Work Unit No.
		11. Contract or Grant No. NAS 5-11295
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Goddard Space Flight Center Technical Monitor: Donald Miller		13. Type of Report and Period Covered Propulsion System Study July 1970 to January 1971
		14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The primary objectives of this study were to concept, evaluate and compare a number of candidate spacecraft propulsion systems of the monopropellant hydrazine type relative to two Venus missions of an Explorer class satellite. The propulsion system of this Planetary Explorer satellite will be utilized to accomplish mid-course trajectory corrections, attitude control, spin control and orbit maneuvers of the spacecraft. The candidate systems evaluated utilized existing flight proven hardware and no new technology is required to accomplish mission objectives. A single 5 lb_f engine was used to accomplish all required functions. Production engine test data was used in all performance analysis of the spacecraft. A computer program was generated as part of the study program to evaluate candidate system performance under simulated mission conditions.

17. Key Words (Selected by Author(s)) "Monopropellant hydrazine system", "Maneuvering and controlling a spin stabilized spacecraft", "reaction control system".	18. Distribution Statement

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	289	

PREFACE

This report describes the effort accomplished under Contract NAS 5-11295. The primary objective of this effort was to concept, evaluate and compare a number of candidate spacecraft propulsion systems, of the monopropellant hydrazine type, relative to two Venus missions of an Explorer class satellite. In addition, effort was expended in conducting parametric and design studies to develop information useful to GSFC in conducting spacecraft level trade-offs beyond the scope of this propulsion study. This study demonstrated that the mission requirements can be satisfied using existing equipment and that no new technology is required.

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ILLUSTRATIONS

ILLUSTRATIONS (Cont)

TABLES

TABLES (Cont)

SECTION 1.0
INTRODUCTION

0 INTRODUCTION

This is the final report for the Plantetary Explorer Liquid Propulsion Study which was conducted by the Hamilton Standard Division of United Aircraft Corporation for the Goddard Space Flight Center (GSFC) of the National Aeronautics and Space Administration with Mr. Donald Miller as Technical Monitor.

The primary objectives of this study were to concept, evaluate and compare a number of candidate spacecraft propulsion systems, of the monopropellant hydrazine type, relative to two Venus missions of an Explorer class satellite. The propulsion system of this Planetary Explorer satellite will be utilized to accomplish mid-course trajectory corrections, attitude control, spin control and orbit maneuvers of the spacecraft.

Technical Contributions to this study effort were provided by the following individuals at Hamilton Standard and at the United Aircraft Research Laboratories.

John McCabe	(Program Manager)
Vincent Sansevero	(Study Technical Manager)
Carl Arvidson	(Design Engineering)
David Jackson	(System Analysis)
Paul Falk	(System Analysis)
Joseph Genovese	(System Analysis)
Richard Toelken	(Reliability)
Dr. Aldo Peracchio -	UARL (Plume Analysis - Consultant)

This study effort, which was funded under NASA Contract NAS 5-11295, was initiated 29 June 1970, and completed with the submittal of this final report.

SECTION 2.0
SUMMARY

0 SUMMARY

The study program conducted for the Planetary Explorer Program is discussed in the following three sections of this report.

- Section 3.0 - STUDY PROGRAM SCOPE AND LOGIC
- Section 4.0 - CANDIDATE SYSTEMS
- Section 5.0 - PARAMETRIC AND DESIGN STUDIES

Study Program Scope and Logic (Section 3.0) - This section describes the scope of the study program -- areas covered, types of studies and analysis performed. Also included is a discussion of the logic implemented -- the technical approach and the time phasing, or sequencing, of the various engineering tasks as well as the major program milestones.

Candidate Systems (Section 4.0) - This section describes the process used to select candidate systems, and then describes the systems selected as candidates. This section also describes the criteria used to evaluate the selected candidate systems, and then presents a comparative evaluation of each of the selected candidates.

Parametric and Design Studies (Section 5.0) - This section describes studies conducted as part of the study program and presents the results of these studies. In some cases, these studies were conducted to develop the data necessary for the evaluation and comparison of the candidate systems, but in general these studies generated information that was either applicable to all candidates, or was in a parametric form such that GSFC could use the information in spacecraft level trade-offs beyond the scope of this propulsion system study.

SECTION 3.0
STUDY PROGRAM SCOPE AND LOGIC

3.0 STUDY PROGRAM SCOPE AND LOGIC

The scope of this study, as defined in the GSFC Statement of Work, is to select, evaluate and compare a number of candidate propulsion systems which are capable of performing the maneuvers defined in GSFC Specification Number S-723-P-10, Revision A. These maneuvers include trajectory corrections, attitude control, spin control and orbital changes. The evaluation and comparison of the selected propulsion systems covers the following major areas·

- Reliability
- Weight
- Performance Margin
- Power
- Cost
- Operations
- Components
- Structural/Physical Considerations

The logic diagram of Figure 3.0-1 provides an overview of the study program. The program defined in the logic diagram is conducted in three successive phases where the technical approach can be described as one which is directed at examining a large sample of potential system approaches in a manner that eliminates those that are unacceptable, or less desirable, with a minimum amount of effort expended, so that the major effort can be applied in the evaluation and comparison of the more promising concepts. This approach is implemented in the three successive phases where the program progresses from a relatively wide scope and shallow depth look at concepts to an in-depth evaluation of a narrow scope of concepts. As part of the study program logic, technical activities were conducted in parallel to the iterative process of "narrowing in" on the preferred system concepts. These parallel technical activities, which included basic design studies, parametric studies, and basic performance analysis, in some cases supported the evaluation and comparison of the various system concepts; but in general, they developed data common to all concepts which is in a form that may be used by GSFC for program level and spacecraft level trade-offs in their final selection of a propulsion system concept.

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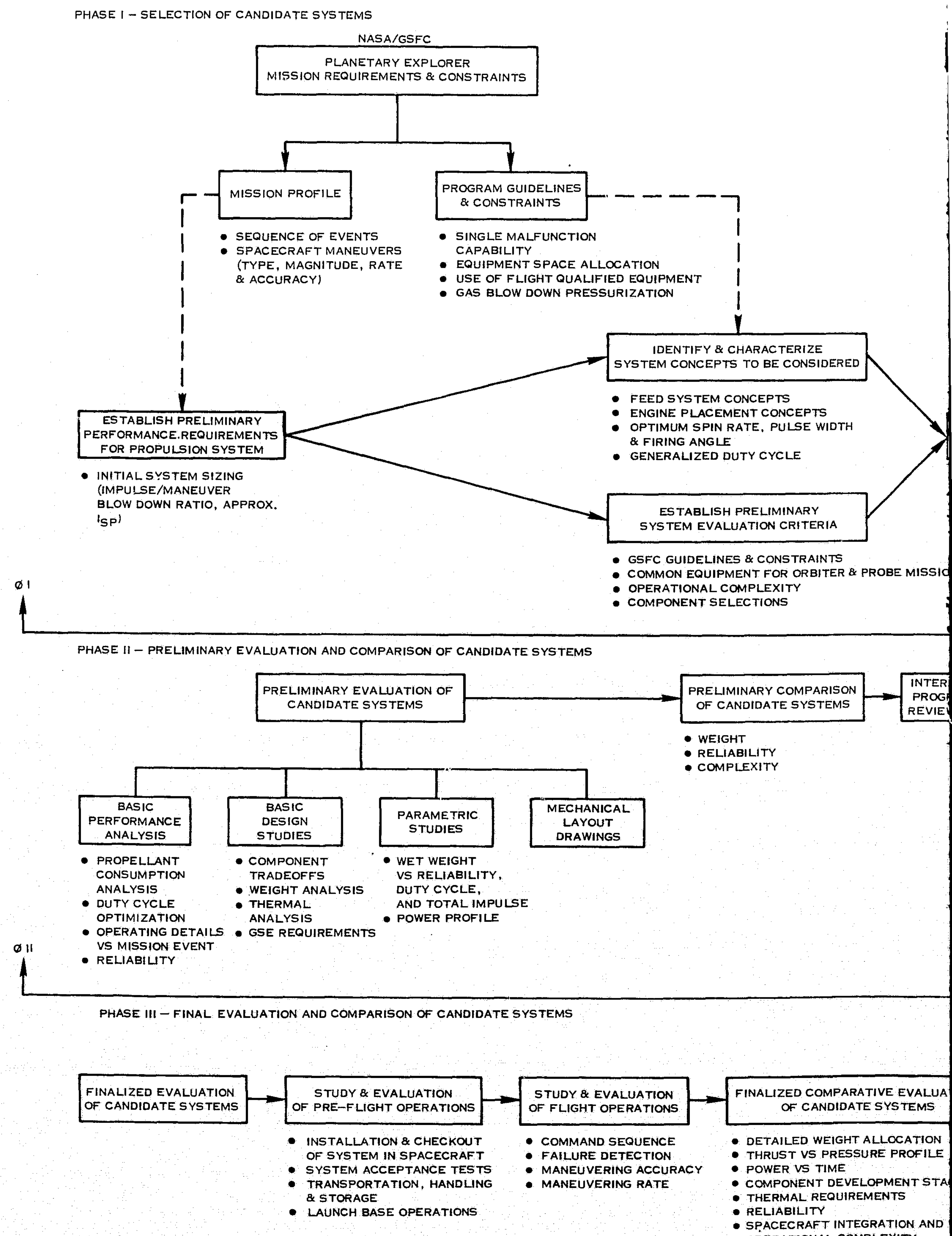

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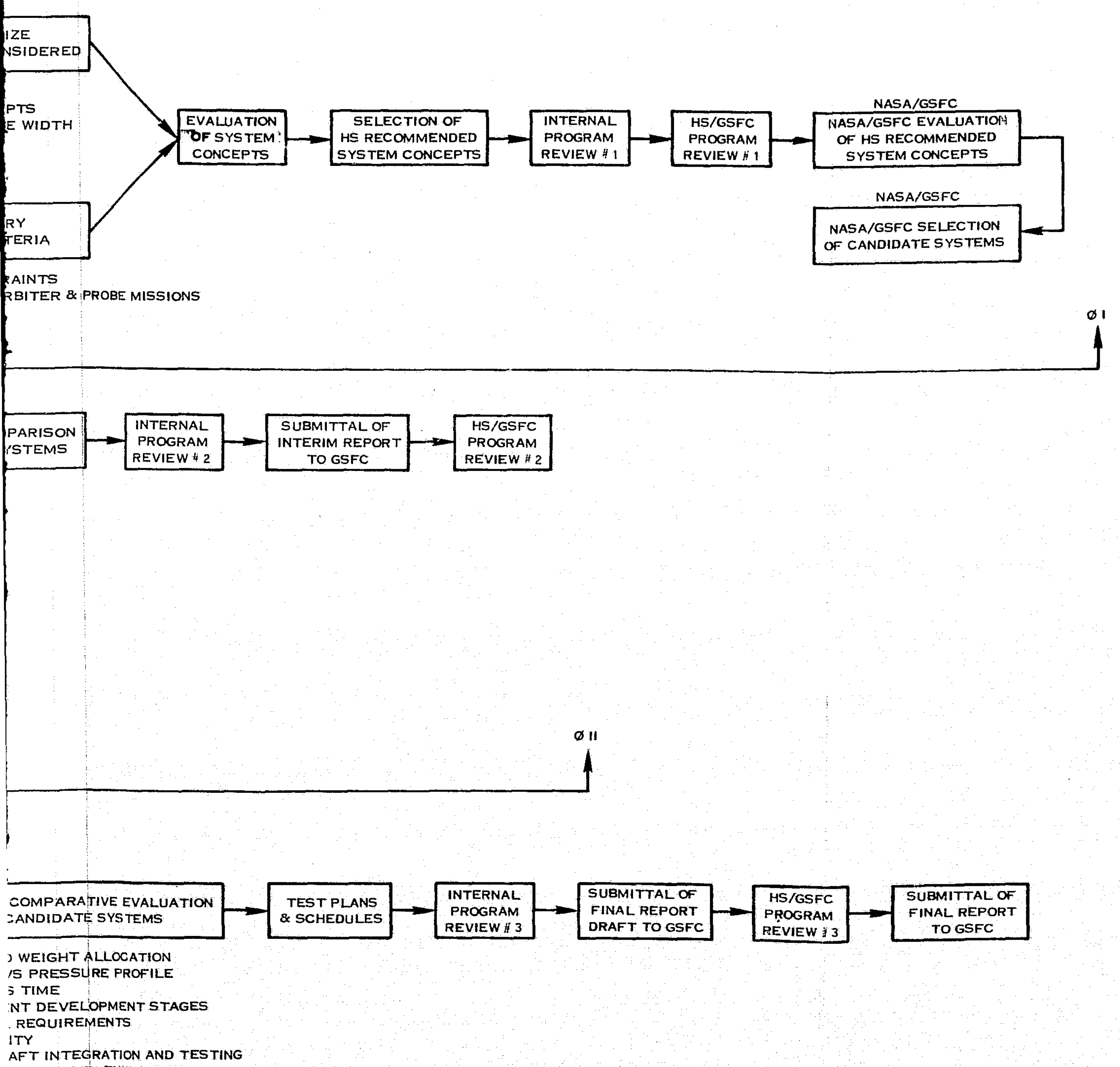

FIGURE 3.0-1. LOGIC DIAGRAM — PLANETARY EXPLORER LIQUID PROPULSION SYSTEM STUDY

SECTION 4.0
CANDIDATE SYSTEMS

4.0 CANDIDATE SYSTEMS

The selection of candidate systems, from among those originally considered, was accomplished during Phase I of the study program. Subsequent to this selection, the candidate systems were evaluated and compared. The evaluation and comparison activities were accomplished during Phase II and Phase III of the program. The following aspects of the candidate systems are discussed below.

- The Process of Selecting Candidate Systems
- Description of Candidate Systems Selected
- Evaluation of Candidate Systems
- Comparison of Candidate Systems

4.1 The Process of Selecting Candidate Systems

Propulsion systems of the type required for the Planetary Explorer application can be characterized and evaluated on the basis of the particular approach taken in implementing each of the basic system functions which are listed below. In almost all cases which can be postulated, the evaluation and selection of the optimum method for any one function can be accomplished independently of the methods selected for the functions.

- Propellant orientation
- Propellant pressurization
- Propellant feed system (to the engines)
- Quantity, location and thrust level of rocket engine assemblies (REA's)

The applicable propulsion subsystem specification for the Planetary Explorer spacecraft stipulates the use of bladderless tanks since orientation will be accomplished by the acceleration forces exerted due to the spin stabilized mode the spacecraft will operate in. Also stipulated was the use of a "gas blowdown" method for propellant pressurization and a 5 lb thrust rated engine assembly. This left the following two system characteristics to be concepted and evaluated.

- Propellant feed system (to the engines)
- Quantity and location of 5 lb thrust rocket engine assemblies

The process of selecting candidate propulsion systems on the basis of evaluating the two basic system characteristics mentioned above is illustrated in the logic diagram of Figure 4.1-1. The various engine placement concepts and engine feed system concepts, combinations of which constitute a propulsion subsystem, considered as part of the selection process, are illustrated in Figures 4.1-2 and 4.1-3, respectively, and referenced in the logic diagram.

4.1 (Continued)

The evolutionary process of developing feed system concepts is illustrated in Figure 4.1-3, along with a "Fundamental System" design concept which is used as a reference. The fundamental system is used to define the simplest system concept which will provide all required capabilities. This baseline system excludes from consideration the possibility of any equipment malfunctions, or error in judgement. The fundamental system then becomes the basic building block which has to be augmented to arrive at the system which most effectively supports the Planetary Explorer mission objectives.

As shown in the logic diagram of Figure 4.1-1, engine placement concepts and feed system concepts were evaluated and concepts recommended for further evaluation were selected using the criteria in Tables 4.1-I and 4.1-II. The recommended concepts, which are illustrated in Figure 4.1-4, were submitted to GSFC. After review and evaluation by GSFC, the final selection of candidate system concepts was made (Reference Figure 4.1-5) and with minor modification were those concepts recommended by Hamilton Standard.

A relative rating of the selected candidate engine placement concepts, based on the evaluation criteria established, is presented in Table 4.1-III. The candidate subsystem concepts identified in Table 4.1-IV were established by combining the selected engine placement concepts with the selected feed system concepts.

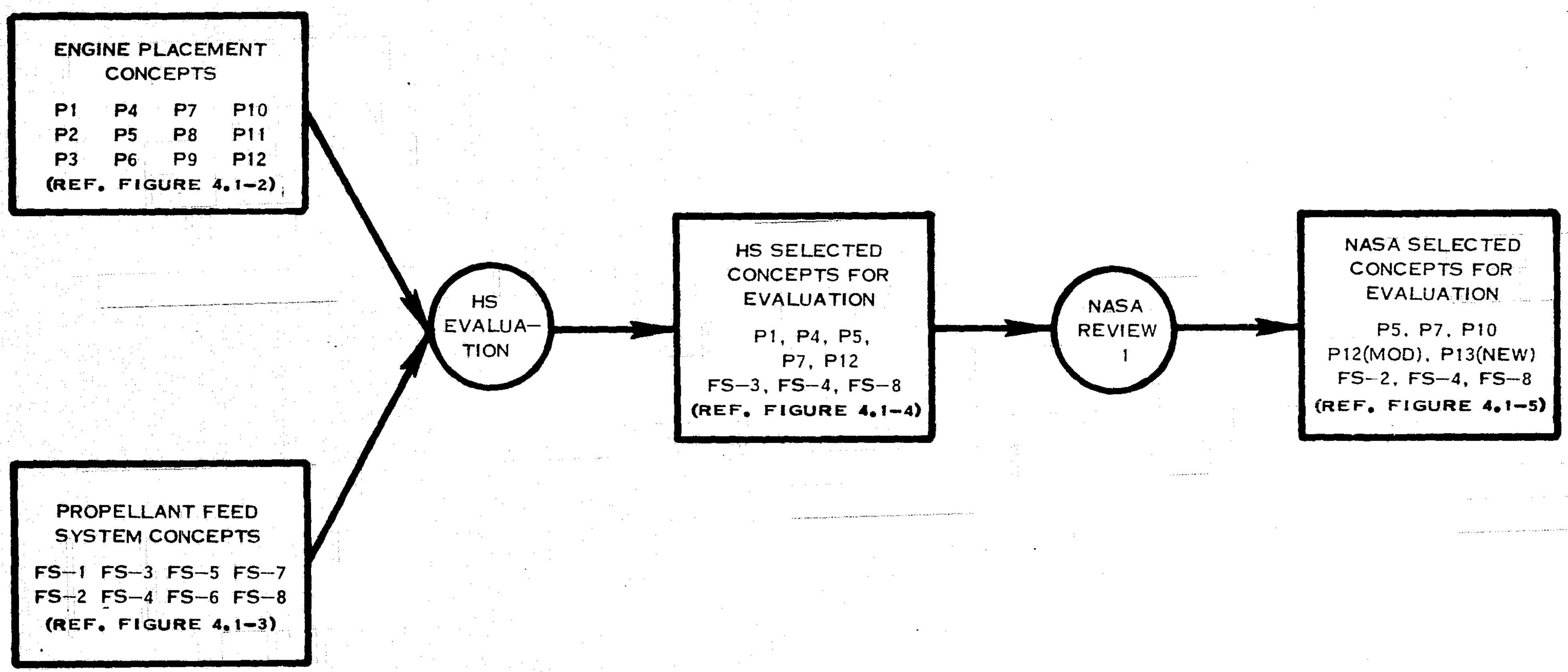

FIGURE 4.1-1. LOGIC DIAGRAM-PROCESS OF SELECTING CANDIDATE SYSTEMS

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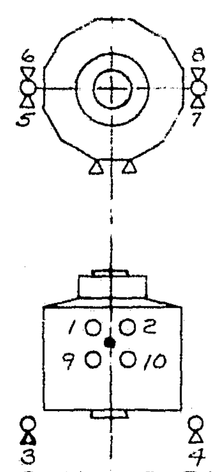


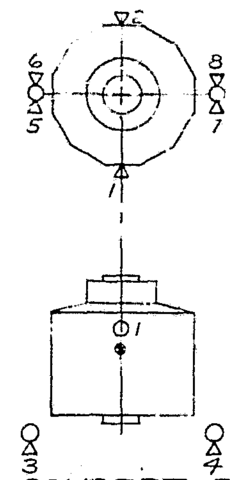
CONCEPT P1

10 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE
- GOOD RESOLUTION FOR ΔV
- CONCEPT PACKAGES WITHIN SPACECRAFT RESTRAINTS

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,2,9,10	1,2,3,10	
ACS	3 OR 4	5,7, OR 6,8	
+ SPIN	6,7	6 OR 7	
- SPIN	5,8	5 OR 8	



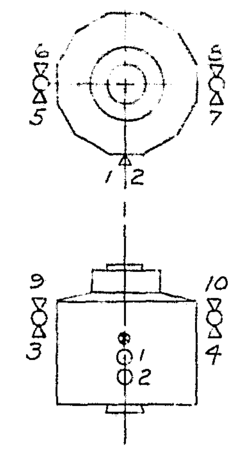
CONCEPT P2

8 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,3,4	2,6,8	
ACS	3 OR 4	5,7, OR 6,8	
+ SPIN	6,7	6 OR 7	
- SPIN	5,8	5 OR 8	



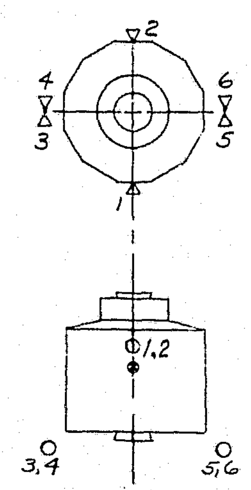
CONCEPT P3

10 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE
- NO TRANSLATION DURING ATTITUDE AND SPIN MANEUVERS

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE ORBITAL	1,5,7	2,6,8	
ACS	4,9 OR 3,10	3 OR 4 9 OR 10	
+ SPIN	6,7	6 OR 7	
- SPIN	5,8	5 OR 8	



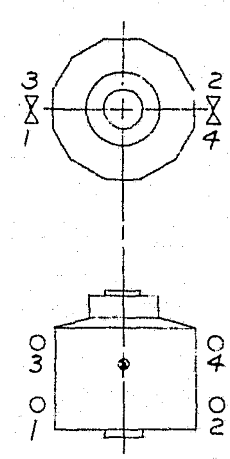
CONCEPT P7

6 ENGINES

FEATURES

- MINIMUM NUMBER OF REA'S FOR NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,3,5	2,4,6	
ACS	2,3,5 OR 1,4,6	1,3,5 OR 2,4,6	
+ SPIN	4,5	4 OR 5	
- SPIN	3,6	3 OR 6	



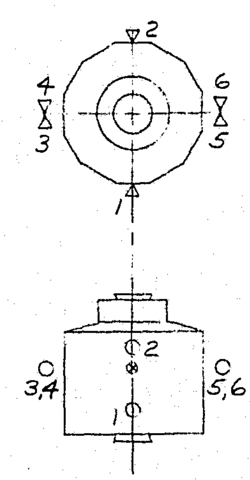
CONCEPT P8

4 ENGINES

FEATURES

- SIMPLEST CONCEPT
- SINGLE REA FAILURE CAUSES SEVERELY DEGRADED ΔV PERFORMANCE

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,2,3,4	1,4 OR 2,3	
ACS	1,3 OR 2,4	1,3 OR 2,4	
+ SPIN	3,4	3 OR 4	
- SPIN	1,2	1 OR 2	



CONCEPT P9

6 ENGINES

FEATURES

- SENSITIVE TO RANDOM REA FAILURE

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	2,4,6	1,3,5	
ACS	1	2	
+ SPIN	4,5	4 OR 5	
- SPIN	3,6	3 OR 6	

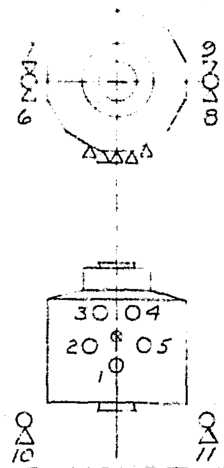
* HIGH PROPELLANT EFFICIENCY

ENGINE PLACEMENT

P3

*ΔV
FAILURE
G ATTITUDE

MARKS



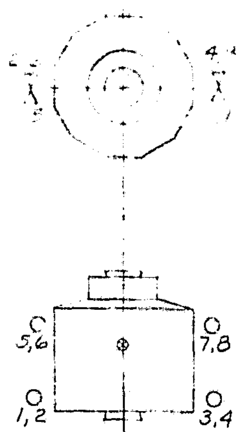
CONCEPT P4

11 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE
- GOOD ΔV RESOLUTION
- CG WITHIN SINGLE REA FAILED POLYGON

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,2,3,4,5	1,2,3,4	
ACS	10 OR 11	6,8 OR 7,9	
+ SPIN	7,8	7 OR 8	
- SPIN	6,9	6 OR 9	



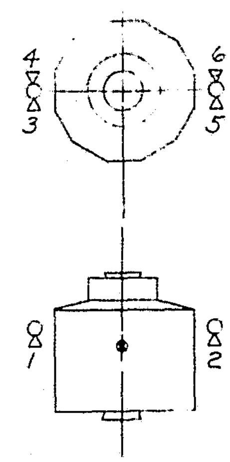
CONCEPT P5

8 ENGINES

FEATURES

- NORMAL MODE* PERFORMANCE WITH ANY TWO REA'S FAILED
- COMMON MODULES

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,3,5,7	2,4,6,8	
ACS	1,6 OR 2,5	4,6, OR 2,8	
+ SPIN	2,3	2,6	
- SPIN	1,4	4,8	



CONCEPT P6

6 ENGINES

FEATURES

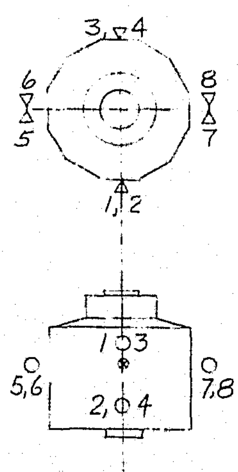
- SIMPLE
- COMMON MODULES

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	3,5	4,6	
ACS	1 OR 2	1 OR 2	
+ SPIN	4,5	4 OR 5	
- SPIN	3,6	3 OR 6	

P9

EA

ARKS



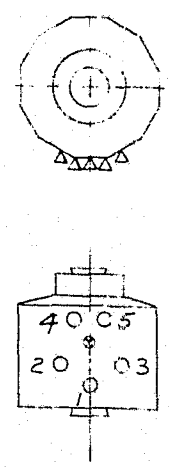
CONCEPT P10

8 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,2,	3,4	
ACS	1,4 OR 2,3	1,2 OR 3,4	
+ SPIN	6,7	6 OR 7	
- SPIN	5,8	5 OR 8	



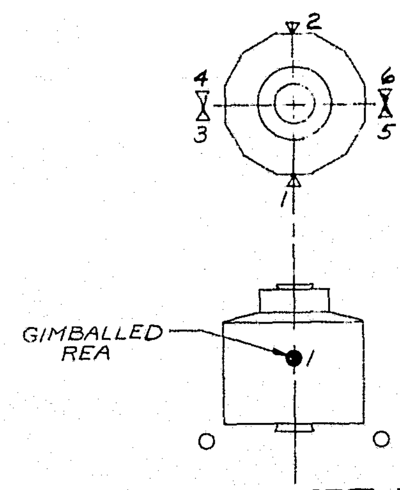
CONCEPT P11

5 ENGINES

FEATURES

- DEGRADED MODE WITH ANY SINGLE REA FAILURE
- MINIMUM REA'S
- CG WITHIN SINGLE REA FAILED POLYGON

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1,2,3,4,5	1,2,3,4	
ACS	1	2,3 OR 4,5	
+ SPIN	3	5	
- SPIN	2	4	



CONCEPT P12

6 ENGINES

FEATURES

- PERFORM NORMAL MODE* ΔV WITH ANY SINGLE REA FAILURE
- GOOD ΔV RESOLUTION

MANEUVER	NORMAL MODE	DEGRADED MODE	REMARKS
MIDCOURSE/ ORBITAL	1	2	
ACS	3,5 OR 4,6	3,5 OR 4,6	
+ SPIN	4,5	4 OR 5	
- SPIN	3,6	3 OR 6	

P_ACEMENT CONCEPTS

PLANETARY EXPLORER
CONTRACT NO. NAS5 11295

Hamilton Standard

FIGURE 4.1-2

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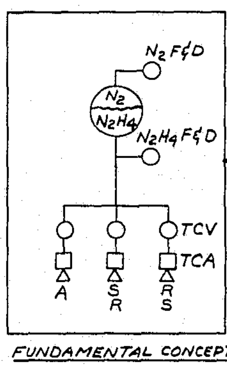
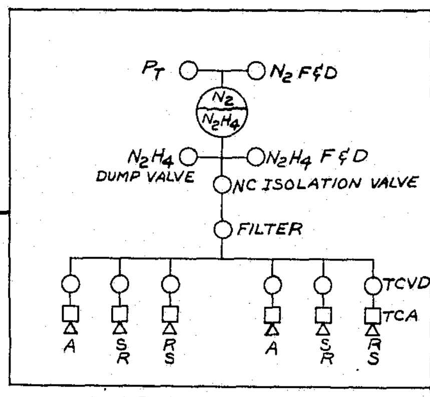


FUNDAMENTAL CONCEPT

BASIC FEED SYSTEM LOGIC DOES NOT SATISFY MISSION REQUIREMENTS

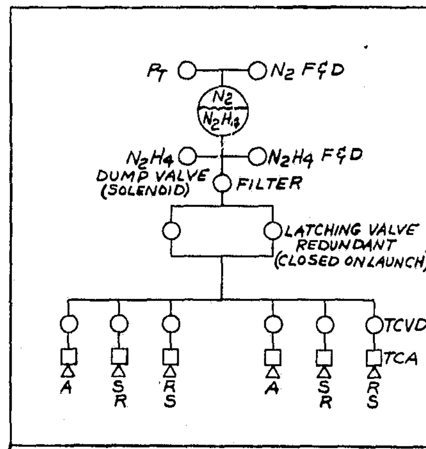
- PERFORMS MANEUVERS
- INEFFICIENT
- NO REDUNDANCY
- DOES NOT SATISFY MIN MISSION REQ'TS



FIRST ITERATION

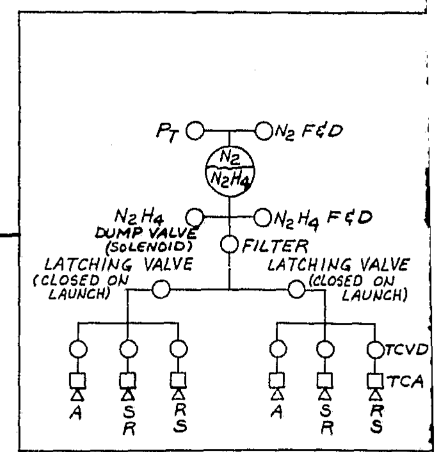
ADDITIONAL COMPONENTS TO SATISFY MISSION REQUIREMENTS

- BACKUP THRUSTERS CONCEPT
- REDUNDANT PROPELLANT VALVE SEAT
- TANK PRESSURE SENSOR
- UPSTREAM ISOLATION
- FILTER
- PROVIDES PREDOMINANT FAILURE MODE PROTECTION(ISOLATION VALVE MUST HAVE REDUNDANT FEATURE)
- F&D VALVE REDUNDANT SEAL
- ADDITIONAL DUMP CAPABILITY



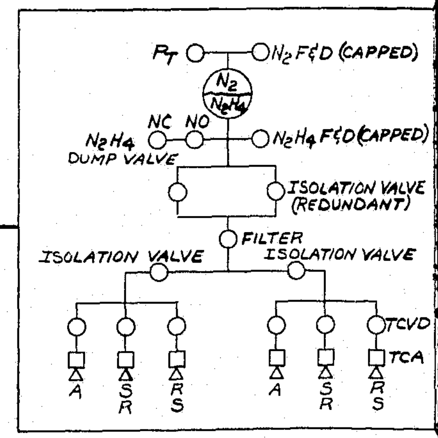
SYSTEM FS-1

- DEFINE ISOLATION DEVICE AS REDUNDANT LATCHING VALVE
- MULTIPLE OPERATIONAL FLEXIBILITY



SYSTEM FS-2

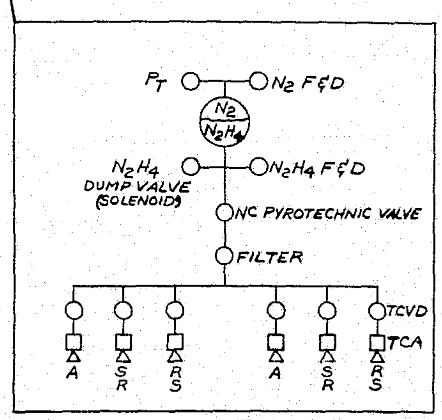
- DEFINE ISOLATION DEVICE AS REDUNDANT LATCHING VALVE
- FAILURE OF ONE LATCHING VALVE FORCE DEGRADED MODE OPERATION
- CAN ISOLATE A FUNCTIONAL GROUP OF IN EVENT OF FAILED OR GROSS LEAK VA



SECOND ITERATION

ADDITIONAL COMPONENTS TO FURTHER INCREASE RELIABILITY

- REDUNDANT ISOLATION VALVE (UPSTREAM
- ISOLATION CAPABILITY OF REA FUNCTIONAL
- PROVIDES PREDOMINANT FAILURE MODE PROTECTION WITH ADDITIONAL REDUNDANCY
- DUMP VALVE POSITIVE SEAL BEFORE AND AFTER ACTUATION



SYSTEM FS-3

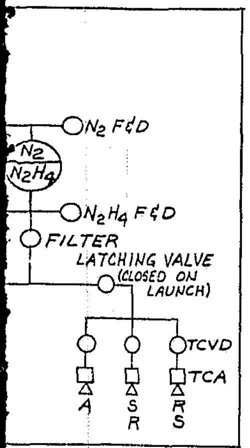
- DEFINE ISOLATION DEVICE AS REDUNDANT PYROTECHNIC VALVE
- FLOW CANNOT BE TERMINATED IN EVENT OF LEAKAGE

~NOTE~

- FILTER POSITIONED IN ALL SYSTEM SCHEMATICS TO PROTECT CONTAMINANT SENSITIVE COMPONENTS.
- LIQUID DUMP CAPABILITY HAS BEEN CONSIDERED HEREIN ALTHOUGH FURTHER STUDY MAY RESULT IN THE USE OF OTHER METHODS.
- ALL FILL AND DRAIN VALVES PROVIDED WITH REDUNDANT SEALS.
- REDUNDANT FEATURE OF PYROTECHNIC VALVES MAY BE DUAL VALVES, SINGLE VALVE-DUAL SQUIBS OR SINGLE VALVE-ONE SQUIB WITH DUAL BRIDGEWIRE
- REA AND PROPELLANT TANK QUANTITIES NOT SHOWN

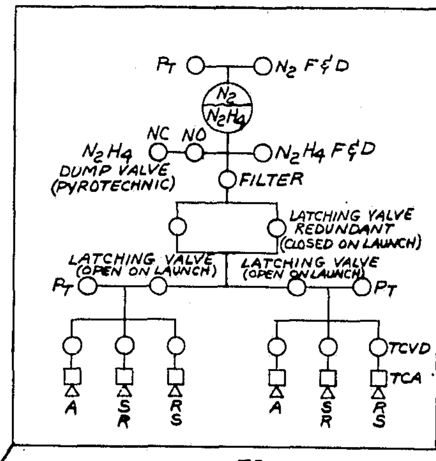
~LEGEND~

A — ATTITUDE
S — SPIN
R — RADIAL
P_T — PRESSURE TRANSDUCER
TCA — THRUST CHAMBER ASSEMBLY
TCV — THRUST CHAMBER VALVE
TCVD — THRUST CHAMBER VALVE-DUAL
NC — NORMALLY CLOSED
NO — NORMALLY OPEN
F&D — FILL AND DRAIN VALVE



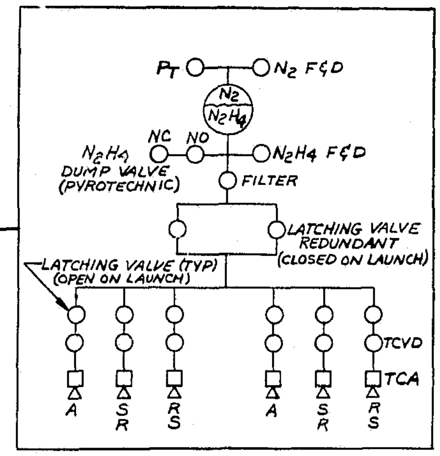
FS-2

ON DEVICE AS
CHING VALVE
LATCHING VALVE FORCES
DE OPERATION
FUNCTIONAL GROUP OF REA'S
LED OR GROSS LEAK VALVE



SYSTEM FS-4

- QUAD REDUNDANT LATCHING VALVES PROVIDE SINGLE MALFUNCTION PROTECTION
- DOWNSTREAM LATCHING VALVES PROVIDE FUNCTIONAL GROUP
- DOWNSTREAM PRESSURE SENSORS FOR DIAGNOSTIC PURPOSES

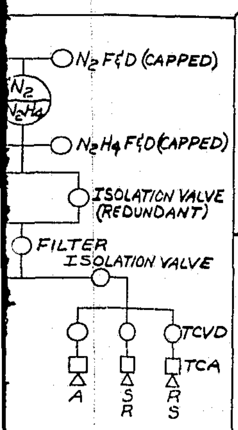


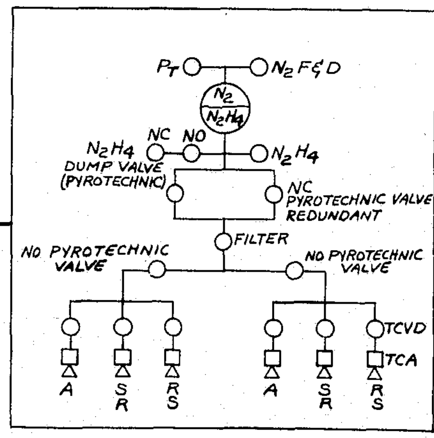
SYSTEM FS-5

- PROVIDES SINGLE MALFUNCTION PROTECTION
- FLEXIBLE SINGLE REA ISOLATION
- ADDITIONAL NORMAL MODE CAPABILITY
- MORE COMPLEX

FOLDOUT FRAME

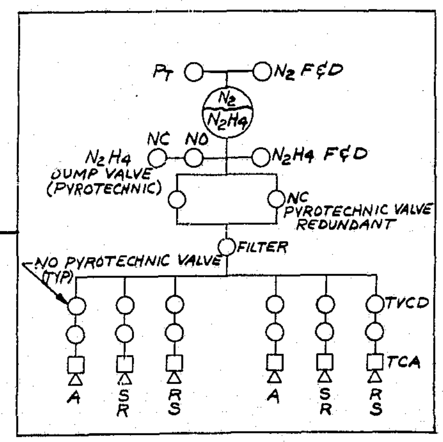
2





SYSTEM FS-6

- DOWNSTREAM PYROTECHNIC VALVES PROVIDE FUNCTIONAL GROUP ISOLATION
- MUST PERFORM IN DEGRADED MODE UPON ACTUATION OF PYROTECHNIC VALVES



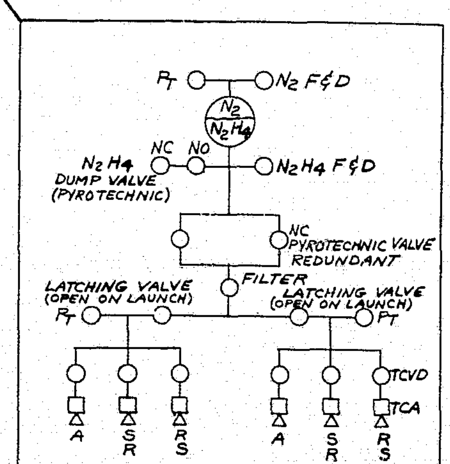
SYSTEM FS-7

- PERMANENT SINGLE REA ISOLATION
- LIMITED ADDITIONAL NORMAL MODE CAPABILITY
- MORE COMPLEX

RATION

ENTS TO FURTHER
IABILITY

TION VALVE (UPSTREAM)
LITY OF REA FUNCTIONAL GROUP
INANT FAILURE MODE
DDITIONAL REDUNDANCY
VE SEAL BEFORE AND



SYSTEM FS-8

- ISOLATION FUNCTION ACCOMPLISHED WITH BOTH LATCHING AND PYROTECHNIC DEVICES
- DOWNSTREAM LATCHING VALVES PROVIDE FUNCTIONAL GROUP ISOLATION
- DOWNSTREAM PRESSURE SENSORS FOR DIAGNOSTIC PURPOSES

END —
DE

URE TRANSDUCER
T CHAMBER ASSEMBLY
ST CHAMBER VALVE
T CHAMBER VALVE-DUAL SERIES SEAT
LLY CLOSED
ALLY OPEN
ND DRAIN VALVE

PROPULSION SUBSYSTEM
SCHEMATIC CONCEPTS

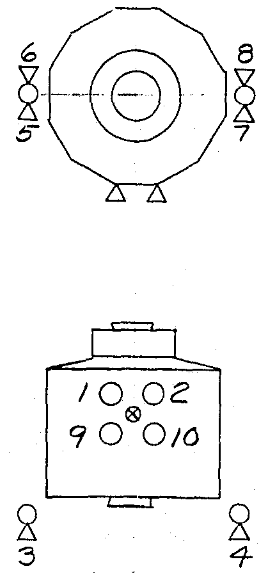
PLANETARY EXPLORER
CONTRACT NO. NAS5-11295

Hamilton Standard

FIGURE 4.1-3

RECOMMENDED ENGINE PLACEMENTS
SYSTEMS FOR ST

ENGINE PLACEMENT CONCEPT



CONCEPT P1

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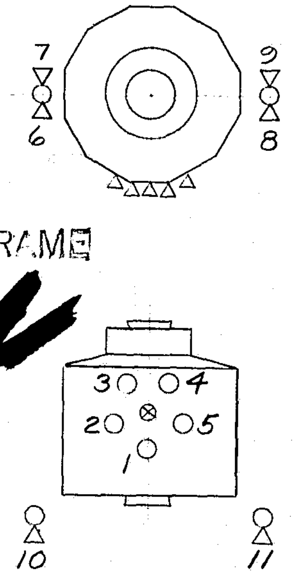
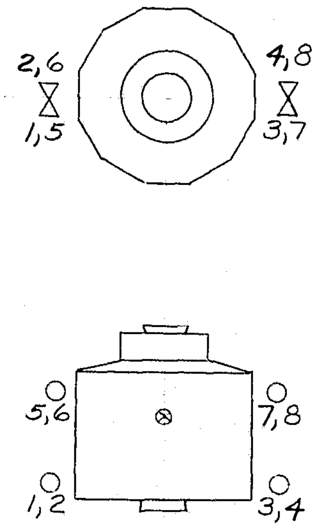
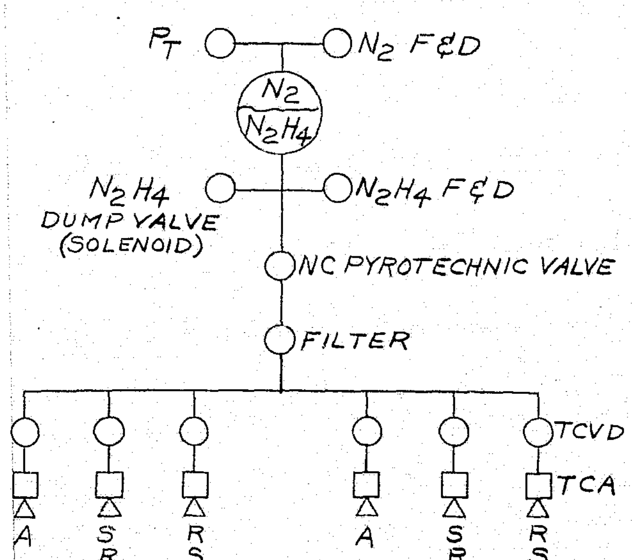


CONCEPT P4

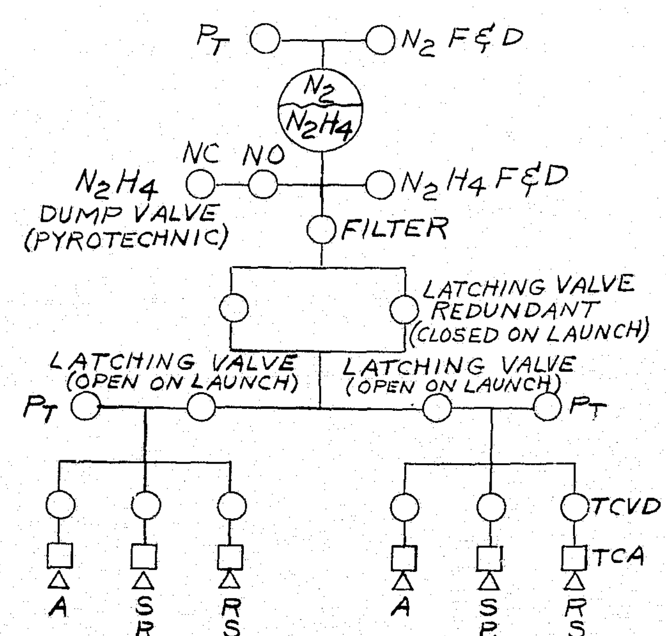


CONCEPT P5

UPSTREAM FEED SYSTEMS



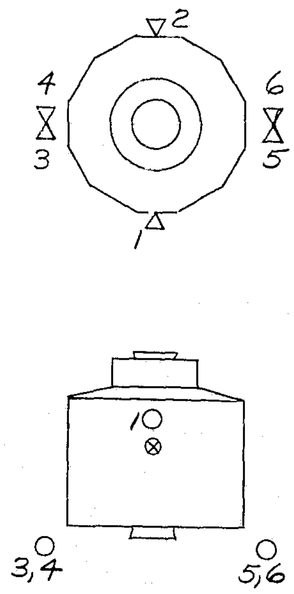
SYSTEM FS-3



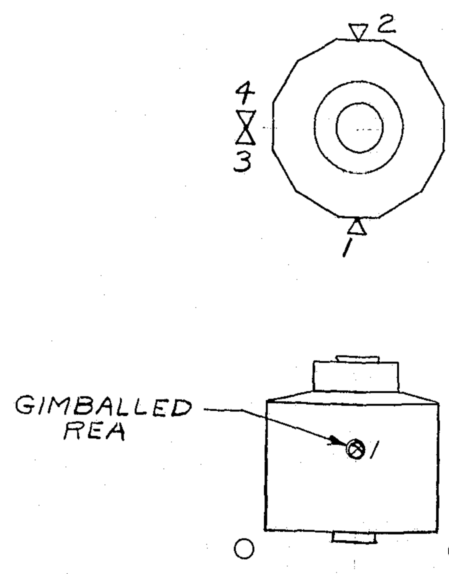
SYSTEM FS-4

EMENTS AND UPSTREAM FOR STUDY

CONCEPTS



CONCEPT P7



CONCEPT P12

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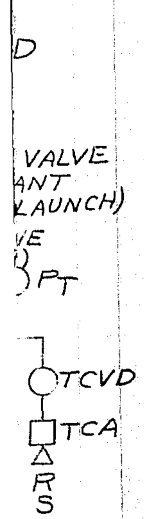
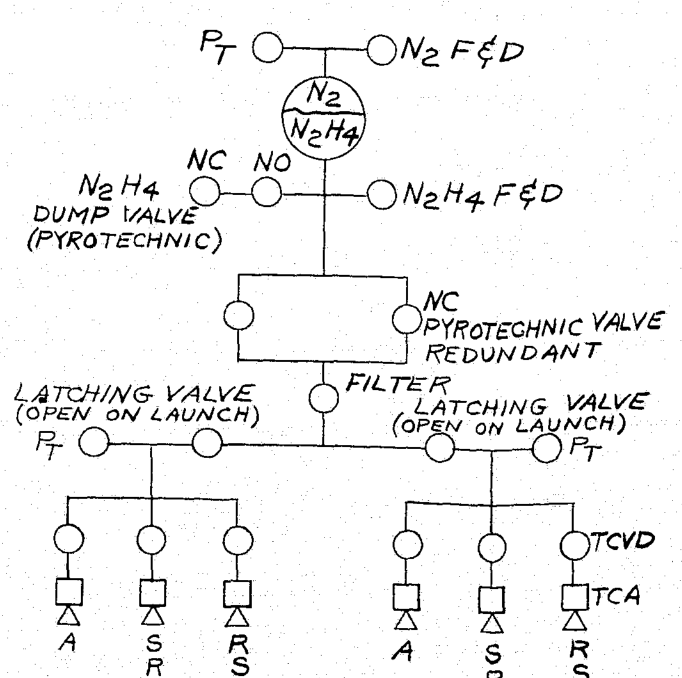
YSTEMS





SYSTEM FS-8

PLANETARY EXPLORER
CONTRACT NO. NAS5-11295

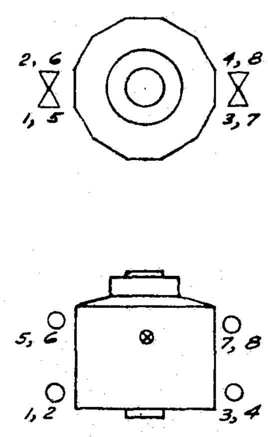
Hamilton Standard

FIGURE 4.1-4

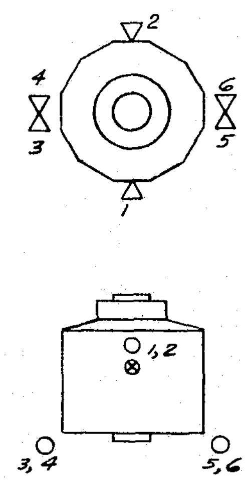
Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

SELECTED ENGINE PLACEMENTS AND UP

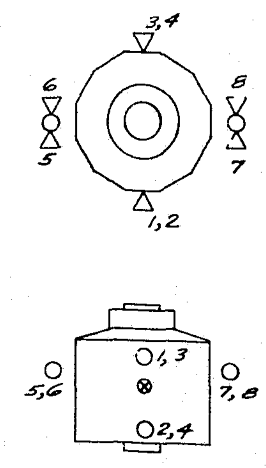
ENGINE PLACEMEN



CONCEPT P5

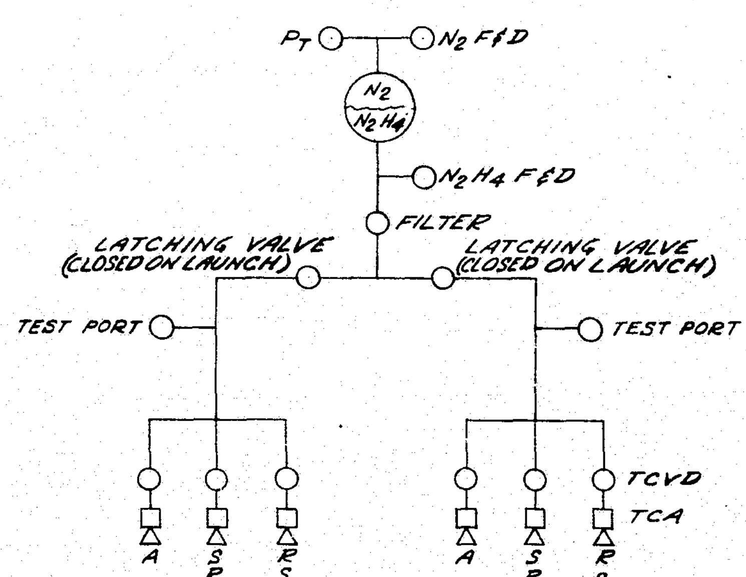


CONCEPT P7

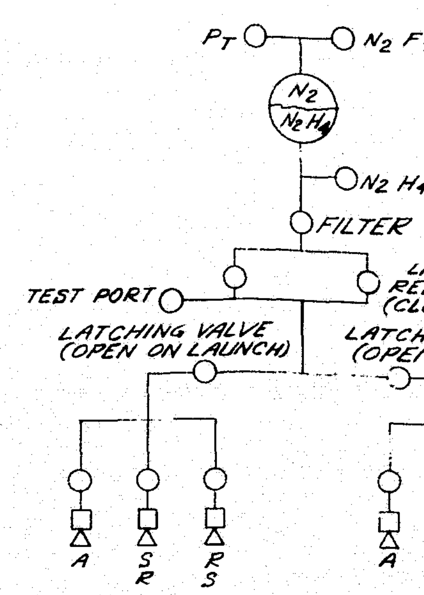


CONCEPT P 10

UPSTREAM FEED



SYSTEM FS-2



SYSTEM FS-4

TS AND UPSTREAM SYSTEMS FOR STUDY

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LACEMENT CONCEPTS

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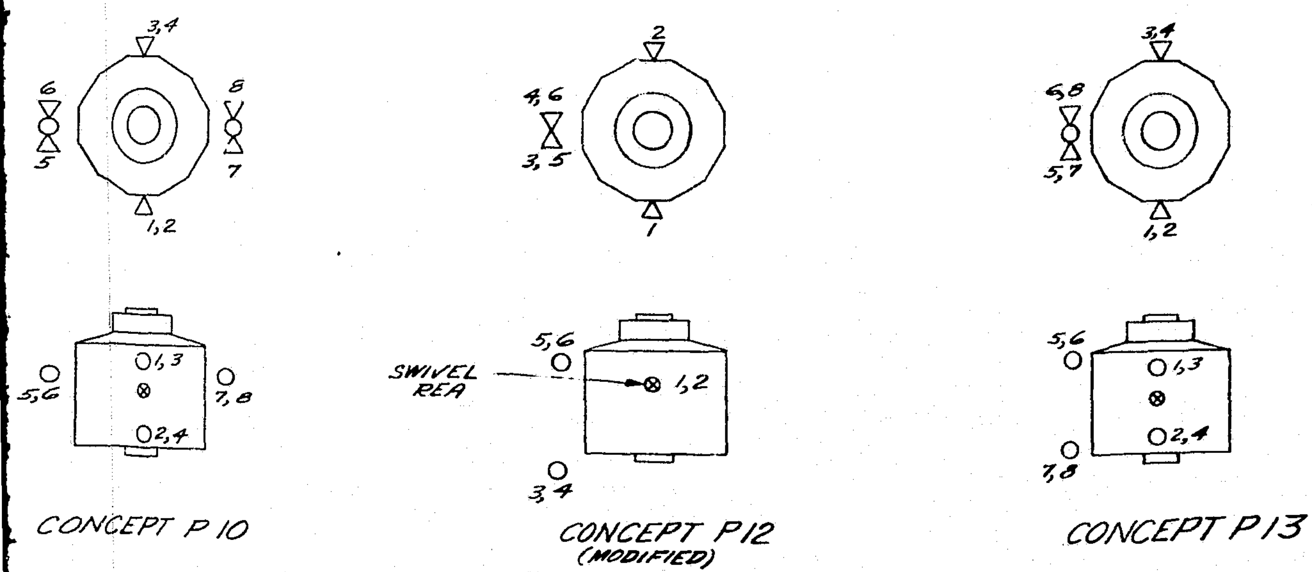
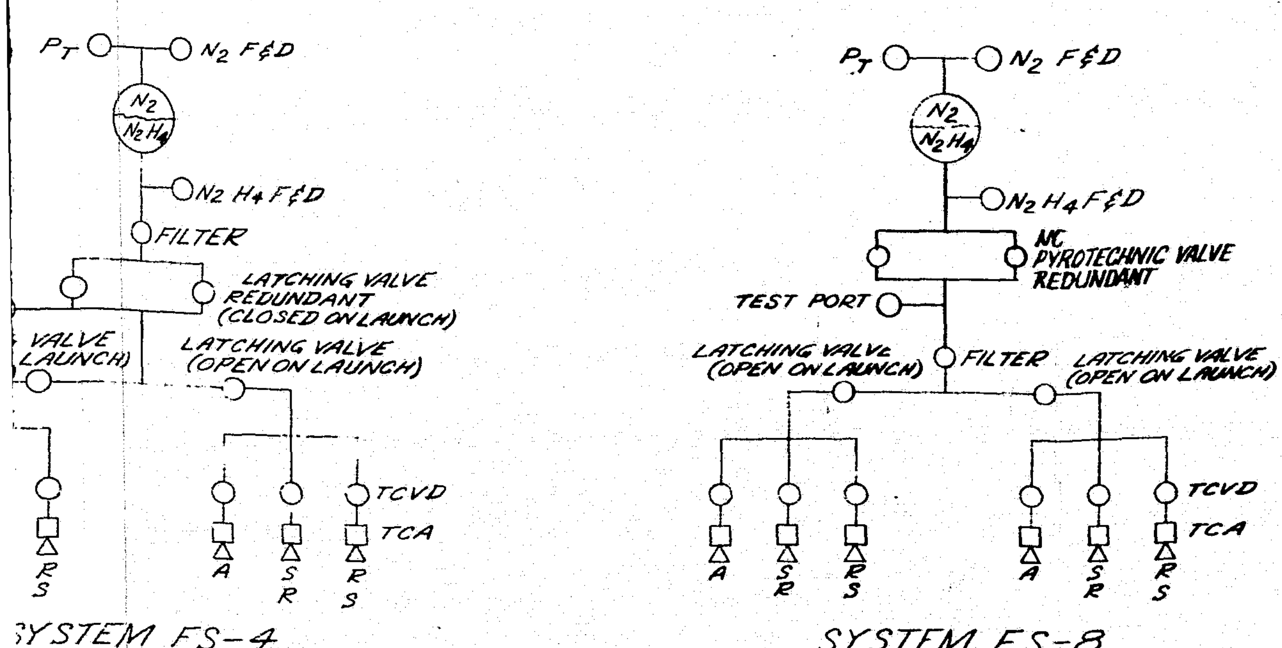


AM FEED SYSTEMS



PLANETARY EXPLORER
CONTRACT NO. NAS5-11295

FIGURE 4.1-5

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TABLE 4.1-I
EVALUATION CRITERIA FOR ENGINE PLACEMENT CONCEPTS

Concept	No. of Engines	No. of Modules	No. of Engines Required for Type of Maneuver: ΔV	Attitude	Spin	Peak Power Req'd Per ΔV Firing Maneuver	ΔV Propellant Efficiency with Any Single Engine Failure	Adaptability of Functional Engine Grouping to Perform Maneuvers with 50% of Engines
P1	10	3	4(3)	1	2(1)	High	Good	Good
P2	8	4	3	1	2(1)	Moderately High	Good	Good
P3	10	3	3	2(1)	2(1)	Moderately High	Good	Good
P4	11	3	5	1	2(1)	High	Good	Good
P5	8	4	4	2(1)	2(1)	High	Good(2)	Excellent
P6	6	2	2	1	2(1)	Moderate	Low	Poor
P7	6	4	3	2(1)	2(1)	Moderately High	Good	Medium
P8	4	4	4	2(1)	2(1)	High	Poor	—
P9	6	4	3	2(1)	2(1)	Moderate	Low	Poor
P10	8	6	2	2(1)	2(1)	Moderate	Good	Excellent
P11	5	5	5	1	1	High	Low	Poor
P12	6	4	1(3)	2	2(1)	Low	Good	Medium

(1) Pure couple
(2) Can perform all mission maneuvers with any two engines failed closed
(3) Good resolution

TABLE 4.1-III
RELATIVE RATING OF SELECTED ENGINE PLACEMENT CONCEPTS

Concept	No. of Engines	No. of Modules	No. of Engines Required for Type of Maneuver: ΔV	Attitude	Spin	Peak Power Req'd Per ΔV Firing Maneuver	ΔV Propellant Efficiency with Any Single Engine Failure	Adaptability of Functional Engine Grouping to Perform Maneuvers with 50% of Engines
P5	8	4	4	2(1)	2(1)	High	Good(2)	Excellent
P7	6	4	3	3(1)	2(1)	Moderately High	Good	Good
P10	8	6	2	2(1)	2(1)	Moderate	Good	Excellent
P12 Modified	6	4	1	2(1)	1 or 2	Low	Good	Excellent
P13	8	6	2	2(1)	1 or 2	Moderate	Good	Excellent

(1) Pure couple
(2) Can perform all maneuvers with any two engines failed

FOLDOUT FRAME

2

TABLE 4.1-II
EVALUATION CRITERIA FOR FEED SYSTEM CONCEPTS

Concepts	Positive Propellant Isolation From Engines During Launch & Test	Isolation of Engine Functional Groups	Multiple Operational Capability of Propellant Isolation Device	Series Redundant Isolation Valves	Number of Isolation Valves	Diagnostic[tt] Capability	Operational Complexity
FS-1	No	No	Yes	No	2	No	Low
FS-2	No	Yes	Yes	No	2	Yes	Low
FS-3	Yes	No	No	No	1	No	Low
FS-4	No	Yes	Yes	Yes	4	Yes	Moderate
FS-5	No	Yes*	Yes	Yes	8[t]	No	High
FS-6	Yes	Yes**	No	No	4	No	Moderate
FS-7	Yes	Yes**	No	No	8[t]	No	High
FS-8	Yes	Yes	Yes	No	4	Yes	Moderate

Notes:
- t For a six engine placement concept
- tt Downstream pressure sensing
- * Isolation valve back-up to each engine control valve
- ** Determination of which ordnance valve to close difficult

TABLE 4.1-IV
PROPULSION SUBSYSTEM CONCEPTS SELECTED AS CANDIDATES

Candidate Subsystem	Engine Placement Concept	Feed System Concept
I	P-5	FS-2
II	P-5	FS-4
III	P-5	FS-8
IV	P-7	FS-2
V	P-7	FS-4
VI	P-7	FS-8
VII	P-10	FS-2
VIII	P-10	FS-4
IX	P-10	FS-8
X	P-12M	FS-2
XI	P-12M	FS-4
XII	P-12M	FS-8
XIII	P-13	FS-2
XIV	P-13	FS-4
XV	P-13	FS-8

4.2 Description of Candidate Systems Selected

The 15 candidate systems selected consist of combinations of 5 different engine placement concepts and 3 different feed system concepts. Table 4.1-IV identifies the combination of engine placement concept and feed system concept which distinguishes each candidate system.

The propulsion subsystem layout drawings illustrated in Figures 4.2-1a through 4.2-1e represent the 5 different engine placement concepts and each layout drawing shows schematically each of the 3 feed system concepts. The only difference in systems with the same feed system concept being in the line routing to the engines. The component layout for each of the selected feed systems is shown in the auxiliary views of the component panel, and the propellant tank arrangement for both the Orbiter and Probe configurations are illustrated, with the Probe tank arrangement in the reduced scale alternate view. The actual difference between the Orbiter and the Probe engine module locations is dependent upon the physical configuration of the spacecraft, and the location of the vehicle center of gravity.

In all of the propulsion subsystems, the bladderless propellant tanks are installed in the spacecraft tankage bay and manifolded together on both the pressurant and propellant outlet ports. The tanks represented in the layouts are the Fansteel Inc. three port tanks used on the IDCSP/A spin stabilized satellite which are suitable for the Planetary Explorer application with respect to both the volumetric capacity and the porting requirements. The pressurant manifold provides pressure equalization between the tanks during operation, and is connected to the pressurant fill and vent valve located on the component panel to perform the fill and vent operations. This portion remains the same for all candidate systems. The tank dual propellant outlet port provides the capability to drain propellant during ground testing of the propulsion subsystem and during flight when the spacecraft is spinning. The port arrangement also permits draining the system for off loading and provides the capability to flow flushing fluids through the system by flowing into the pressurant fill and vent port and out the propellant fill and drain port. The tank outlet ports are connected to a circumferential propellant manifold with a line going to the component panel where the propellant fill and drain valve, pressure transducer, filter and isolation valves are located. The number and type of isolation valves is represented by the three component panel arrangements illustrated in the auxiliary views. Propellant lines are then routed to the respective engines and are manifolded into two basic subsystems controlled by latching valves permitting isolation of one or both of the subsystems from the propellant supply. Each of the candidate subsystems has the engines necessary to perform the velocity correction, attitude control and spin control maneuvers required.

The engine locations conform to the positions represented in Figure 4.1-5 for each of the layouts. Engines required at the ends of the vehicle, either radial or tangential, are located such that they are mounted to brackets or supports on the ends of the spacecraft with no penetration through the solar arrays. Positioning the engines in this manner permits ease of installation and maintenance. There should be no interference with the spacecraft antenna since the engines shown have a low profile. In subsystem concepts where radial and tangential engines are located

4.2 (continued)

within the spacecraft, these engines are presently shown not protruding through or interfering with the solar arrays. Where engines protrude through the spacecraft, the engines are shown semi-buried where the valve and mounting structure are internal and, in the case of the gimballed engine, the gimbal actuator is internal.

Since all the candidate subsystems are similar except for the number of engine modules, the number of components on the component panel and the propellant line routing to the engine modules, the basic installation method for each of the subsystems into the spacecraft is considered the same. The engine modules being removable from the subsystem facilitates the installation of a completely fabricated subsystem, less the engines, into the spacecraft. The subsystem components are assembled and positioned on a fixture representing the vehicle mechanical interface mounts. This fixture may be either removable, with the support of the subassembly transferred to a handling fixture, or the handling fixture utilized both for assembly and handling. The integrated subassembly, consisting of the tanks, component panel with components mounted, and the propellant lines which mate with the engines attached to the handling fixture can be "dropped" into the vehicle frame and mounted at the subsystem/vehicle interfaces. During installation of the subsystem into the spacecraft, propellant lines required for any aft mounted engines can be routed through clearance holes in lateral bulkheads of the vehicle. The engine modules are installed after the integrated system has been mounted to the spacecraft and mechanical joints are provided for the tubing connections. Figure 4.2-2 illustrates a typical installation of a propulsion subsystem into the spacecraft. Candidate system Number I has been shown as a typical example in this illustration.

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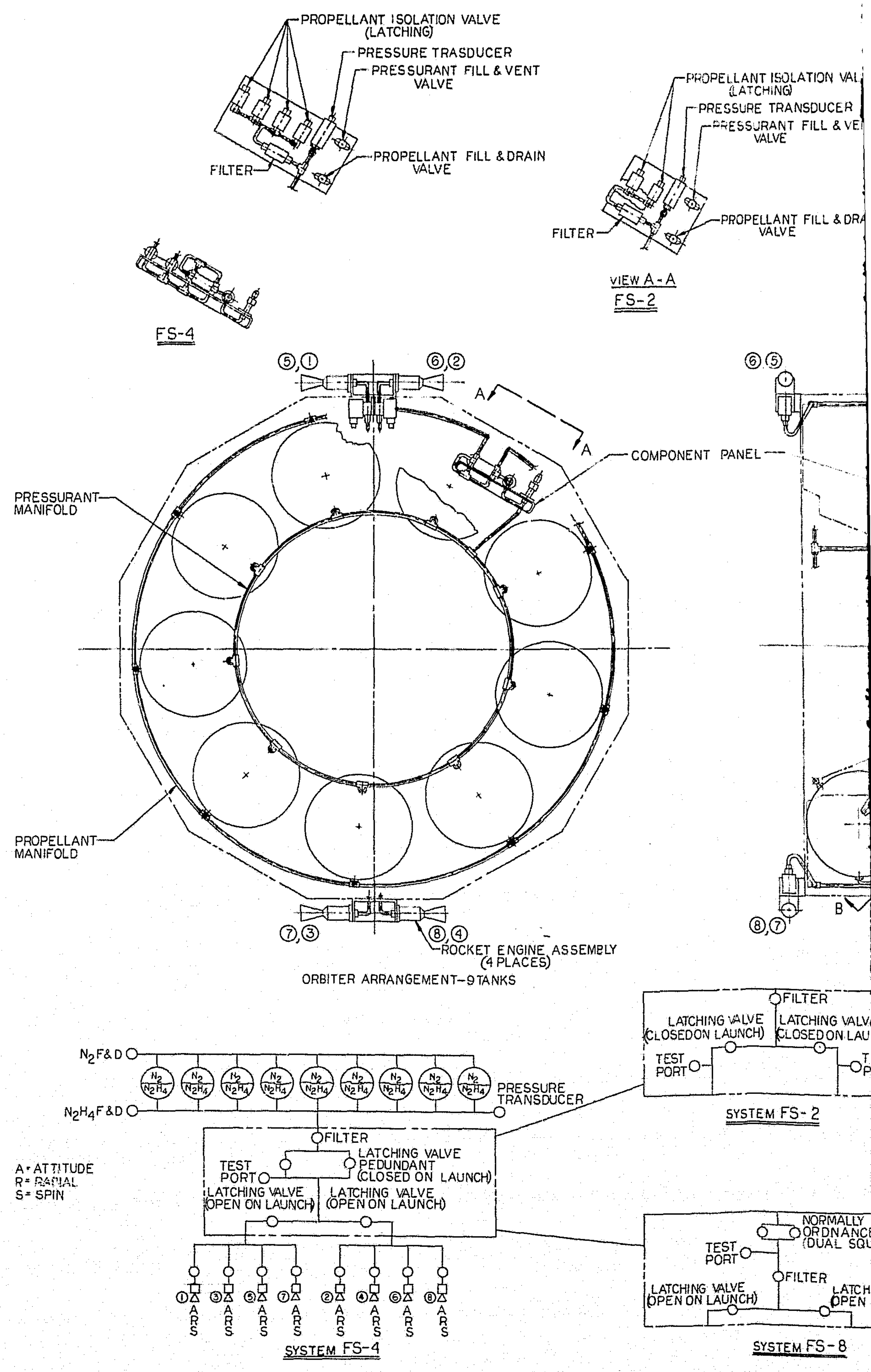
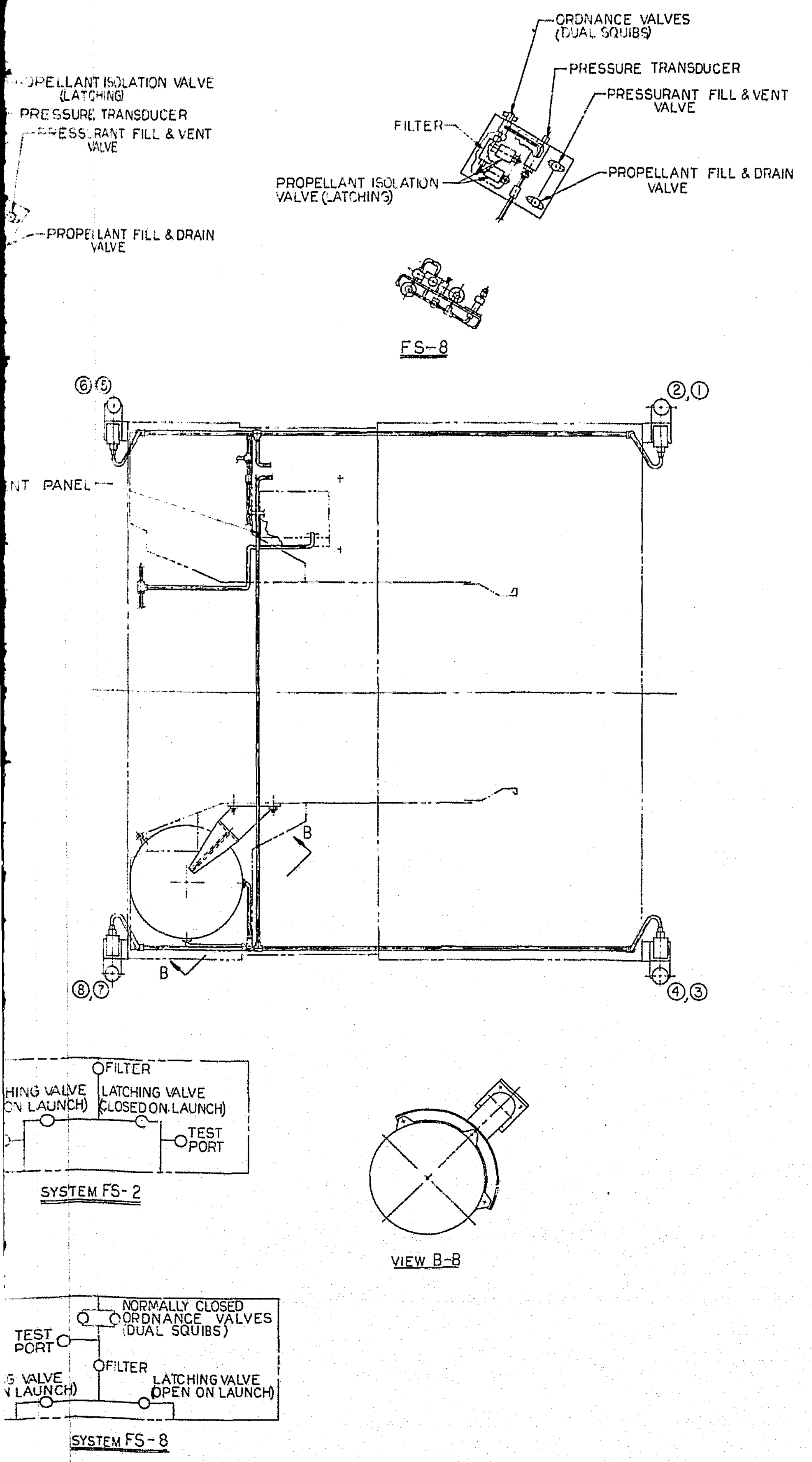
FOLDOUT FRAME 1





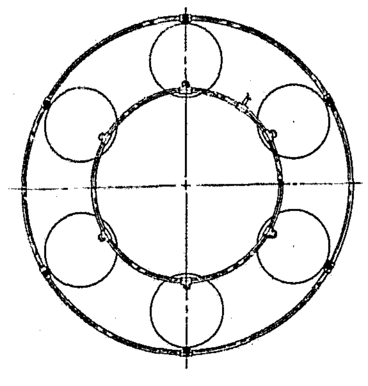


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FIGURE 4.2–1A. CANDIDATE PROPULSION SUBSYSTEM P–5

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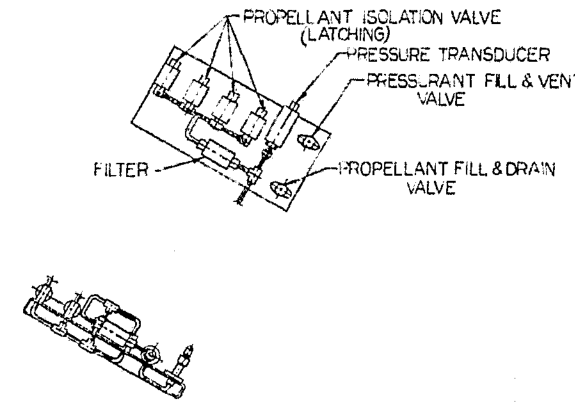
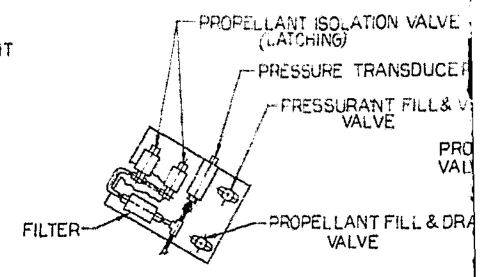
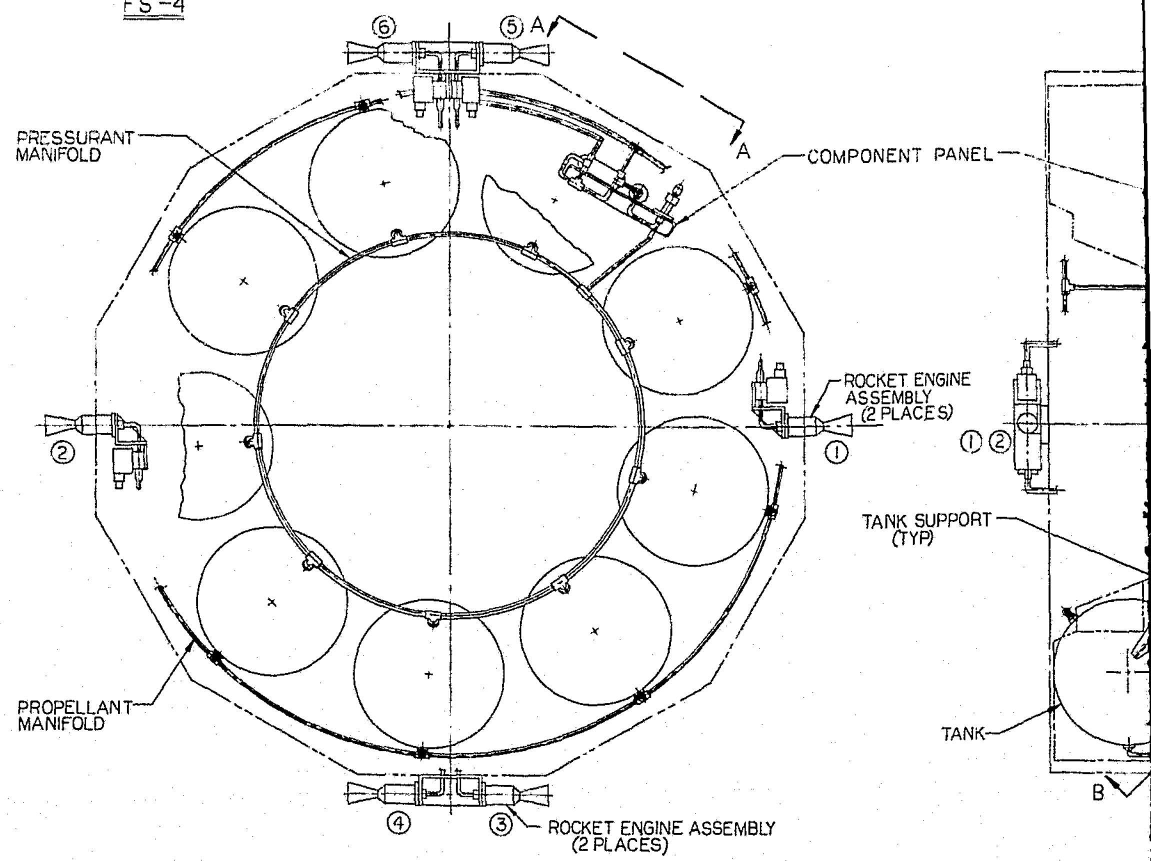


FS-4

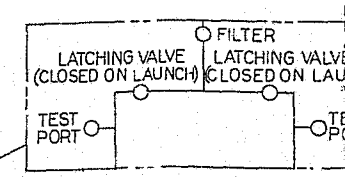


VIEW A-A

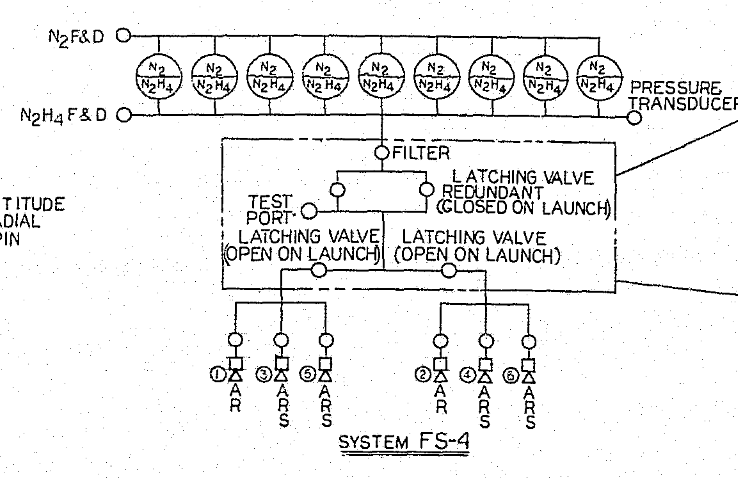
FS-2



ORBITER ARRANGEMENT–9 TANKS

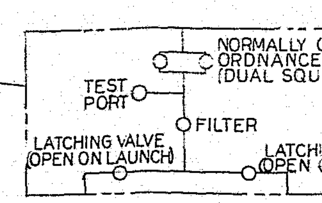


SYSTEM FS-2



A=ATTITUDE
R=RADIAL
S=SPIN

SYSTEM FS-4



SYSTEM FS-8

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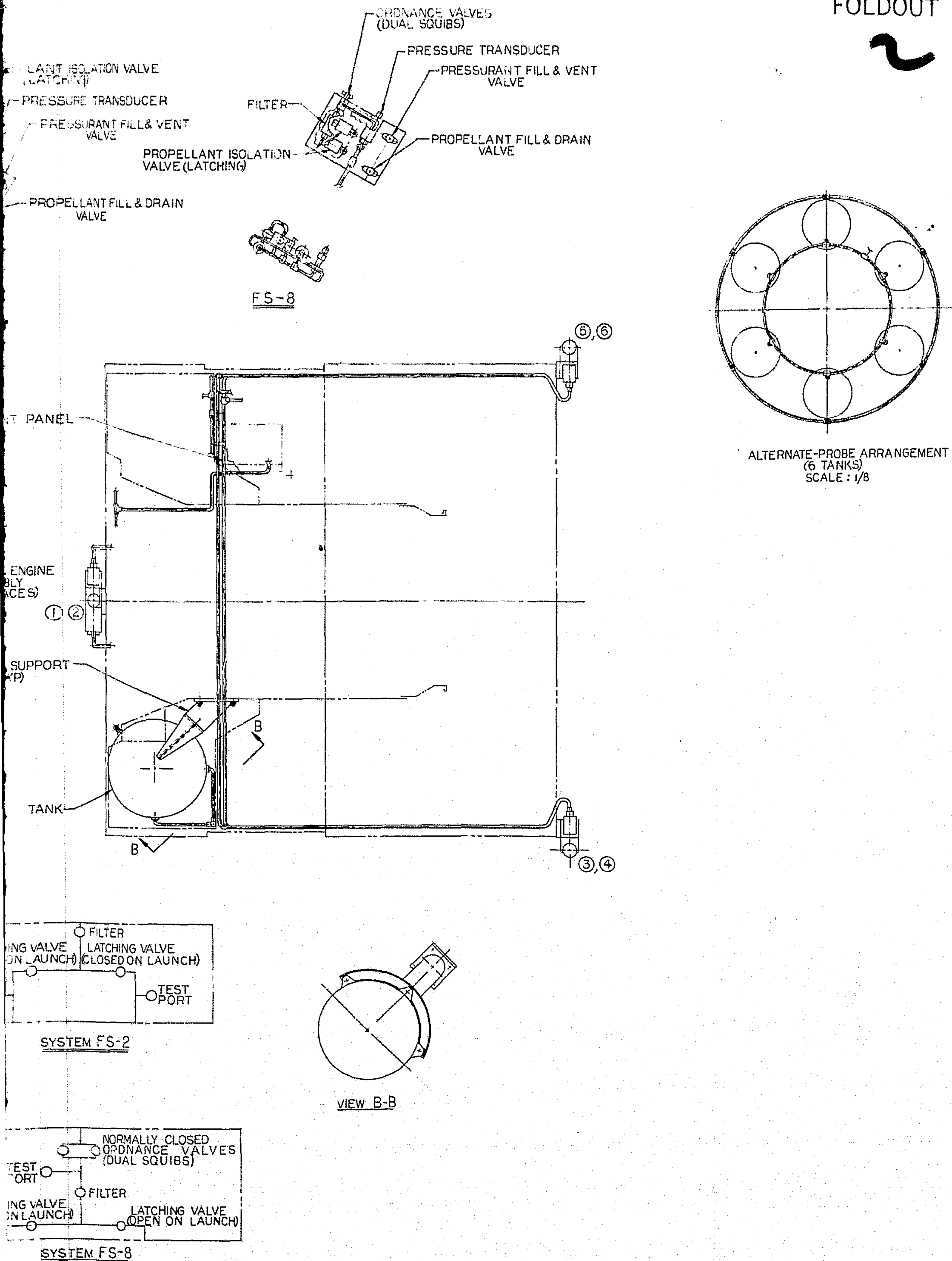

FIGURE 4.2–1B. CANDIDATE PROPULSION SUBSYSTEM P–7

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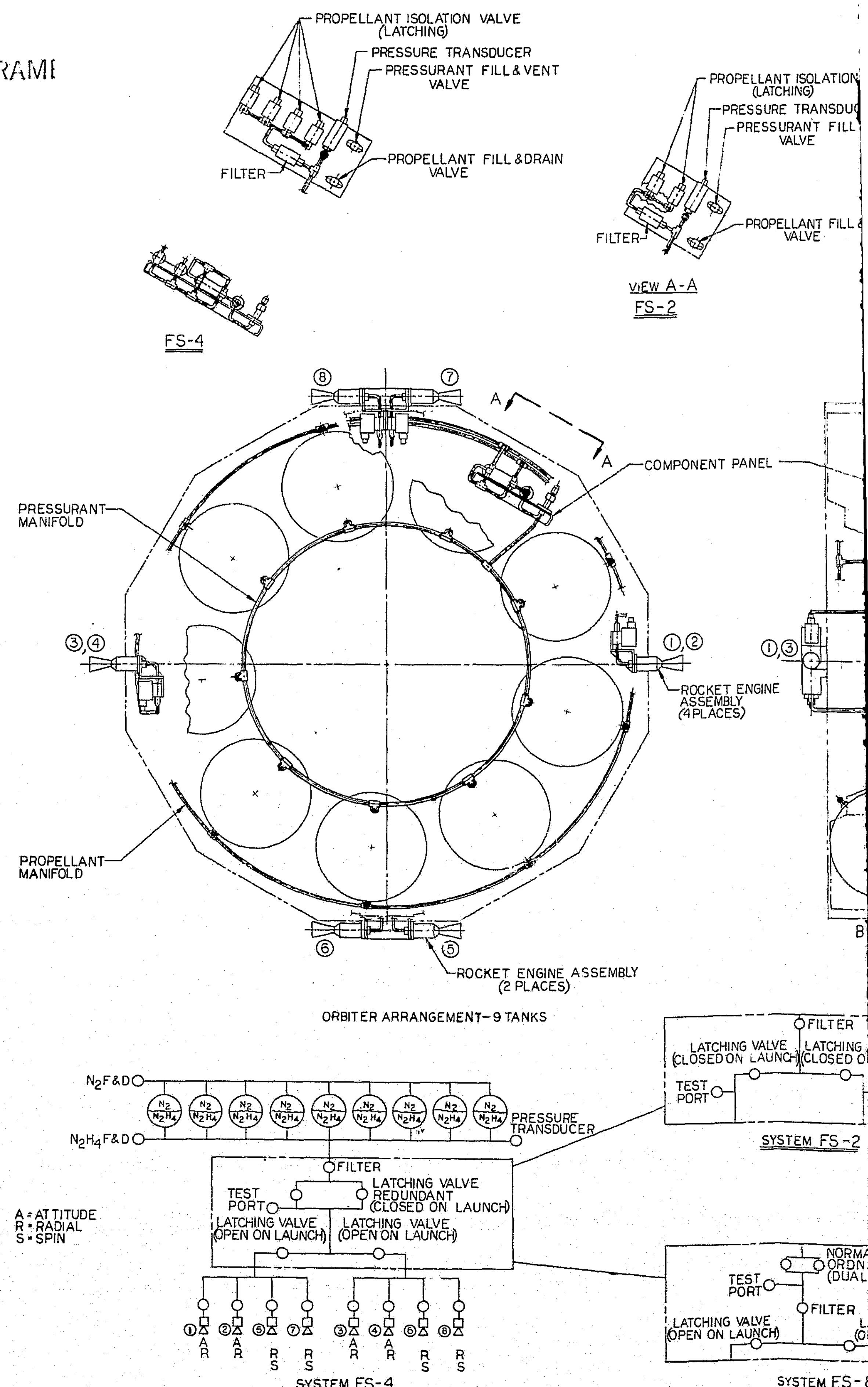

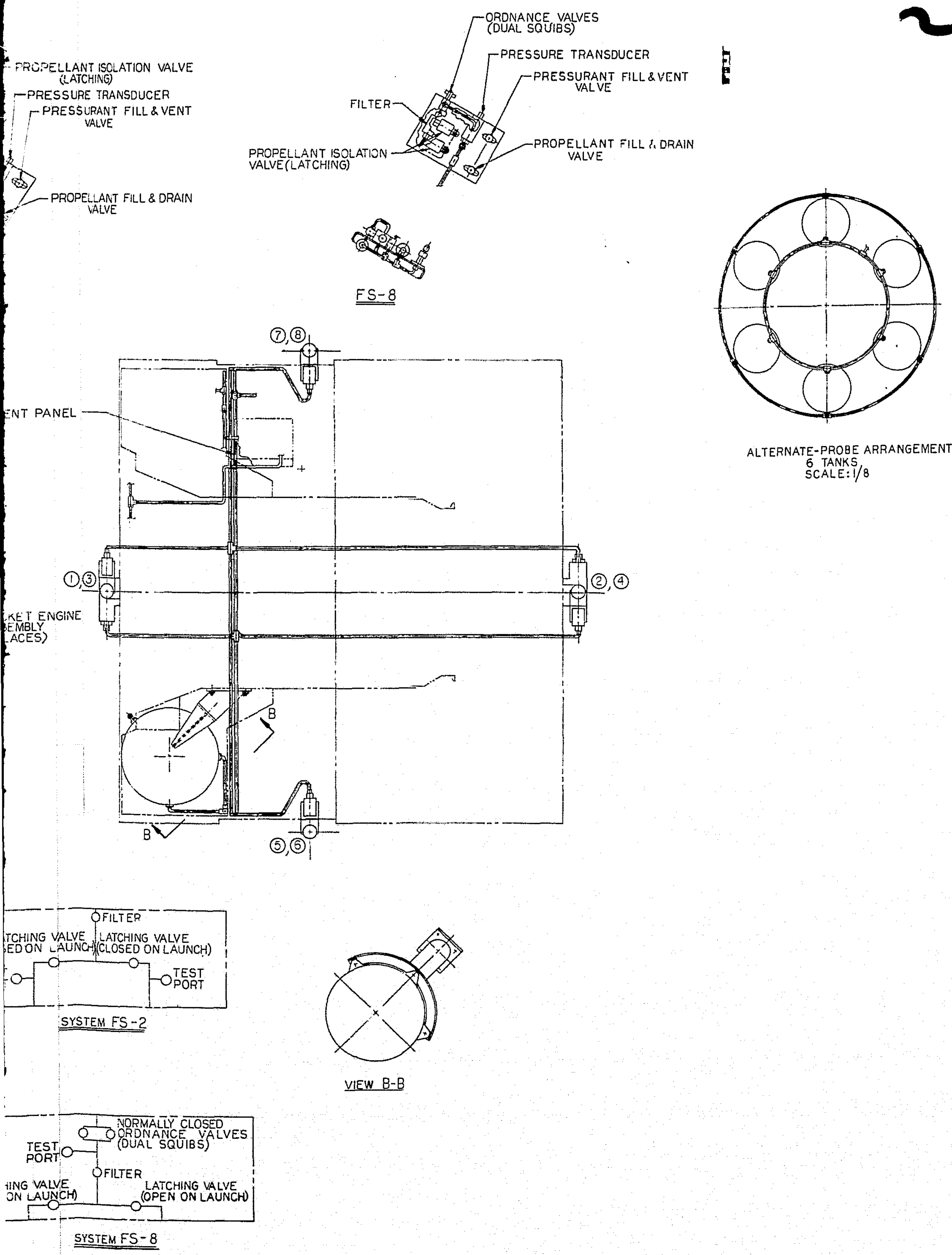

FIGURE 4.2-1C. CANDIDATE PROPULSION SUBSYSTEM P-10

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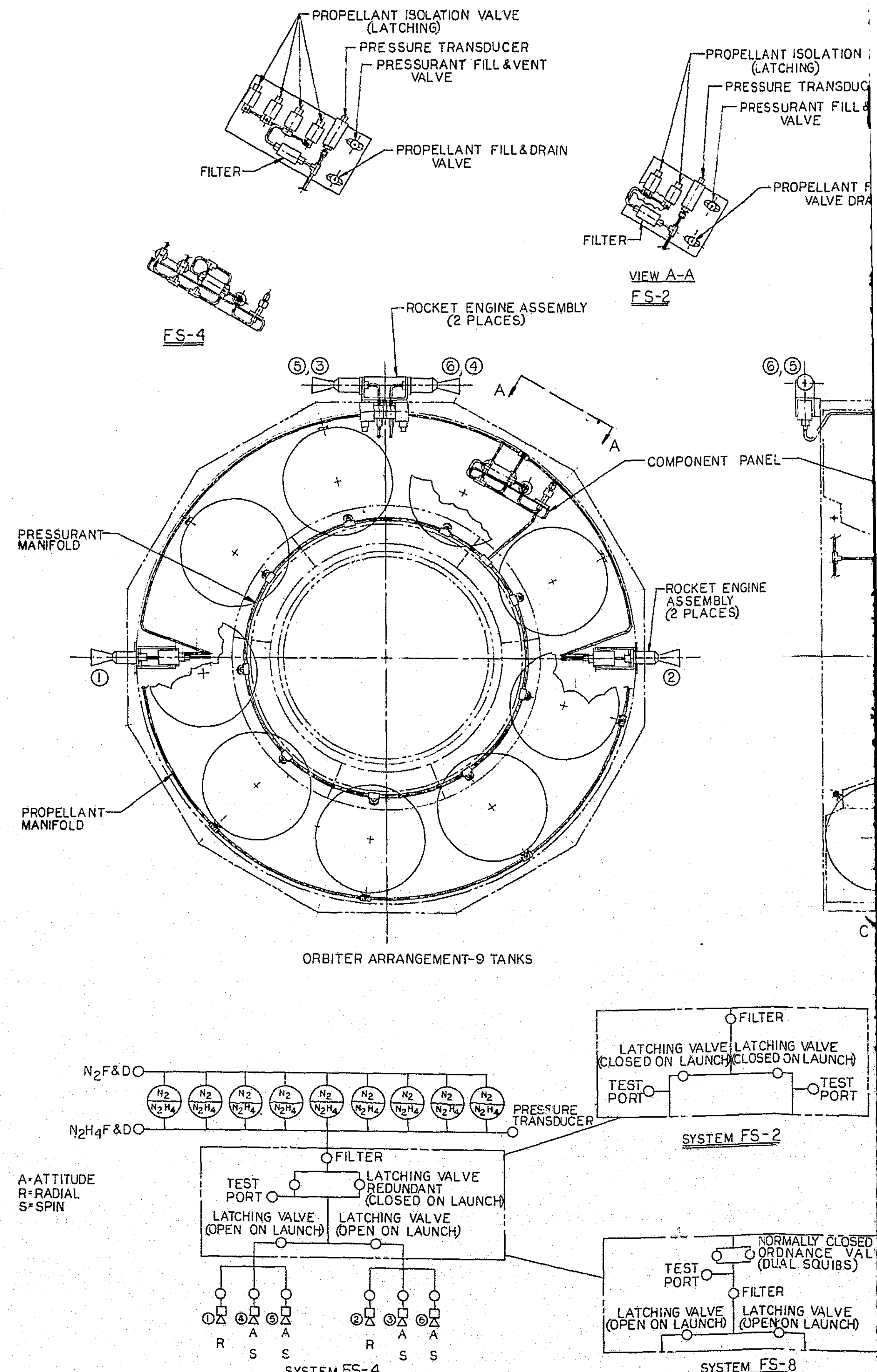

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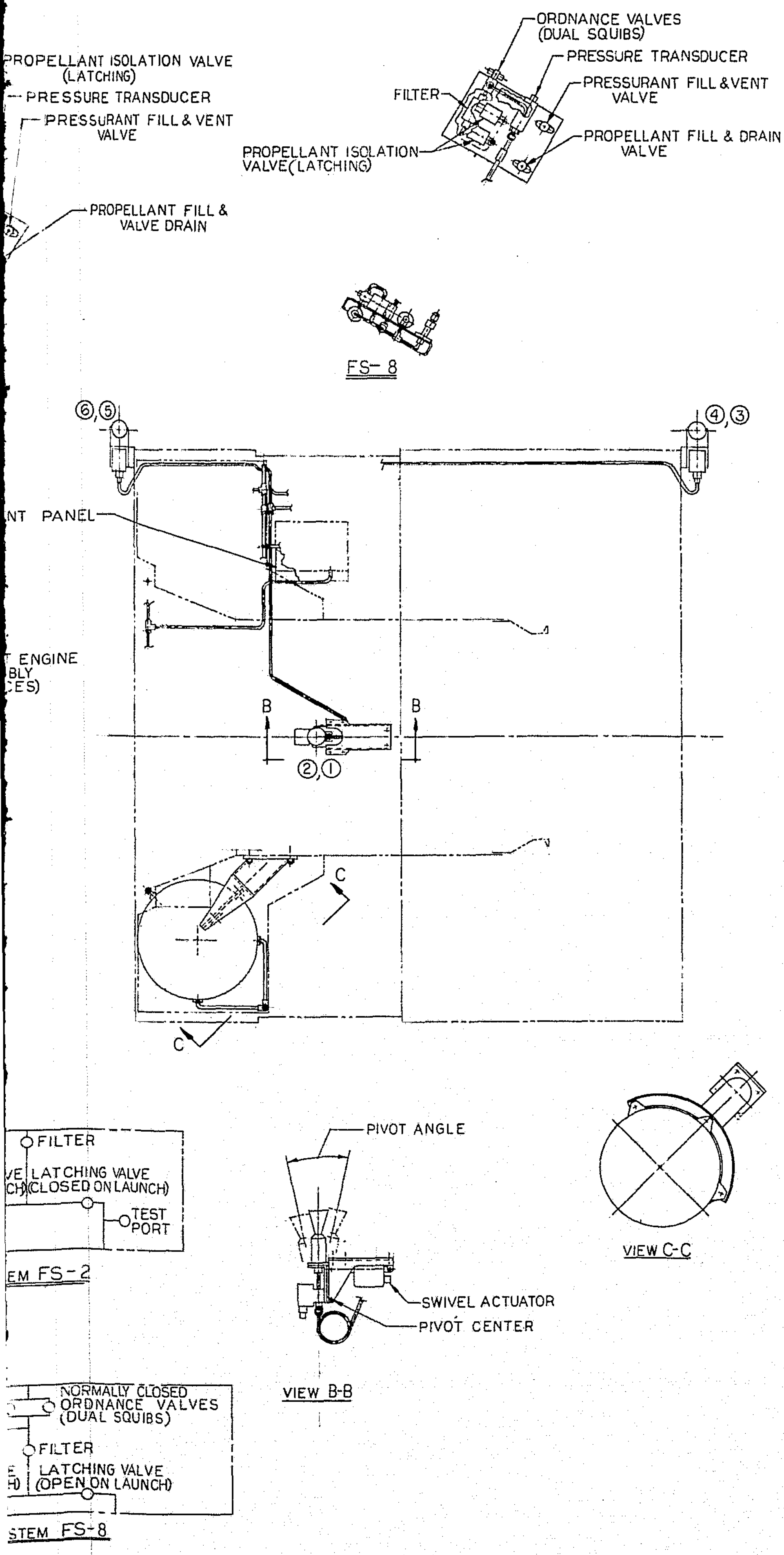

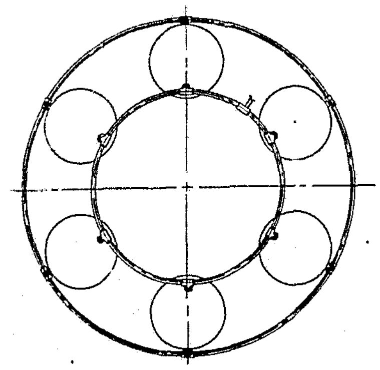

ALTERNATE-PROBE ARRANGEMENT
6 TANKS
SCALE: 1/8

FIGURE 4.2-1D. CANDIDATE PROPULSION SUBSYSTEM P-12

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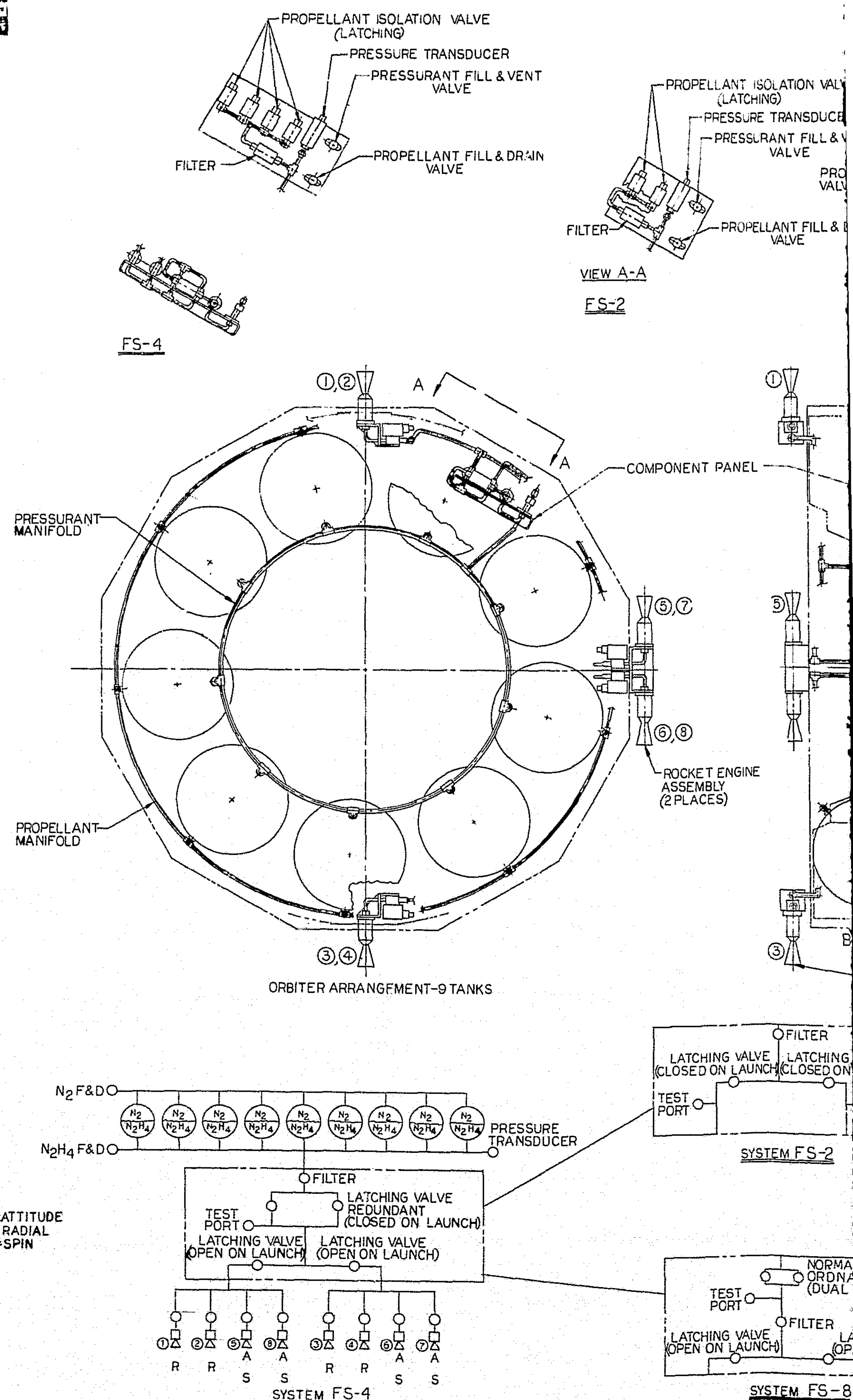

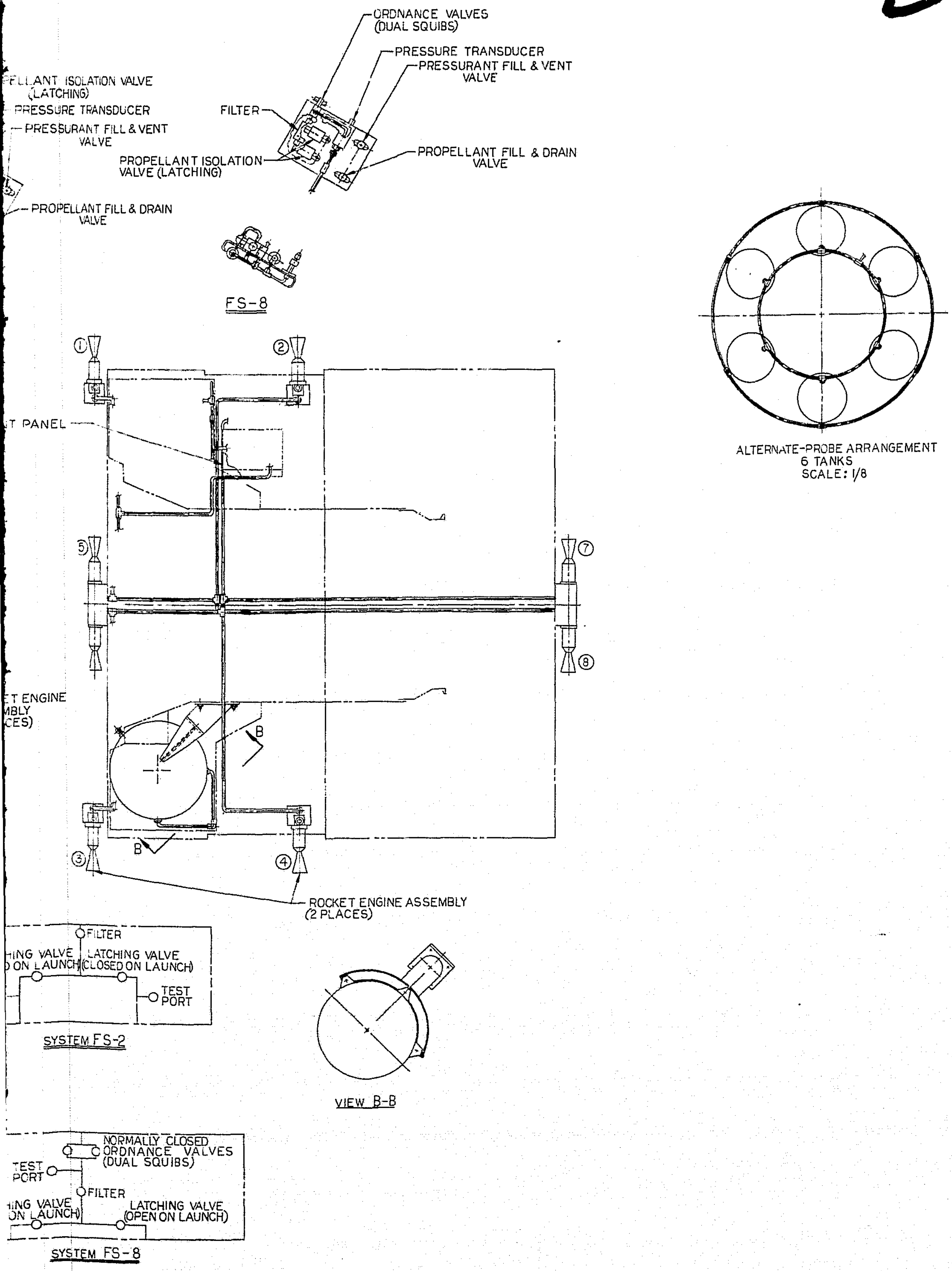

FIGURE 4.2–1E. CANDIDATE PROPULSION SUBSYSTEM P–13

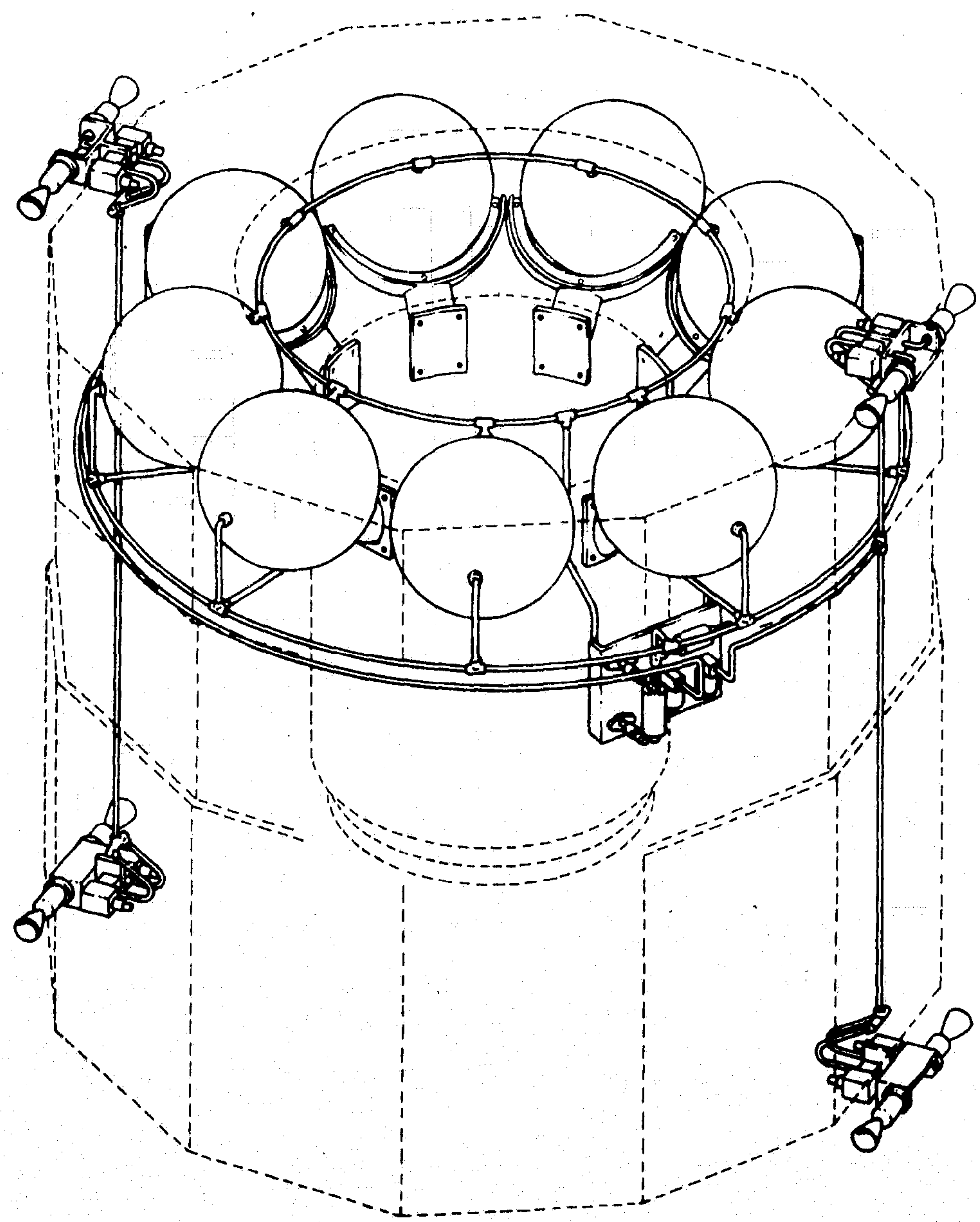

FIGURE 4.2–2. TYPICAL INSTALLATION OF A CANDIDATE PROPULSION SUBSYSTEM INTO THE SPACECRAFT

4.3 Evaluation of Candidate Systems

The following section (4.4) presents a comparative evaluation of the candidate systems. The criteria which were used to evaluate and compare these candidate systems, and the manner in which these criteria were applied, are described below.

The criteria used to evaluate the candidate systems were organized into the following major categories:

- Absolute Criteria
- Quantitative Criteria
- Qualitative Criteria

Initially each candidate had to satisfy an absolute criterion, which is essentially the requirements of GSFC Subsystem Specification No. S-723-P-10. After this was achieved, the remaining candidates were studied both quantitatively and qualitatively to yield visibility into how they compared to each other in satisfying the Planetary Explorer technical and programmatic requirements.

In the quantitative evaluation five parameters were considered, each of which is broken down into end items which are either graphical relationships, finite quantities, or considerations which each system can easily be numerically ranked against.

In the qualitative evaluation three parameters are considered, each of which is broken down into end items which are discussed for each candidate system. Comparative items which are common to each candidate concept (such as component vendor choice, preflight operations, etc.) were not discussed as part of the qualitative evaluation, but are presented as separate studies in Section 5.0 of this report.

Considerations and tasks which were included in each criteria category are listed below. Where one of these items was specified in the GSFC Statement of Work (SOW), the applicable paragraph number of the Statement of Work is referenced.

- ABSOLUTE CRITERIA

  All candidate systems must meet NASA subsystem specification No. S-723-P-10 Rev. "A" as modified by NASA correspondence.

- QUANTITATIVE CRITERIA

  The following parameters which can be compared quantitatively will be evaluated for each candidate system.

4.3 (continued)

A. Reliability

1.	Weight vs reliability	SOW (2.1.1)
2.	FMEA and redundancy analysis	SOW (2.4)
3.	Design complexity	SOW (2.7.3)
4.	Numerical reliability	SOW (2.7.2) (3.6)

B. Weight

1.	Propellant weight vs spin rates	SOW (2.1.2)
2.	Engine performance curves	SOW (2.1.5)
3.	System Weight	SOW (2.7.1)
4.	Mission propellant weights	SOW (3.1)

C. Performance Margin

1.	Accuracy analysis	SOW (2.1.6)
2.	Fuel Dump	SOW (3.8)
3.	Maneuver time margin	
4.	Accuracy margin	
5.	Life margin	

D. Power

1.	Power profile vs spin rate	SOW (2.1.3)
2.	Power conditioning	SOW (2.3)
3.	Power vs time	SOW (3.3)
4.	Peak and ave power (normal and degraded)	

E. Cost

1. Design/analysis cost
2. Development test cost
3. Qual test cost
4. Acceptance test cost
5. Fabrication costs

- QUALITATIVE

The following parameters which cannot be readily compared quantitatively will be discussed in general, and for specific candidate systems, if differences exist between systems.

4.3 (continued)

A. Operations

1.	Pre-flight operations	SOW (2.8) (3.7)
2.	Flight operations	SOW (2.8) (3.7)
3.	Operational complexity	SOW (3.9)
4.	Failure identification	
5.	Safety	

B. Components

1.	Tankage weight vs total impulse	SOW (2.14)
2.	Valve data	SOW (2.5)
3.	Component weight data	SOW (3.1)
4.	Component development status	SOW (3.4)
5.	Component requirements	

C. Structural/Physical Considerations

1.	Thermal analysis	SOW (2.2)
2.	Plume study	SOW (2.6)
3.	Layouts	SOW (2.9)
4.	Thermal requirements	SOW (2.5)
5.	Leakage paths	
6.	Magnetic effects	
7.	Environmental (vibration/shock ACCeleration)	
8.	Feed system dynamics	
9.	Contamination control	
10.	C.G. tolerances	

4.4 Comparison of Candidate Systems

The following sections, 4.4.1 through 4.4.8, present a comparison of the 15 candidate systems against the evaluation criteria outlined in the preceding section (4.3). A summary of the comparison data developed in these sections is shown in Tables 4.4.0-I and 4.4.0-II for the Orbiter and Probe missions, respectively.

The numerical reliability assessment of each of the candidate systems satisfies the GSFC specification requirement. However, the values are so close to each other that it is meaningless to attempt to draw a significant conclusion from these numbers alone. The trends do indicate a slight advantage for candidates using feed system FS-8 (the system with two latching valves and one squib valve), and a more significant disadvantage for engine placement concept P-12 -- the gimballed engine system.

In comparing system weights, the total spread is less than 6 pounds between any two propulsion system candidates. Those candidate systems which utilize feed system concepts FS-2 and FS-8 are at approximately the same weight, and there is about a 2 pound penalty for feed system candidates that utilize concepts FS-4 -- the system with four latching solenoid valves. Systems utilizing engine placement concept P7 (6 engine system) realize a 2 to 3 pound weight advantage over the candidate systems utilizing 8 engines.

All systems for both missions meet the requirements for maneuver rate and resolution. However, candidate systems which utilize engine placement concept P-5 (the system with 8 engines in four two-engine clusters firing alongside the vehicle) have a very significant flight operational advantage. In the performance of ΔV maneuvers, this is the only engine placement concept that provides a one-for-one correspondence between random engine thrust vector angular tolerances and spacecraft thrust vector tolerance. In all other systems, these errors cause a change in vehicle spin rate. The change in vehicle spin rate magnifies the spacecraft thrust vector error. As a result, in order to maintain the spin rate tolerance and minimize the vehicle thrust vector error, all candidate systems using engine placement concepts other than P-5 require breaking the large ΔV maneuvers into many small maneuvers. Tables 4.4.0-I and 4.4.0-II list the maximum number of maneuvers required to perform ΔV maneuvers under the title of "ΔV complexity."

The question may arise as to the effect that engine thrust magnitude changes have on spin rate changes in system P-5. The answer to this question is that the major source of this error is engine-to-engine repeatability and this error is calibrated out during the flight calibration firing.

The relative program costs of the candidate propulsion systems, and their evaluation against the qualitative criteria, are shown in the summary tables using a ranking technique. The lowest numbers are applied to the systems which rank the highest qualitatively, or have the lowest program cost.

4.4 (continued)

Without the benefit of weighting factors which can be applied to the various evaluation criteria, it is difficult to make a propulsion system selection based upon the preceding summary of the comparison study. However, some trends are indicated by the data. First, engine placement concept P-5 offers a significant operational advantage. Second, feed system FS-8 offers a slight reliability advantage with a negligible cost and weight penalty. As a result, Hamilton Standard recommends that candidate system III be selected for the Planetary Explorer application based upon our present understanding of the Planetary Explorer mission requirements. This system consists of engine placement concept P-5 combined with feed system concept FS-8.

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TABLE 4.4.0-I. SUMMARY COMPARISON OF CANDIDATE SYSTEMS - ORBITER MISSION

Characteristic \ Candidate Subsystem	I	II	III	IV	V	VI	VI
		P-5			P-7		
	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8	FS-2
Reliability	.995696	.995697	.995697	.995879	.995880	.995880	.995
System Weight (wet) lbs	134.84	136.27	135.65	131.42	132.84	133.23	135.
Maneuvering							
Rate: Δ V (m/sec^2)	.0181	.0181	.0181	.0091	.0091	.0091	.0091
Δ α (°/min)	50.1	50.1	50.1	50.1	50.1	50.1	50.2
Δ N (rpm/min)	121	121	121	122	122	122	122
Resolution: Δ V (m/sec)	.0049	.0049	.0049	.0018	.0018	.0018	.0030
* Δ α (°)	.133	.133	.133	.134	.134	.134	.133
Δ N (rpm)	.11	.11	.11	.11	.11	.11	.11
ΔV Complexity	3	3	3	29	29	29	29
Power (Watts) Max							
(Without Conditioning)							
Mission Ave.	.001814	.001814	.001814	.001820	.001820	.001820	.001
Peak	44.64	44.64	44.64	33.58	33.58	33.58	44.6
(With Conditioning)							
Mission Ave.	.000925	.000925	.000925	.000928	.000925	.000928	.000
Peak	22.32	22.32	22.32	16.79	16.79	16.79	22.3
Program Cost	2	3	2	1	3	1	3
Relative Rating Against Qualitative Criteria	1	3	1	1	3	2	2

*Resolution for Δα is obtained by firing one engine .025 seconds for orbital mission onl

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I	VII	VIII	IX	X	XI	XII	XIII	XIV	XV
3	FS-2	P-10 FS-4	FS-8	FS-2	P-12 FS-4	FS-8	FS-2	P-13 FS-4	FS-8
5880	.995696	.995696	.995697	.993015	.993016	.993016	.995696	.995697	.995697
.23	135.04	136.46	135.85	133.65	135.07	134.46	133.52	134.94	134.34
l R	.0091 50.2 122 .0030 .133 .11	.0091 50.2 122 .0030 .133 .11	.0091 50.2 122 .0030 .133 .11	.0060 50.3 122 .0020 .133 .11	.0060 50.3 122 .0020 .133 .11	.0060 50.3 122 .0020 .133 .11	.0076 50.2 122 .0025 .133 .11	.0076 50.2 122 .0025 .133 .11	.0076 50.2 122 .0025 .133 .11
	29	29	29	29	29	29	29	29	29
820 8	.001815 44.64	.001815 44.64	.001815 44.64	.001813 22.32	.001813 22.32	.001813 22.32	.001815 22.32	.001815 22.32	.001815 22.32
928 9	.000927 22.32	.000927 22.32	.000927 22.32	.000924 11.16	.000924 11.16	.000924 11.16	.000927 11.16	.000927 11.16	.000927 11.16
	3	5	3	4	6	4	3	5	3
	2	4	3	3	5	4	2	4	3

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TABLE 4.4.0-II. SUMMARY COMPARISON OF CANDIDATE SYSTEMS - PROBE MISSION

Characteristic \ Candidate Subsystem	I	II	III	IV	V	VI	VII
	P-5			P-7			
	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8	FS-2
Reliability	.995696	.995697	.995697	.995879	.995880	.995880	.99569
System Weight (wet) lbs	101.84	103.26	102.67	98.17	99.59	98.98	102.03
Maneuvering							
Rate: Δ V (m/sec^2)	.0094	.0094	.0094	.0050	.0050	.0050	.005
Δ α (°/min)	11.5	11.5	11.5	11.5	11.5	11.5	11.5
Δ N (rpm/min)	46	46	46	46	46	46	46
Resolution: Δ V (m/sec)	.0055	.0055	.0055	.0029	.0029	.0029	.0029
Δ α (°)	.128	.128	.128	.128	.128	.128	.128
Δ N (rpm)	.08	.08	.08	.08	.08	.08	.08
Δ V Complexity	3	3	3	13	13	13	13
Power (Watts) Max							
Without Conditioning							
Mission Ave.	.002810	.002810	.002810	.002673	.002673	.002673	.00281
Peak	44.64	44.64	44.64	33.58	33.58	33.58	44.64
With Conditioning							
Mission Ave.	.001433	.001433	.001433	.001363	.001363	.001363	.00143
Peak	22.32	22.32	22.32	16.79	16.79	16.79	22.32
Program Cost	2	3	2	1	3	1	3
Relative Rating Against Qualitative Criteria	1	3	1	1	3	2	2

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VI	VII	VIII	IX	X	XI	XII	XIII	XIV	XV
		P-10			P-12			P-13	
S-8	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8
995880	.995696	.995696	.995697	.993015	.993016	.993016	.995696	.005697	.995697
8.98	102.03	103.45	102.84	100.60	102.02	101.41	100.48	101.90	101.29
0050 1.5 6 0029 128 08	.005 11.5 46 .0029 .128 .08	.005 11.5 46 .0029 .128 .08	.005 11.5 46 .0029 .128 .08	.004 11.5 46 .0023 .128 .08	.004 11.5 46 .0023 .128 .08	.004 11.5 46 .0023 .128 .08	.007 11.5 46 .0039 .128 .08	.007 11.5 46 .0039 .128 .08	.007 11.5 46 .0039 .128 .08
3	13	13	13	13	13	13	13	13	13
002673 3.58	.002810 44.64	.002810 44.64	.002810 44.64	.002691 22.32	.002691 22.32	.002691 22.32	.002692 22.32	.002692 22.32	.002692 22.32
001363 6.79	.001433 22.32	.001433 22.32	.001433 22.32	.001372 11.16	.001372 11.16	.001372 11.16	.001373 11.16	.001373 11.16	.001373 11.16
	3	5	3	4	6	4	3	5	3
	2	4	3	3	5	4	2	4	3

4.4.1 Reliability:

All of the candidate subsystems were analyzed and each of them meet the reliability requirement, using the failure rates and analytical methods described in paragraphs 5.2.2. and 5.2.4. The several feed systems have the same reliability, out to the sixth significant place after the decimal. The engine placement concepts are very close to each other, to the third significant place. Thus, it is evident that the engine placement is the stronger influence on the subsystem reliability as determined by the referenced methods.

It should be recognized that the precision of the failure rates does not, of itself, justify calculating subsystem reliabilities to six significant places. Each of the calculations applies the same failure rates consistently, so that their value lies in the comparison of the candidate subsystems, regardless of the unknown inaccuracies which may exist in the failure rates.

The quantitative analyses of reliability have not considered any electrical control components or equipment except the electrical portions of those items shown on the failure rate listing in paragraph 5.2.2. It is recommended that all signal sources providing commands for the candidate subsystem be redundant.

The principal sources of potential unreliability are, in order of their importance:

a. Closing and reopening of manifold valves between engine firings

b. The engine gimballing actuation on engine placement concept P12

c. Propellant-line connections of the individual engines to the feed system

d. Engine valves (if it were not for the many redundancies, and the manifold valves, this would be the greatest single influence on subsystem reliability)

Diagnosis and correction of troubles will be facilitated if temperature sensors are provided on each of the engines, to provide a clear indication of whether the engines are responding to command. It is also recommended that the manifold valves be kept closed except for the periods in which thrust will be required. This will minimize navigational error which would result if an engine operated inadvertently for the time of communication to-and-from a ground station. It may also enable normal operations with an engine valve having a minor internal leak failure.

The numerical reliability assessments for each of the candidate systems are shown in Table 4.4.1-I. As the listed values indicate, there is no significant numerical difference between the candidates, but feed system concept FS-8 indicates a preferential trend.

A summary comparison of candidate system launch weight and numerical reliability assessments is presented in Table 4.4.1-II.

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TABLE 4.4.1-I. NUMERICAL RELIABILITY ASSESSMENT FOR CANDIDATE SUBSYSTEMS

Candidate Subsystem	Engine Arrangement	Feed System Arrangement	Engine Reliability	Feed System Reliability	Subsystem Reliability	Reliab Rank
I	P5	FS-2	.995 784	.999 912	.995 696	8
II	P5	FS-4	.995 784	.999 912	.995 697	6
III	P5	FS-8	.995 784	.999 912	.995 697	4
IV	P7	FS-2	.995 967	.999 912	.995 888	3
V	P7	FS-4	.995 967	.999 912	.995 888	2
VI	P7	FS-8	.995 967	.999 912	.995 880	1
VII	P10	FS-2	.995 783	.999 912	.995 696	9
VIII	P10	FS-4	.995 783	.999 912	.995 696	7
IX	P10	FS-8	.995 783	.999 912	.995 697	5
X	P12	FS-2	.993 103	.999 912	.993 015	12
XI	P12	FS-4	.993 103	.999 912	.993 016	11
XII	P12	FS-8	.993 103	.999 912	.993 016	10
XIII	P13	FS-2	.995 784	.999 912	.995 696	8
XIV	P13	FS-4	.995 784	.999 912	.995 697	6
XV	P13	FS-8	.995 784	.999 912	.995 697	4

UBSYSTEMS

em ity	Reliability Ranking
[illegible]	8
7	6
7	4
8	3
8	2
0	1
[illegible]	9
[illegible]	7
7	5
[illegible]	12
[illegible]	11
[illegible]	10
[illegible]	8
[illegible]	6
[illegible]	4

TABLE 4.4.1-II. WEIGHT VS RELIABILITY SUMMARY - PLANETARY ORBITER

Candidate	Total Weight (Lbs)	Estimated Reliability
I	134.04	.99[illegible] [illegible]9[illegible]
II	136.27	.995 697
III	134.43	.995 697
IV	131.42	.995 879
V	133.64	.995 880
VI	131.81	.995 880
VII	135.04	.995 696
VIII	137.26	.995 696
IX	135.43	.995 697
X	133.65	.993 015
XI	135.87	.993 016
XII	134.04	.993 016
XIII	133.52	.995 696
XIV	135.74	.995 697
XV	133.92	.995 697

4.4.2 Weight

A weight summary for the Planetary Explorer Orbiter and Probe candidate propulsion subsystems is presented in Table 4.4.2-I. This table summarizes the dry weight, propellant and pressurant weight, and the total charged weight for the candidate subsystems. A breakdown of the dry weight, and of the propellant weight allocation by maneuver, is presented in Tables 4.4.2-II and 4.4.2-III for the Orbiter and Probe missions, respectively. The candidate subsystems are very similar with the significant differences being in the number and physical location of engines for each subsystem, and in the number and type of isolation valves in the upstream feed system. The number of engines obviously affects the subsystem weight, but the location of the engines on the spacecraft also influences the weight because of the following factors:

- Engine mounting structure
- Propellant feed tube length
- Electrical wiring harness and connectors

The upstream feed system component arrangement selection influences the subsystem weight for the following reasons:

- Number of isolation valves, pressure transducers and test ports
- Type of isolation valves - ordnance vs latching solenoids
- Component panel size - dependent upon component arrangement

The weight variation between all candidate subsystems and between subsystems with the same number of engines and similar feed systems is presented in Tables 4.4.2-IV thruough 4.4.2- VI.

Comparison of subsystems with the same number of engines and similar feed systems as listed in Table 4.4.2-VI shows weight variations of approximately 1.1% and 1.5% for the eight engine concepts and 1.7% and 2.5% for the six engine concepts, for the Orbiter and Probe, respectively. The weight variation influence in these cases is in the propellant tubing length, mounting brackets and electrical wiring harness. The two six engine candidate subsystems have more significant differences because of the gimbal actuators required for the one system.

The variation in the propellant required for performing the spacecraft maneuvers between the various candidate subsystems is insignificant, and is less than 0.4% of the launch weight of the systems. In other words, the mission average specific impulse for the candidate systems is approximately the same for all candidates.

TABLE 4.4.2-I. WEIGHT SUMMARY - ORBITER AND PROBE

Candidate Subsystem	ORBITER			PROBE		
	Dry Wt (lbs)	N_2H_4 & N_2(1.46 lbs) Wt. (lbs)	Total Wt(lbs)	Dry	N_2H_4 + N_2(.88 lbs) Wt. (lbs)	Total Wt (lbs)
I (P-5/FS-2)	44.3	90.54	134.84	37.3	64.54	101.84
II (P-5/FS-4)	45.72	90.54	136.27	38.72	64.54	103.26
III (P-5/FS-8)	45.11	90.54	135.65	38.13	64.54	102.67
IV (P-7/FS-2)	40.42	91.0	131.42	33.42	64.75	98.17
V (P-7/FS-4)	41.84	91.0	132.89	34.84	64.75	99.59
VI (P-7/FS-8)	41.23	91.0	132.23	34.23	64.75	98.98
VII (P-10/FS-2)	44.35	90.69	135.04	37.35	64.68	102.03
VIII (P-10/FS-4)	45.77	90.69	136.46	38.77	64.68	103.45
IX (P-10/FS-8)	45.16	90.69	135.85	38.16	64.68	102.84
X (P-12M/FS-2)	43.07	90.58	1336.5	36.07	64.53	100.6
XI (P-12M/FS-4)	44.49	90.58	135.07	37.49	64.53	102.02
XII (P-12M/FS-8)	44.88	90.58	134.46	36.88	64.53	101.41
XIII (P-13/FS-2)	43.0	90.52	133.52	36.0	64.48	100.48
XIV (P-13/FS-4)	44.42	90.52	134.94	37.42	64.48	101.90
XV (P-13/FS-8)	43.81	90.52	134.34	36.81	64.48	101.29

4.4.2 (continued)

Propulsion subsystem balancing weight has not been included in the weight analysis. Factors which have the greatest impact on subsystem unbalance are the unsymmetrical location of the engines on some of the candidate systems, and the modularized component panel located near the vehicle skin panels. The "dead weight" required to compensate for this unbalance would be prohibitive, approximately 5 and 10 lbs for engine placement concepts P-12 and P-13, respectively; and the unbalance in these cases could be better accomplished by judiciously locating the spacecraft packages to offset the unbalance. Unbalance caused by the tank location and weight tolerances is discussed in Section 5.12 (C.G. Tolerances).

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TABLE 4.4.2-II WEIGHT

COMPONENT	UNIT WT.	SYSTEMS I, II & III (P-5) FEED SYSTEM FS-2 QTY	WT	FS-4 QTY	WT	FS-8 QTY	WT	SYSTEMS IV, V & VI (P-7) FEED SYSTEM FS-2 QTY	WT	FS-4 QTY	WT	FS-8 QTY	WT
TANK (9.83 ID)	1.6	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4
ROCKET ENGINE ASS'Y	.88	8	7.04	8	7.04	8	7.04	6	5.28	6	5.28	6	5.28
FILL & DRAIN VALVE	.28	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56
FILTER	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42
PRESSURE TRANSDUCER	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32
ISOLATION VALVE-SQUIB	.42	—	—	—	—	2	.84	—	—	—	—	2	.84
ISOLATION VALVE-LATCH.	.6	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2
REA STRUCTURE - 2 REA	.75	4	3.0	4	3.0	4	3.0	2	1.5	2	1.5	2	1.5
- 1 REA	.40	—	—	—	—	—	—	2	.80	2	.80	2	.80
- 1 REA	.70	—	—	—	—	—	—	—	—	—	—	—	—
TUBING & FITTINGS	AR	—	2.1	—	2.1	—	2.1	—	2.6	—	2.6	—	2.6
MISC. BKTS, SCREWS, ETC.	AR	—	1.2	—	1.2	—	1.2	—	1.2	—	1.2	—	1.2
COMPONENT PANEL	AR	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50
TANK MT'G STRUTS	.70	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3
WIRING HARNESS	2 OZ./CONN .7 LB. WIRE	8	1.7	10	1.95	10	1.95	8	1.7	10	1.95	10	1.95
TEMP. SENSORS	.06	10	.6	10	.6	10	.6	8	.48	8	.48	8	.48
REA THERMAL - 2 REA	1.10	4	4.4	4	4.4	4	2.2	2	2.2	2	2.2	2	2.2
INSUL./SHIELD - 1 REA	.35	—	—	—	—	—	—	—	—	—	—	—	—
- 1 REA	.20	—	—	—	—	—	—	2	.4	2	.4	2	.4
TEST PORT VALVE	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28
ACTUATOR-SWIVEL	1.05	—	—	—	—	—	—	—	—	—	—	—	—
BALANCING WT.	—	—	—	—	—	—	—	—	—	—	—	—	—
TOTAL DRY WT.			44.3		45.72		45.11		40.42		41.84		41.23
PROPELLANT													
MIDCOURSE CORRECTION			41.23		41.23		41.23		41.22		41.22		41.22
ATTITUDE CONTROL			1.99		1.99		1.99		2.06		2.06		2.06
SPIN CONTROL			1.52		1.52		1.52		1.53		1.53		1.53
ORBIT MANEUVER			42.86		42.86		42.86		43.12		43.12		43.12
RESIDUALS - LINES			.73		.73		.73		.86		.86		.86
LOADING TOL. (±.75%)			.66		.66		.66		.66		.66		.66
EXPULSION EFF. (99.9%)			.09		.09		.09		.09		.09		.09
PROPELLANT (N_2H_4) TOT.			89.08		89.08		89.08		89.54		89.54		89.54
PRESSURANT (N_2)			1.46		1.46		1.46		1.46		1.46		1.46
TOTAL CHARGED WT.			134.84		136.27		135.65		131.42		132.84		132.23

/EIGHT SUMMARY - ORBITER

	& VI (P-7)		SYSTEMS VII, VIII & IX (P-10)						SYSTEMS X, XI & XII (P-12M)						SYSTEMS XIII, XIV & XV (P-13)					
	TEM		FEED SYSTEM						FEED SYSTEM						FEED SYSTEM					
	FS-8		FS-2		FS-4		FS-8		FS-2		FS-4		FS-8		FS-2		FS-4		FS-8	
	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT
1	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4	9	14.4
8	6	5.28	8	7.04	8	7.04	8	7.04	6	5.28	6	5.28	6	5.28	8	7.04	8	7.04	8	7.04
6	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56
2	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42
2	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32
	2	.84	–	—	–	—	2	.84	–	—	–	—	2	.84	–	—	–	—	2	.84
4	2	1.2	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2
	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5
0	2	.80	4	1.6	4	1.6	4	1.6	–	—	–	—	–	—	4	1.6	4	1.6	4	1.6
	–	—	–	—	–	—	–	—	2	1.4	2	1.4	2	1.4	–	—	–	—	–	—
	–	2.6	–	2.6	–	2.6	–	2.6	–	2.3	–	2.3	–	2.3	–	2.1	–	2.1	–	2.1
	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2
	1	.50	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50
	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3	9	6.3
5	10	1.95	10	1.95	12	2.2	12	2.2	10	1.95	12	2.2	12	2.2	8	1.7	10	1.95	10	1.95
3	8	.48	10	.6	10	.6	10	.6	8	.48	8	.48	8	.48	10	.6	10	.6	10	.6
	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2
	–	—	4	1.4	4	1.4	4	1.4	–	—	–	—	–	—	–	—	–	—	–	—
	2	.4	–	—	–	—	–	—	2	.4	2	.4	2	.4	4	.8	4	.8	4	.8
	1	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28
	–	—	–	—	–	—	–	—	2	2.1	2	2.1	2	2.1	–	—	–	—	–	—
	–	—	–	—	–	—	–	—	–	—	–	—	–	—	–	—	–	—	–	—
		41.23		44.35		45.77		45.16		43.07		44.49		44.88		43.0		44.42		43.81
		41.22		41.23		41.23		41.23		41.21		41.21		41.21		41.21		41.21		41.21
		2.06		1.99		1.99		1.99		1.99		1.99		1.99		1.98		1.98		1.98
		1.53		1.52		1.52		1.52		1.52		1.52		1.52		1.53		1.53		1.53
		43.12		42.86		42.86		42.86		42.86		42.86		42.86		42.86		42.86		42.86
		.86		.88		.88		.88		.79		.79		.79		.73		.73		.73
		.66		.66		.66		.66		.66		.66		.66		.66		.66		.66
		.09		.09		.09		.09		.09		.09		.09		.09		.09		.09
		89.54		89.23		89.23		89.23		89.12		89.12		89.12		89.06		89.06		89.06
		1.46		1.46		1.46		1.46		1.46		1.46		1.46		1.46		1.46		1.46
		132.23		135.04		136.46		135.85		133.65		135.07		134.46		135.52		134.94		134.34

FOLDOUT FRAME

TABLE 4.4.2-III WEIGHT SUMM

		Systems I, II & III (P-5)						Systems IV, V & VI (P-7)						Sys
		Feed System						Feed System						
		FS-2		FS-4		FS-8		FS-2		FS-4		FS-8		FS-
	Unit Wt	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y	WT	QT'Y
Tank (9.83 ID)	1.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6
Rocket Engine Ass'y	.88	8	7.04	8	7.04	8	7.04	6	5.28	6	5.28	6	5.28	8
Fill & Drain Valve	.28	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2
Filter	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1
Pressure Transducer	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1
Isolation Valve - Squib	.42	–	–	–	–	2	.84	–	–	–	–	2	.84	–
Isolation Valve - Latch	.6	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2	2
REA Structure - 2REA	.75	4	3.0	4	3.0	4	3.0	2	1.5	2	1.5	2	1.5	2
-1REA	.40	–	–	–	–	–	–	2	.80	2	.80	2	.80	4
-1REA	.70	–	–	–	–	–	–	–	–	–	–	–	–	–
Tubing & Fittings	AR	–	2.0	–	2.0	–	2.0	–	2.5	–	2.5	–	2.5	–
Misc. Brkts, Screws, Etc.	AR	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–	1.2	–
Component Panel	AR	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50	1
Tank Mt'g Struts	.70	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6
Wiring Harness	2 oz/conn .7 lb wire	8	1.7	10	1.95	10	1.95	8	1.7	10	1.95	10	1.95	10
Temp Sensors	.06	10	.6	10	.6	10	.6	8	.48	8	.48	8	.48	10
REA Thermal - 2REA	1.10	4	4.4	4	4.4	4	4.4	2	2.2	2	2.2	2	2.2	2
Insul./Shield - 1REA	.35	–	–	–	–	–	–	–	–	–	–	–	–	4
1REA	.20	–	–	–	–	–	–	2	.4	2	.4	2	.4	–
Test Port Valve	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28	2
Actuator - Swivel	1.05	–	–	–	–	–	–	–	–	–	–	–	–	–
Balancing Wt.	–	–	–	–	–	–	–	–	–	–	–	–	–	–
Total Dry Wt.			37.3		38.72		38.13		33.42		34.84		34.23	
Propellant														
Midcourse Correction			49.38		49.38		49.38		49.37		49.37		49.37	
Attitude Control			6.82		6.82		6.82		6.94		6.94		6.94	
Spin Control			3.44		3.44		3.44		3.43		3.43		3.43	
Orbit Maneuver			2.79		2.79		2.79		2.78		2.78		2.78	
Residuals - Lines			.70		.70		.70		.82		.82		.82	
- Loading Tol ($\pm$.75%)			.47		.47		.47		.47		.47		.47	
- Expulsion Eff. (99.9%)			.06		.06		.06		.06		.06		.06	
Propellant (N_2H_4) Tot.			63.66		63.66		63.66		63.87		63.87		63.87	
Pressurant (N_2)			.88		.88		.88		.88		.88		.88	
Total Charged Wt.			101.84		103.26		102.67		98.17		99.57		98.98	

GHT SUMMARY - PROBE

(P-7)		SYSTEMS VII, VIII & IX (P-10)						SYSTEMS X, XI & XII (P-12M)						SYSTEMS XIII, XIV & XV (P-13)					
		FEED SYSTEM						FEED SYSTEM						FEED SYSTEM					
FS-8		FS-2		FS-4		FS-8		FS-2		FS-4		FS-8		FS-2		FS-4		FS-8	
QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT	QTY	WT
6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6	6	9.6
6	5.28	8	7.04	8	7.04	8	7.04	6	5.28	6	5.28	6	5.28	8	7.04	8	7.04	8	7.04
2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56	2	.56
1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42	1	.42
1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32	1	.32
2	.84	-	—	-	—	2	.84	-	—	-	—	2	.84	-	—	-	—	2	.84
2	1.2	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2	2	1.2	4	2.4	2	1.2
2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5	2	1.5
2	.80	4	1.6	4	1.6	4	1.6	-	—	-	—	-	—	4	1.6	4	1.6	4	1.6
-	—	-	—	-	—	-	—	2	1.4	2	1.4	2	1.4	-	—	-	—	-	—
-	2.5	-	2.5	-	2.5	-	2.5	-	2.2	-	2.2	-	2.2	-	2.0	-	2.0	-	2.0
-	1.2	-	1.2	-	1.2	-	1.2	-	1.2	-	1.2	-	1.2	-	1.2	-	1.2	-	1.2
-	.50	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50	1	.50	1	.75	1	.50
6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2	6	4.2
10	1.95	10	1.95	12	2.2	12	2.2	10	1.95	12	2.2	12	2.2	8	1.7	10	1.95	10	1.95
8	.48	10	.6	10	.6	10	.6	8	.48	8	.48	8	.48	10	.6	10	.6	10	.6
2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2	2	2.2
-	—	4	1.4	4	1.4	4	1.4	-	—	-	—	-	—	-	—	-	—	-	—
2	.4	-	—	-	—	-	—	2	.4	2	.4	2	.4	4	.8	4	.8	4	.8
	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28	2	.56	1	.28	1	.28
-	—	-	—	-	—	-	—	2	2.1	2	2.1	2	2.1	-	—	-	—	-	—
-	—	-	—	-	—	-	—	-	—	-	—	-	—	-	—	-	—	-	—
	34.23		37.35		38.77		38.16		36.07		37.49		36.88		360		37.42		36.81
	49.37		49.38		49.38		49.38		49.37		49.37		49.37		49.37		49.37		49.37
	6.94		6.82		6.82		6.82		6.79		6.79		6.79		6.79		6.79		6.79
	3.43		3.44		3.44		3.44		3.44		3.44		3.44		3.44		3.44		3.44
	2.78		2.79		2.79		2.79		2.77		2.77		2.77		2.77		2.77		2.77
	.82		.84		.84		.84		.75		.75		.75		.70		.70		.70
	.47		.47		.47		.47		.47		.47		.47		.47		.47		.47
	.06		.06		.06		.06		.06		.06		.06		.06		.06		.06
	63.87		63.80		63.80		63.80		63.65		63.65		63.65		63.60		63.60		63.60
	.88		.88		.88		.88		.88		.88		.88		.88		.88		.88
	98.98		102.03		103.45		102.84		100.6		102.02		101.41		100.48		101.9		101.29

TABLE 4.4.2-IV

MAX. WEIGHT VARIATION BETWEEN SUBSYSTEMS

Candidate Subsystem	Subsystem Charged Weight Orbiter	Probe
VIII (P-10/FS-4)	136.46	103.45
IV (P-7/FS-2)	131.42	98.17
Weight Variation	5.04	5.28

TABLE 4.4.2-V
MAX. WEIGHT VARIATION BETWEEN SUBSYSTEMS
WITH SAME NUMBER OF ENGINES

Candidate Subsystem	Subsystem Charged Wt Orbiter 8 REA's	Orbiter 6 REA's	Probe 8 REA's	Probe 6 REA's
VIII (P-10/FS-4)	136.46		103.45	
XIII (P-13/FS-2)	133.52		100.48	
XI (P-12M/FS-4)		135.07		102.2
IV (P-7/FS-2)		131.42		98.17
Weight Variation	2.94	3.65	2.97	3.85

TABLE 4.4.2-VI

MAX. WEIGHT VARIATION BETWEEN SUBSYSTEMS

OF SIMILAR FEED SYSTEMS AND SAME NUMBER OF ENGINES

Candidate Subsystem	Subsystem Charged Wt Orbiter 8 REA's	Orbiter 6 REA's	Probe 8 REA's	Probe 6 REA's
VII (P-10/FS-2)	135.04		102.03	
XIII (P-13/FS-2)	133.52		100.48	
X (P-12M/FS-2)		133.65		100.6
IV (P-7/FS-2)		131.42		98.17
Weight Variation	1.52	2.23	1.55	2.43

4.4.3 Performance Margin

The performance margin for the candidate propulsion systems has been considered from the following two aspects which are discussed in detail in the subsequent sections.

a. The margin available for performing all required maneuvers within rates, accuracy and resolution required.

b. The operating life margin available in the Rocket Engine Assemblies (REA's) selected for each candidate system. Operating life margin is the difference between the operating life required for the Planetary Explorer Orbiter and Probe missions, and the operating life capability which has previously been demonstrated.

4.4.3(a) MANEUVERING RATES, RESOLUTION AND ACCURACY:

A performance analysis of the five candidate systems has been conducted for the purposes of comparing the relative suitability of each in performing maneuvers required for the Planetary Explorer Probe and Orbiter Missions. The results of this study indicate that all candidate systems are capable of meeting the maneuvering rate, accuracy and resolution requirements defined in the subsystem specification (S-723-0-10 Rev. A). There are, however, significant differences in performance margin and operational complexity which allow the candidate systems to be ranked. Performance parameters typical of the Hamilton Standard IDCSP/A 5 lb thrust engine were used as a basis for the study.

4.4.3(a)1 Maneuvering Rate

Minimum maneuvering rates for the candidate subsystems were calculated and are presented in Table 4.4.3(a)-I (Orbiter) and Table 4.4.3(a)-II (Probe) along with the mission requirements and the mission event at which the minimum rate occurs.

With regard to the rate of velocity change, the distinguishing factor between candidates is the number of ΔV engines firing; hence systems with engine placement configurations P5 and P10 (four ΔV engines) exhibit the highest ΔV maneuvering rates, while systems with engine placement configuration P12 (one ΔV engine) exhibit the lowest rate for both Probe and Orbiter missions. All of the ΔV rates for both Probe and Orbiter missions are within specified limits.

The precession ($\Delta \alpha$) maneuver rates are equal for four of the five candidates, with candidate systems utilizing engine placement configuration P7 having a lower rate of change due to the smaller moment arm between engines.

TABLE 4.4.3.1-I. MANEUVERING RATE MARGIN - ORBITER MISSION

Candidate Systems	Maneuver	Minimum Rate	Mission Event	Specification Requirement
I, II & III (P-5)	ΔV Δa ΔN	1.81×10^{-2} m/sec^2 50.1°/min 121 rpm/min	21, 2 m/sec V 28, 6° Precession 13.14 ± 2.5 rpm	3.74×10^{-3} m/sec^2 * 3°/min 20 rpm/min
IV, V & VI (P-7)	ΔV Δa ΔN	$.91 \times 10^{-2}$ m/sec^2 50.1°/min 121 rpm/min	21 28 13, 14	3.74×10^{-3} m/sec^2 3°/min 20 rpm/min
VII, VIII & IX (P-10)	ΔV Δa ΔN	$.91 \times 10^{-2}$ m/sec^2 50.2 °/min 122 rpm/min	21 28 13, 14	3.74×10^{3} m/sec^2 3°/min 20 rpm/min
X, XI & XII (P-12)	ΔV Δa ΔN	$.60 \times 10^{-2}$ m/sec^2 50.3 °/min 122 rpm/min	21 28 13, 14	3.74×10^{-3} m/sec^2 3°/min 20 rpm/min
XIII, XIV & XV (P-13)	ΔV Δa ΔN	$.76 \times 10^{-2}$ m/sec^2 50.2 °/min 122 rpm/min	21 28 13, 14	3.74×10^{-3} m/sec^2 3°/min 20 rpm/min

TABLE 4.4.3.1-II MANEUVERING RATE MARGIN - PROBE MISSION

Candidate System	Maneuver	Minimum Rate	Mission Event	Specification Requirement
I, II & III (P-5)	ΔV $\Delta \alpha$ ΔN	9.9×10^{-3} m/sec^2 11.5 °/min 46 rpm/min	15, 2.0 m/sec V 28, Final Precession (85 rpm) 23, spin up mini-probes	3.74×10^{-3} m/sec^2 (midcourse) 5.74×10^{-3} m/sec^2 bus retarget 3°/min 20 rpm/min
IV, V & VI (P-7)	ΔV $\Delta \alpha$ ΔN	5×10^{-3} m/sec^2 11.5 °/min 46 rpm/min	15 28 23	3.74×10-3 m/sec 5.74×10^{-3} m/sec 3°/min 20 rpm/min
VII, VIII & IX (P-10)	ΔV $\Delta \alpha$ ΔN	$\times 10^{-3}$ m/sec^2 °/min rpm/min	15 28 23	3.74×10^{-3} m/sec^2 5.74×10^{-3} m/sec^2 3°/min 20 rpm/min
VII, VIII & IX (P-10)	ΔV $\Delta \alpha$ ΔN	5.0×10^{-3} m/sec^2 11.5 °/min 46 rpm/min	15 28 23	3.74×10^{-3} m/sec^2 5.74×10^{-3} m/sec^2 3°/min 20 rpm/min
X, XI & XII (P-12)	ΔV $\Delta \alpha$ ΔN	4.0×10^{-3} m/sec^2 11.5 °/min 46 rpm/min	15 26, Bus retarget 28 23	3.74×10^{-3} m/sec^2 5.74×10^{-3} m/sec^2 3°/min 20 rpm/min
XIII, XIV & XV (P-13)	ΔV $\Delta \alpha$ ΔN	7×10^{-3} m/sec^2 11.5 °/min 46 rpm/min	15 28 23	3.74×10^{-3} m/sec^2 5.74×10^{-3} m/sec^2 3°/min 20 rpm/min

4.4.3(a)1 (continued)

There is no difference in spin speed rate of change among the candidate systems due to their geometrical similarity (all have two active spin engines and virtually identical moment arms).

In general, the maneuvering rates for the Probe mission are characteristically slower than those for the Orbiter due to the greater mass, moment of inertia, and lower propellant tank pressure (thrust) which exist over most of the mission.

4.4.3(a)2 Resolution

Orbiter:

Worst case resolution values, corresponding mission events, and specification requirements for the Orbiter mission are presented in Table 4.4.3(a)-III for the nominal and single engine firing cases. Nominal velocity resolution values are below the specified maximum 0.1 m/sec for all candidates. As would be expected, systems with engine configurations such as P5 with four ΔV engines exhibit coarser resolutions than those with engine configuration P12 (one ΔV engine), and there is also no difference between the resolutions obtainable with the various candidates when only one engine is fired.

In order to satisfy the $\Delta\alpha$ resolution requirement of .2°, it is necessary to utilize pulse widths of shorter duration than the nominal .050 sec, and to fire the $\Delta\alpha$ engines singly for all Orbiter missions. The values quoted in Table 4.4.3(a)-III or for $\Delta\alpha$ resolution are based on a pulse width of .025 sec and are all within the specified maximum. The ΔV produced by firing a single $\Delta\alpha$ thrustor for .025 sec will be negligible (.0020 m/sec).

There are no differences in spin rate resolution among the candidate systems and the nominal resolution (.11 rpm) is well below the specification value of .25 rpm.

Probe:

Maneuver resolution values for the Probe mission are given in Table 4.4.3(a)-IV. The resolution characteristics of the Probe candidate systems are inter-related in the same manner as those for the Orbiter Mission except that the greater mass and moment of inertia of the Probe vehicle result in more favorable (smaller) resolution values. Single engine firings are not required to achieve the specified .2° precession resolution.

TABLE 4.4.3.1-III. MANEUVERING RESOLUTION - ORBITER MISSION

Candidate Systems	Maneuver	Worst Case Resolution Nominal	Worst Case Resolution Single Engine	Mission Event	Specification Requirement
I, II & III (P-5)	ΔV	.0050 m/sec	.0025 m/sec	32, Periapsis reduction	.1 m/sec
*	$\Delta \alpha$	.266°	.133°	9, Orient for cruise	.2°
	ΔN	.11 rpm	.055 rpm	1, initial spin control	.25 rpm
IV, V & VI (P-7)	ΔV	.0018 m/sec	.0009 m/sec	32	.1 m/sec
	$\Delta \alpha$	.267°	.134°	9	.2°
	ΔN	.11 rpm	.055 rpm	1	.25 rpm
VII, VIII & IX (P-10)	ΔV	.0030 m/sec	.0015 m/sec	32	.1 m/sec
*	$\Delta \alpha$	.266°	.133°	9	.2°
	ΔN	.11 rpm	.055 rpm	1	.25 rpm
X, XI & XII (P-12)	ΔV	.0080 m/sec	.0030 m/sec	32	.1 m/sec
*	$\Delta \alpha$	.266°	.133°	9	.2°
	ΔN	.11 rpm	.055 rpm	1	.25 rpm
XIII, XIV & XV (P-13)	ΔV	.0076	.0038 m/sec	32	.1 m/sec
*	$\Delta \alpha$	.266°	.133°	9	.2°
	ΔN	.11 rpm	.055 rpm	1	.25 rpm

NOTE: UNLESS OTHERWISE SPECIFIED, VALUES ABOVE ARE BASED ON .050 SEC PULSE WIDTH

* .025 sec pulse width

TABLE 4.4.3.1-IV MANEUVERING RESOLUTION - PROBE MISSION

Candidate Systems	Maneuver	Worst Case Resolution		Mission Event	Specification Requirement
		Nominal	Single Engine		
I, II & III (P-5)	ΔV Δa ΔN	.0029 m/sec .256° .08 rpm	.0015 m/sec .128° .04 rpm	26, Retarget bus 7, Orient for cruise 1, initial spin control	.1 m/sec .2° .25 rpm
IV, V & VI (P-7)	ΔV Δa ΔN	.0029 m/sec .128° .08 rpm	.0015 m/sec .064° .04 rpm	26 7 1	.1 m/sec .2° .25 rpm
VII, VIII & IX (P-10)	ΔV Δa ΔN	.0029 m/sec .128° .08 rpm	.0015 m/sec .064° .04 rpm	26 7 1	.1 m/sec .2° .25 rpm
X, XI & XII (P-12)	ΔV Δa ΔN	.0023 m/sec .128° .08 rpm	.0012 m/sec .064° .04 rpm	26 7 1	.1 m/sec .2° .25 rpm
XIII, XIV & XV (P-13)	ΔV Δa ΔN	.0039 m/sec .128° .08 rpm	.0020 m/sec .064° .04 rpm	26 7 1	.1 m/sec .2° .25 rpm

NOTE: .050 second pulse width used to generate this table.

4.4.3(a)2 (Continued)

Velocity and spin rate resolution valves for all candidate systems are well within the specification minumums for nominal operation.

4.4.3(a)3 Maneuvering Accuracy

Candidate system accuracies for Probe and Orbiter missions are shown in Tables 4.4.3(a)-V and 4.4.3(a)-VI. These accuracy values were derived from the mission operations analysis (Section 5.4) and the maneuver error analysis (Section 4.4.3(a)4) and are, therefore, representative of the maneuver accuracy likely to occur during a real mission rather than the ultimate values obtainable in a given candidate system without regard to the attendant complexity of the maneuver. The specification requirements for accuracy were used as constraints in the Flight Operations Study, with the result that all of the accuracy values in Tables 4.4.3(a)-V and 4.4.3(a)-VI are within specification limits. The relative merits of the candidate system, therefore, must be based on the complexity (number of steps necessary to accomplish) required in execution of the maneuver within the accuracy constraints. For instance, in executing the first mid-course correction (Event #11, Orbiter Mission), the accuracy associated with candidate systems with engine configuration P5 appears to be inferior to that available with other candidate systems. It should be noted, however, that for configuration P5, only three increments are required to execute the maneuver, whereas 29 steps are required for the other systems in order to keep the large spin disturbance error common to all systems except P5 within specification limits (±.3 rpm). The extra maneuver increments required for candidate systems with engine placement configuration P7, P10, P12 and P13 are responsible for their superior accuracy in this maneuver, but these systems are obviously inferior to those using P5 on the basis of maneuver complexity. A general comparison of the accuracies and disturbance errors is presented in the error matrix, Table 4.4.3(a)-IX. The systems are ranked on a performance basis in the Flight Operations Analysis, Section 4.4.6, Tables 4.4.6-I, 4.4.6-II and 4.4.6-III.

TABLE 4.4.3.1-VI MANEUVERING ACCURACY - PROBE MISSION

Candidate System	Maneuver	Mission Worst Case Accuracy	Number of Maneuver Increments	Mission Event	Spec Value	Remarks
I, II & III (P-5)	ΔV: $V > 2$ m/sec	.44%	3	9	2 1/2%	Spin control needed
	ΔV: $V \leq 2$ m/sec	3%	2	15	5%	
	$\Delta\alpha$	.11°	5	7	± .20°	Spin control needed
	ΔN	.049 rpm	3	23	± .10 rpm	
IV, V & VI (P-7)	ΔV: $V > 2$ m/sec	.236%	13	9	2 1/2%	Spin control needed
	ΔV: $V \leq 2$ m/sec	2.65%	2	15	5%	
	$\Delta\alpha$	.066°	3	7	± .20°	
	ΔN	.049 rpm	3	23	± .10 rpm	
VII, VIII & IX (P-10)	ΔV: $V > 2$ m/sec	.202%	13	9	2 1/2%	Spin control needed
	ΔV: $V \leq 2$ m/sec	2.3%	2	15	5%	
	$\Delta\alpha$	.096°	2	7	± .20°	
	ΔN	.049 rpm	3	23	± .10 rpm	
X, XI & XII (P-12)	ΔV: $V > 2$ m/sec	.174%	13	9	2 1/2%	Spin control needed
	ΔV: $V \leq 2$ m/sec	3.62%	1	15	5%	
	$\Delta\alpha$	.11°	5	7	± .20°	Spin control needed
	ΔN	.049 rpm	3	23	± .10 rpm	
XIII, XIV & XV (P-13)	ΔV: $V > 2$ m/sec	.202%	13	9	2 1/2%	Spin control needed
	ΔV: $V \leq 2$ m/sec	2.3%	2	15	5%	
	$\Delta\alpha$	.11°	5	7	± .20°	Spin control needed
	ΔN	.049 rpm	3	23	± .10 rpm	

TABLE 4.4.3.1-V. MANEUVERING ACCURACY - ORBITER MISSION

Candidate Systems	Maneuver	Mission Worst Case Accuracy	Number of Maneuver Increments	Mission Event	Spec Value	Remarks
I, II & III (P-5)	ΔV, $V > 2$ m/sec	1.3%	3	11	2 1/2%	Spin control needed
	ΔV, $V \leq 2$ m/sec	3%	2	21	5%	
	$\Delta \alpha$	.17°	2	31	± .20°	One engine, spin control needed
	ΔN	.041 rpm	2	7	± .10 rpm	
IV, V & VI (P-7)	ΔV, $V > 2$ m/sec	.085%	29	11	2 1/2%	Spin and orientation control needed
	ΔV, $V \leq 2$ m/sec	2.65%	2	21	5%	
	$\Delta \alpha$	.14°	2	31	± .20°	
	ΔN	.041 rpm	2	7	± .10 rpm	
VII, VIII & IX (P-10)	ΔV, $V > 2$ m/sec	.068%	29	11	2 1/2%	Spin and orientation control needed
	ΔV, $V \leq 2$ m/sec	2.3%	2	21	5%	
	$\Delta \alpha$	.11°	2	31	± .20°	One engine
	ΔN	.041 rpm	2	7	± .10 rpm	
X, XI, & XII (P-12)	ΔV, $V > 2$ m/sec	.058%	29	11	2 1/2%	Spin and orientation control needed
	ΔV, $V \leq 2$ m/sec	3.62%	1	21	5%	
	$\Delta \alpha$	.17°	2	31	± .20°	One engine, spin control needed
	ΔN	.041 rpm	2	7	± .10 rpm	
XIII, XIV & XV (P-13)	ΔV, $V > 2$ m/sec	.068%	29	11	2 1/2%	Spin and orientation control needed
	ΔV, $V \leq 2$ m/sec	2.3%	2	21	5%	
	$\Delta \alpha$	.17°	2	31	± .20°	One engine, spin control needed
	ΔN	.041 rpm	2	7	± .10 rpm	

FOLDOUT FRAME 2

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4.4.3(a)4 Maneuvering Error Study

The error analysis presented in the following paragraphs is the basis for determining the maneuvering accuracy associated with the candidate systems and, in particular, the five candidate engine placement concepts. The results of the error study were also used to determine the operational procedures necessary to perform the mission maneuvers within the allowable accuracy and disturbance limits (such as velocity perturbation during spin rate change). The origins of the various maneuver errors are described in this section and the error equations are presented in summary form by Table 4.4.3(a)-IX. Specification allowable errors are presented in Table 4.4.3(a)-VII.

PLANETARY EXPLORER ERROR ANALYSIS:

There are nine basic errors associated with the three propulsion system maneuvers (ΔV, ΔN, $\Delta\alpha$). These errors may be represented as follows:

RESULT

	Error in → / Caused by ↓	V	α	N
Intended Maneuver	ΔV	$\frac{dV}{\Delta V}$	$d\alpha_{\Delta V}$	$dN_{\Delta V}$
	$\Delta\alpha$	$dV_{\Delta\alpha}$	$\frac{d\alpha}{\Delta\alpha}$	$dN_{\Delta\alpha}$
	ΔN	$dV_{\Delta N}$	$d\alpha_{\Delta N}$	$\frac{dN}{\Delta N}$

$$= \begin{pmatrix} E_{11} & E_{12} & E_{13} \\ E_{21} & E_{22} & E_{23} \\ E_{31} & E_{32} & E_{33} \end{pmatrix}$$

Error Matrix

$\frac{dV}{\Delta V}$, $\frac{d\alpha}{\Delta\alpha}$ and $\frac{dN}{\Delta N}$ are the direct errors in ΔV, $\Delta\alpha$ and ΔN expressed as fractions of the magnitudes of the intended maneuvers.

$d\alpha_{\Delta V}$ (typically) is the subsidiary error or disturbance in α caused by a ΔV maneuver.

The following pages present equations for the nine errors expressed in terms of engine and vehicle parameters for which values can be obtained from test data and/or judicious estimates. The symbols used in the equations are defined in Table 4.4.3(a)-VII

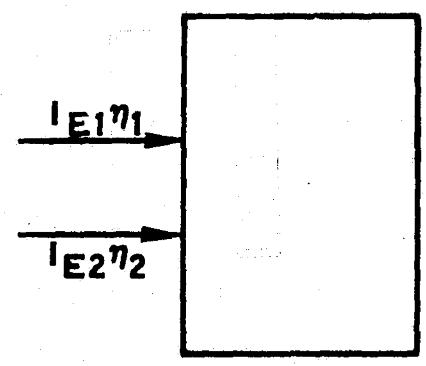
TABLE 4.4.3.1-Vii ERROR EQUATION NOMENCLATURE

Symbol	Nomenclature	Units
V	Spacecraft velocity	Ft/Sec
α	Angle of spin axis to ecliptic plane	Radians
N	Spacecraft spin rate - RPM	RPM
ΔV	Maneuver magnitude - velocity change	Ft/Sec
$\Delta \alpha$	Maneuver Magnitude - Precession	Radians
ΔN	Maneuver Magnitude - Spin Rate Change	RPM
dV	Error Magnitude - velocity	Ft/Sec
$d\alpha$	Error Magnitude - Spin axis angle	Radians
dN	Error Magnitude - Spin rate	RPM
ω	Angular velocity about spin axis $(\omega \frac{60}{2\pi}) = N$	Rad/Sec
M_v	Vehicle mass	LBM
g_o	32.16 lbm/slug or ft/sec^2	
I_{zz}	Vehicle moment of inertia about spin axis	Slug-Ft^2
n	Impulse Effectiveness	--
$I_{E\,\pi}$	Engine impulse produced by ΔN engine(s) firing for a time equivalent to 180° rotation of the vehicle about the spin axis	LBF-SEC
$r\alpha$	Moment arm or $\Delta\alpha$ thrustors	
$\Delta \eta$	Deviation from nominal of impulse effectiveness for a Δ V engine.	--
I_E	Raw impulse produced by an engine	
ϵ	$\epsilon = \frac{dI_E}{I_{E_{Nom}}}$ = Fraction deviation of raw engine impulse from nominal	
$r_{\epsilon\alpha}$, $r_{\epsilon N}$	Axial and radial components of the C.G. displacement from its presumed location relative to the planes of the Δ N and $\Delta\alpha$ couples.	
$drE\alpha C$	Portion of $r_{\epsilon\alpha}$ caused by calibration errors	FT
$drE\alpha F$	Portion of $r_{\epsilon\alpha}$ resulting during flight	FT
dRENC	Portion of $r_{\epsilon N}$ caused by calibration errors	FT
dRENC	Portion of $r_{\epsilon N}$ resulting during flight	FT
ξ	$\frac{I_{E1} - I_{E2}}{I_{E\ NOM.}}$ = Engine to engine impulse repeatability	--
$r\,\alpha N$	Distance from plane or $\Delta\alpha$ couple to spin axis	FT
$r\,N\alpha$	Axial distance between C.G. and plane of Δ N couple (or single engine)	
$d\Theta G$	Thrust vector angular perturbation due to gas flow direction	
$d\Theta N$	Thrust vector angular perturbation due to nozzle characteristics	
dX	Uncertainty in geometrical location of thrust vector line of action	
dI_{ET}	Test to test engine impulse uncertainty	

4.4.3(a)4 (continued)

E11

$$\frac{dV}{\Delta V} = \sqrt{\epsilon^2 + \Delta\eta^2}$$

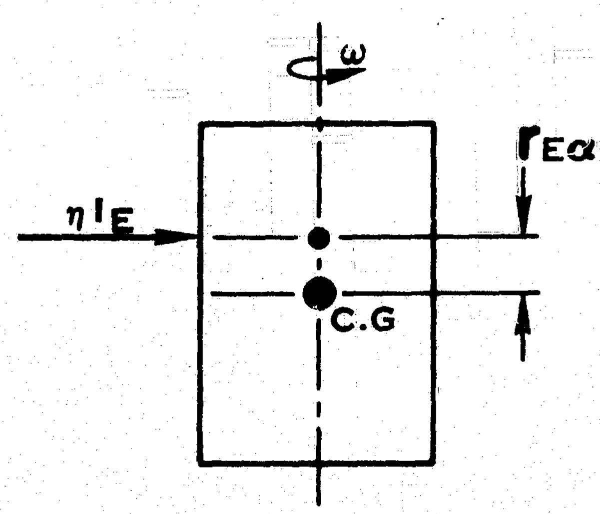


Source of Error:

The raw engine impulse and/or the impulse effectiveness deviate from the values upon which the firing time for the maneuver was based.

E12

$$d\alpha_{\Delta V} = \frac{M_V r_{E\alpha} \Delta V}{\omega \; I_{zz} \; g_o}$$



Source of Error:

Line of action of effective impulse vector from ΔV engines is displaced by $r_{E\alpha}$ from the CG, causing an unwanted precession torque.

$$r_{E\alpha} = \sqrt{dr_{E\alpha c}^2 + dr_{E\alpha f}^2}$$

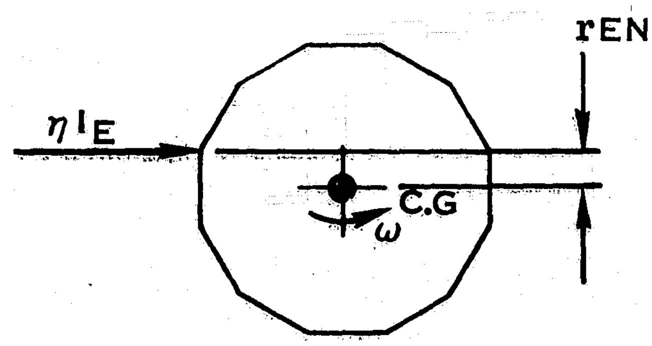
The uncertainty in CG location, $r_{E\alpha}$, is the statistical sum of the calibration error $dr_{E\alpha}$ c and the flight uncertainty $dr_{E\alpha}$ F.

A description of the factors comprising $dr_{E\alpha}$ c, $dr_{E\alpha}$ F are given in Table 4.4.3(a)-VIII.

4.4.3(a)4 (continued)

E13

$$dN_{\Delta V} = \frac{M_V \, r_{EN} \, \Delta V}{\eta \, I_{zz} \, g_o} \, \frac{30}{\pi}$$



Source of Error:

Line of action of effective ΔV engine impulse vector is displaced radially from the spin axis (CG), thus producing a spin torque.

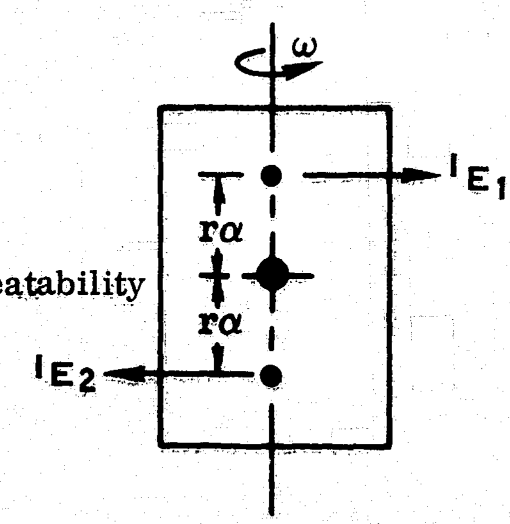
$$r_{EN} = \sqrt{(dr_{ENC})^2 + (dr_{ENF})^2}$$

The reasoning here is identical to that used for $r_{E\alpha}$ in E_{12} (page).

E21

$$dV_{\Delta a} = \frac{g_o \, \xi \, \omega \, I_{zz} \, \Delta a}{2 \, r_a \, M_V}$$

$$\xi = \frac{I_{E_1} - I_{E_2}}{I_{E_{nom}}} = \text{Unit-to-Unit Impulse Repeatability}$$



Source of Error:

Variation in impulse delivered by members of Δa couple produce a net impulse imbalance and hence an unwanted ΔV.

4.4.3(a)4 (continued)

E22

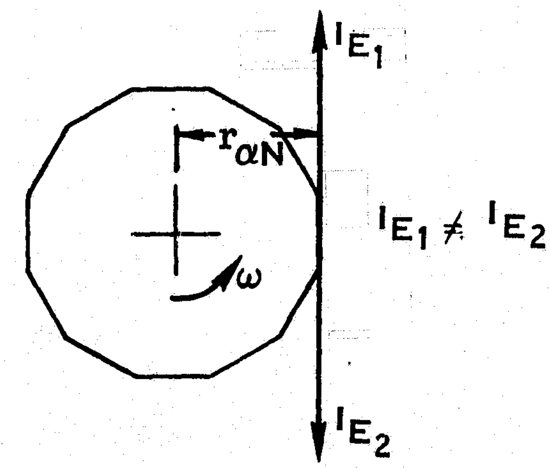
$$\frac{d\alpha}{\Delta\alpha} = \frac{\eta r_a \, I_{bit\ min}}{\omega I_{zz} \Delta\alpha}$$

For this maneuver, it is assumed that one or more refirings will be conducted to trim residual errors left by previous firings. The final residual error, therefore, will be no larger than the $\Delta\alpha$ associated with the minimum impulse bit of the $\Delta\alpha$ system.

E23

$$dN_{\Delta\alpha} = \frac{r\alpha_N \quad \omega \ \xi \ \Delta\alpha}{2 \ \eta \ r_a} \ \frac{30}{\Pi}$$

$$\xi = \frac{I_{E1} - I_{E2}}{I_{E\ nom}} = \text{Unit-to-Unit Impulse Repeatability}$$



Source of Error:

Imbalance in $\Delta\alpha$ couple produces spin error torque.

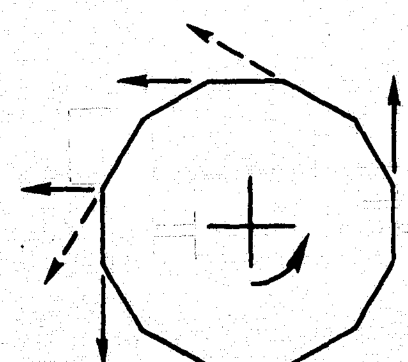
E31

$dV_{\Delta N}$ (max) =

1. $= \dfrac{2 \ go \ I_E \pi \xi}{\pi M_V}$ (pure couple)

ξ reflects impulse imbalance of couple

2. $= \dfrac{2 \ go \ I_E \pi}{\pi M_V}$ (single engine)



Source of Error:

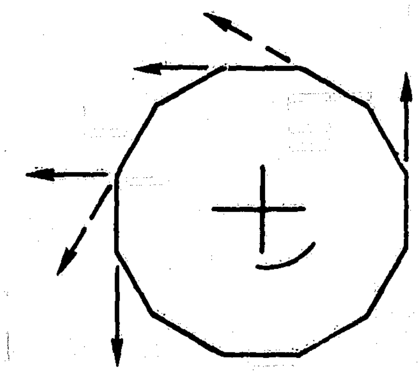
The maximum error shown here is of a residual nature and is caused by either an unbalanced ΔN couple or (even worse) a single ΔN unit being fired for an odd number of 1/2 revolutions.

4.4.3(a)4 (continued)

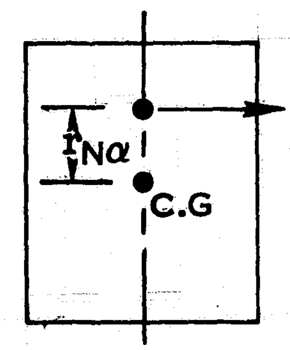
E32

$d\alpha_{\Delta N}$
(max)

1. $\dfrac{2 I_E \pi \xi r_{N\alpha}}{\pi \omega I_{zz}}$ two engines thrusting in opposite directions



2. $\dfrac{2 I_E \pi r_{N\alpha}}{\pi \omega I_{zz}}$ single engine or two engines thrusting in same direction

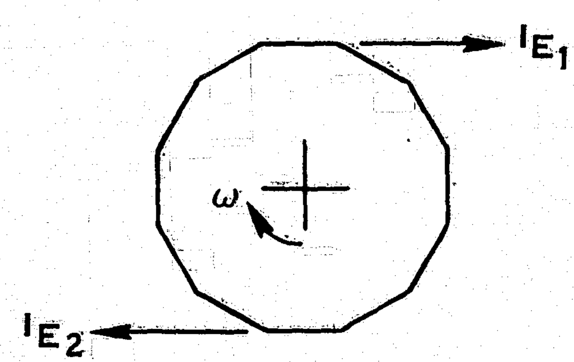


Source of Error:

An imbalance in impulse between the two spin engines or the use of a single engine or two engines thrusting in the same direction for spin rate change causes a precession impulse if the firing period occurs over an uneven number of 1/2 revolutions.

E33

$$\frac{d\Delta N}{\Delta N} = \epsilon\sqrt{\#\text{ of engines}}$$



$I_{E1} = I_{E\,nom}(1 \pm \epsilon)$ etc.

Source of Error:

Impulse of $\Delta\omega$ thrusters differs from nominal.

4.4.3(a)4 (continued)

$dr_{E\alpha C}$ (appears in E_{12})

It is assumed that a calibration maneuver will be performed to determine $r_{E\alpha c}$

$$r_{E\alpha C} = \frac{\Delta\alpha_C \, \omega \, I_{zz}}{\eta I_{ET}} \qquad \Delta\alpha_C \quad \text{observed attitude change}$$

which is the vertical offset of the ΔV cluster thrust vector - spin axis intersected from the CG. Future firings will be modulated to displace the thrust vector this amount to trim out the error.

The term $dr_{E\alpha C}$ is the second order error introduced by uncertainties encountered during the calibration itself.

$$dr_{E\alpha C} = \frac{\omega I_{zz}}{\eta I_{ET}} d\Delta\alpha_C + \frac{\Delta\alpha_C \, \omega \, I_{zz}}{\eta^2 I_{ET}} d\eta_C + \frac{\Delta\alpha_C \, \omega}{\eta I_{ET}} dI_{zz}$$

$$+ \frac{\Delta\alpha_C I_{zz}}{\eta I_{ET}} d\omega_C + \frac{\Delta\alpha_C \, \omega \, I_{zz}}{\eta I_{ET}^{\,2}} dI_{ET}$$

(all + signs indicate statistical addition)

or

$$\frac{dr_{E\alpha C}}{r_{E\alpha C}} = \sqrt{\left(\frac{d\Delta\alpha_C}{\Delta\alpha_C}\right)^2 + \left(\frac{d\eta_C}{\eta}\right)^2 + \left(\frac{dI_{zz}}{I_{zz}}\right)^2 + \left(\frac{d\omega_C}{\omega_C}\right)^2 + \left(dr_{E\alpha F}\right)^2}$$

The terms $\frac{d\Delta\alpha_C}{\Delta\alpha_C}$ etc. represent the fractional uncertainties in the parameters supposedly known or measured during the calibration.

dr_{ENC} (appears in E_{13})

During the same calibration used to measure $r_{E\alpha C}$, the spin rate perturbation $\Delta\omega$ will be measured and r_{ENC} will be determined.

$$r_{ENC} = \frac{\Delta\omega \, I_{zz}}{I_{ET}}$$

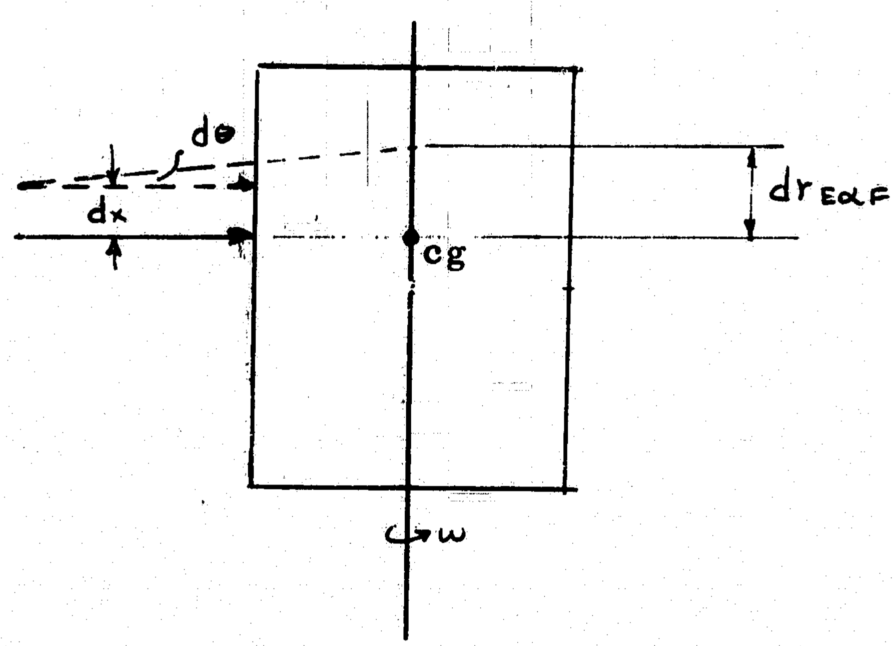
4.4.3(a)4 (continued)

r_{ENC} is the "off axis" displacement of the ΔV resultant thrust vector. This calibration will allow r_{ENC} to be trimmed out by ΔV cluster pulse modulation. An error, caused by inaccuracies in the calibration, is given by:

$$\frac{dr_{ENC}}{r_{ENC}} = \sqrt{\left(\frac{d\Delta\omega}{\Delta\omega}\right)^2 + \left(\frac{dI_{zz}}{I_{zz}}\right)^2 + \left(dr_{E\alpha F}{}^2\right)}$$

Flight Uncertainty

$dr_{E\alpha F}$ (appears in E_{12})



The flight uncertainty in $r_{E\alpha}$ results from the thrust vector location uncertainty d x and the angular uncertainty dθ.

$$dr_{E\alpha F} = \sqrt{\left(\frac{\partial r_{E\alpha}}{\partial\theta} d\theta_G\right)^2 + \left(\frac{\partial r_{E\alpha}}{\partial\theta} d\theta_N\right)^2 + \left(\frac{\partial r_{E\alpha}}{\partial X} dx\right)^2 + \left(\frac{dI_{ET}}{I_{ET}}\right)^2}$$

Similarly, dr_{ENF} (appears in E_{13})

$$dr_{ENF} = \sqrt{\left(\frac{\partial r_{EN}}{\partial\theta} d\theta\, G\right)^2 + \left(\frac{\partial r_{EN}}{\partial\theta} d\theta_N\right)^2 + \left(\frac{\partial r_{EN}}{\partial X} dX\right)^2 + \left(\frac{dI_{ET}}{I_{ET}}\right.}$$

TABLE 4.4.3.A-VIII. ERROR LIMITS FOR MANEUVERS

Error In → / Caused By ↓	ΔV	$\Delta \alpha$	ΔN
ΔV	2 1/2% $\Delta V > 2$ 5% ($.1 \leq \Delta V \leq 2$) m/sec .5° direction	± 6° (Normal) ± 7° (Degraded)	0.3 rpm (Normal) 0 - .6 rpm (Degraded)
$\Delta \alpha$	N. R.	.2° Resolution ± 10% Accuracy	0.3 rpm (N) 0.6 RPM (Deg)
ΔN	N. R.	± 6° (Normal) ± 7° (Degraded)	± .1 rpm

NOTE: The above error limits are those maximum allowable errors specified in GSFC Specification Number S-723-P-10, Rev. A.

TABLE 4.4.3(a) - IX. MANEUVERING ERROR EQUATIONS

Error In / Caused By	V	α	N
ΔV	$\frac{dV}{\Delta V} = \sqrt{\epsilon^2 + \Delta\eta^2}$	$d\alpha_{\Delta V} = \frac{M_V \; r_{E\alpha} \; \Delta V}{\omega \, I_{zz} \; g_o}$	$dN_{\Delta V} = \frac{M_V \, r_{EN} \Delta V \; 30}{\eta \, I_{zz} \; g_o \; \pi}$
$\Delta\alpha$	$dV_{\Delta\alpha} = \frac{g_o \quad \xi \omega \, I_{zz} \Delta\alpha}{2 \quad r\alpha \quad M_V}$	$\frac{d\alpha}{\Delta\alpha} = \frac{\eta \, r\alpha \;\; I_{Bit\,min}}{\omega \, I_{zz} \, \Delta\alpha}$	$dN_{\Delta\alpha} = \frac{(r\alpha N)\xi \;\; \omega \, \Delta\alpha \, 30}{2 \; \eta \; r \quad \pi}$
ΔN	$dV_{\Delta N}^{*} =$ (max) 1) $= \frac{2 \; g_o \; (I_E \pi)\xi}{\pi \, M_V}$ (pure) 2) $= \frac{2 \, g_o \, (I_E \pi)}{\pi \;\; M_V}$ (one engine)	$d\alpha_{\Delta N}^{*} =$ (max) 1) $= \frac{2(I_E \pi)\xi \quad r_N \alpha}{\pi \omega \; I_{zz}}$ (pure) 2) $= \frac{2(I_E \pi) \;\; r_N \alpha}{\pi \, \omega \, I_{zz}}$ (one engine)	$\frac{dN}{\Delta N} = \epsilon \sqrt{\text{No. of engines}}$

* Corresponds to steady state firing of an odd number of 1/2 revolutions

4.4.3(a)5 Error Matrix

The error matrix Table 4.4.3(a)-X presents the magnitudes of the significant errors, and also the resulting perturbations produced by maneuvers. Since the purpose of the matrix is to compare the candidate systems, the matrix is not presented for one mass configuration, but each error is presented for the worst part of each mission. The spin speed is assumed to be constant for each mission, even though it varies considerably at the beginning of both missions and the end of the probe mission.

VELOCITY ERROR MAGNITUDE (dV/DV):

The velocity error magnitude is a function of the fractional deviation of raw engine impulse from the nominal, and the deviation of the impulse effectiveness from the nominal, both added statistically. For candidate systems with one to four ΔV engines, the fractional deviation of raw engine impulse (ϵ) varies from .0362 to .0597, while the deviation of impulse effectiveness ranges from only .0011 to .0022. It is apparent from this that the predominant influence on the velocity error magnitude is the fractional deviation of the raw engine impulse. For candidate systems with engine placement configuration P5, which has four ΔV engines, the magnitude of the velocity error is 5.98%. This necessitates trim maneuvers for all ΔV maneuvers since the maximum allowable error is 2 1/2% on ΔV maneuvers greater than 2 meters/sec and 5% on ΔV maneuvers equal to or less than 2 meters/sec. The P10, P12 and P13 engine placement configurations have errors lower than 5%, while P7 is slightly higher (5.31%).

PRECESSION ERROR MAGNITUDE ($d\alpha / \Delta\alpha$):

The magnitude of this error is also a function of the fractional deviation of the raw engine impulse from the nominal statistically added to the deviation of the impulse effectiveness from the nominal. However, the accuracy requirement of $\Delta\alpha$ maneuvers is such that one or more trim maneuvers will be required which will give a final error which is a function of the minimum impulse of the $\Delta\alpha$ engines. The latter error magnitude is the one presented in the error matrix and is shown assuming all of the $\Delta\alpha$ engines are fired. This error can be reduced to approximately one-half the values shown in the error matrix for all configurations except P7 by using only one $\Delta\alpha$ engine for the final fine trim maneuver. The minimum impulse used was based upon a 39 millisecond electrical pulse width which results from the 128 sector device in the spacecraft logic circuit and a spin speed of 12 rpm. Since this error is inversely proportional to spin speed, the matrix presents the worst case errors. These errors for all the $\Delta\alpha$ engines firing range from .17 degrees for the P7 configuration during the Orbiter Mission up to .36 degrees for P5, P10, P12 and P13 configurations for the Probe Mission. As previously mentioned, these errors can be reduced to approximately one-half by using only one $\Delta\alpha$ engine for the final fine trim maneuver. The only exception to this is the P7 configuration, which has three $\Delta\alpha$ engines and, depending upon the center of gravity location of the spacecraft, firing one engine could reduce the residual error to less than one-third the value in the matrix.

4.4.3(a)5 (continued)

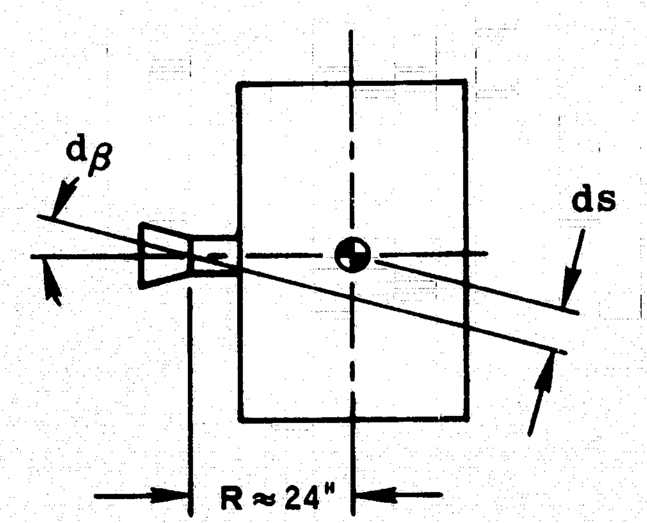
SPIN SPEED ERROR (dN/ΔN):

The only significant contributor to spin speed error is the fractional deviation of the raw engine impulse from the nominal value used in determining engine on-time. Since each candidate system has two ΔN engines, the resulting statistical sum of the deviation of raw engine impulse is the same (6.4%).

PRECESSION PERTURBATION DUE TO VELOCITY CHANGES ($D\alpha/\Delta V$)

A precession perturbation results from any maneuver where the thrust vector is displaced from the instantaneous location of the spacecraft's center of gravity. In this error analysis, it was assumed that at any instant in time the location of the spacecraft's center of gravity can be calculated to within ±.10 inch parallel to the spin axis and .050 inch radially from the spin axis. Knowing the center of gravity location, the ΔV engines would be modulated in order to place the final resulting impulse line of action through the calculated center of gravity location. Therefore, the precession perturbation becomes a function of the uncertainty in the spacecraft's center of gravity location ($r_{E\alpha}$), which, in turn, is the statistical sum of the calibration error ($dr_{E\alpha_C}$) and the flight uncertainty ($dr_{E\alpha F}$). Before the first mid-course correction in both the Probe and Orbiter missions, a calibration will be performed where the ΔV engines are fired and the resulting perturbations measured. This will make it possible to determine the spacecraft's center of gravity location and form a reference point for future calculations.

For all candidate system engine placement configurations, except P5, the major contributor to the uncertainty in center of gravity location ($r_{E\alpha}$) is the flight uncertainty ($dr_{E\alpha F}$). In these candidate systems, the ΔV engine locations are such that small angular movements of the line of action of the effective impulse cause significant displacements where the vector passes the center of gravity.



4.4.3(a)5 (continued)

For example, a $d\beta$ of only .20 degrees would give a displacement, dS, of .084 inch, which is sufficient to give the size errors presented for candidate systems with engine configurations P7 to P13. Candidate systems with engine configuration P5 have the engines on the sides of the spacecraft arranged such that R, as defined above, equals zero. As a result, this configuration only produces a 2.75-4.6 degree perturbation for a 100 meter/second maneuver, while perturbations as high as 28.7 degrees would be produced for systems with configurations P7 to P13 if this maneuver were made in one step.

SPIN PERTURBATION DUE TO VELOCITY CHANGES ($dN/\Delta V$):

A spin perturbation results from any maneuver where the thrust vector is displaced from the instantaneous location of the spacecraft's center of gravity. This perturbation is similar to the $d\alpha/\Delta V$. In this case, the resulting moment about the spacecraft's center of gravity is about the spin axis, while for the $d\alpha/\Delta V$ perturbation, the moment was about an axis normal to the spin axis.

The spin perturbation is a function of the uncertainty in the spacecraft's center of gravity location (r_{EN}) which, in turn, is largely dependent upon the flight uncertainty (dr_{ENF}) in all cases except candidate systems with engine configuration P5. The geometric locations of the ΔV thrusters have the same effect on the displacement of the line of action of the effective ΔV impulse as was described for the precession perturbation.

For the P5 engine configuration, this perturbation amounts to .34 to .71 rpm for a 100 meters/second maneuver. The other candidate systems produce perturbations in the range of 3.77 to 7.9 rpm for a 100 meter/second maneuver. In view of the fact that the spacecraft's spin speeds during ΔV maneuvers are only 12 rpm for the Probe Mission and 15 rpm for the Orbiter Mission, these perturbations will necessitate complex operational procedures.

VELOCITY PERTURBATIONS DUE TO PRECESSION MANEUVERS ($dV/\Delta\alpha$):

Under ideal conditions, the engines performing a precession maneuver produce an identical total impulse. However, variations in impulse represented by the engine-to-engine impulse repeatability factor (ξ) are inherent in engines, and this results in an impulse imbalance between the $\Delta\alpha$ engines which imparts a velocity change to the spacecraft. For all the two engine configurations (P5, P10, P12, and P13) this disturbance amounts to only .053 meters/second per radian precession for the Orbiter Mission. For the P7 configuration, this error is only as high as .077 meters/second per radian precession. Since the smallest mid-course corrections in both missions are 2 meters/second, this error should not be a major determining factor in selecting a candidate system.

4.4.3(a)5 (continued)

SPIN PERTURBATION DUE TO PRECESSION MANEUVERS ($dN/\Delta\alpha$):

The impulse unbalance resulting from the engine-to-engine impulse repeatability also produces spin error torque in addition to the velocity change earlier described. This torque is produced for engine configurations P5, P12 and P13 only because their $\Delta\alpha$ engines are displaced a distance $r_{\alpha N}$ from the spin axis. Candidate systems P7 and P10 have their $\Delta\alpha$ thrusters arranged such that $r_{\alpha N} = 0$. Any angular motions or displacements of their engines would only produce equivalent $r_{\alpha N}$'s two to three orders of magnitude lower than those of configurations P5, P12 and P13, which would therefore be negligible. For a typical 90° precession maneuver, these errors are as high as 1.22 rpm, which for the P5, P12 and P13 configurations, does represent a significant factor in the comparative evaluation of the systems.

VELOCITY PERTURBATIONS DUE TO SPIN MANEUVERS ($dV/\Delta N$):

There are two basic engine configurations used in performing spin rate maneuvers: P5, P7 and P10 have thrusters which fire in opposite directions, while in configurations P12 and P13, the two ΔN engines fire in the same direction. In both of these arrangements, if all ΔN maneuvers resulted in the ΔN engines being fired an even number of one-half revolutions, the resulting velocity vector caused by the impulse in P12 and P13, and the impulse unbalance in P5, P7 and P13, would sum to zero. Therefore, the maximum error for any size maneuver results from the ΔN engines being fired an odd number of one-half revolutions. For configurations P12 and P13, this perturbation amounts to a maximum of only .12 meter/second for the Orbiter Mission. Configurations P5, P7 and P10 have perturbations nearly two orders of magnitude lower because the thrust unbalance in these systems is producing the change in velocity, rather than the full thrust from two engines.

PRECESSION PERTURBATION DUE TO SPIN MANEUVERS ($d\alpha/\Delta N$):

Performing a spin maneuver with anything but an even number of one-half revolutions will also produce a precession change in addition to the velocity change already described in the previous perturbation description. In configurations P5, P7 and P10, the impulse giving the precession perturbation is caused by the engine-to-engine impulse repeatability (ξ). In configurations P12 and P13, where the ΔN engines point in the same direction, the impulse giving the precession perturbation is the raw impulse of both engines. In these two cases, $r_{N\alpha}$ is the distance from the center of the two engines to the spacecraft's center of gravity. For configurations P5, P17 and P10, this perturbation is relatively small, .197 degrees maximum for the Probe Mission. However, for configurations P12 and P13, this perturbation is as high as 2.32 degrees for the Probe Mission. In view of the accuracy required for precession maneuvers (.20 degrees), which frequently precede spin control maneuvers, this size error would present serious operational restraints.

FOLDOUT FRAME

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TABLE 4.4.3(a) - Xa. ERROR

Cand. System / Error In / Error Caused by	I, II, III (P-5)			IV, V and VI (P-7)			VII, VIII and IX (P	
	2ΔN, 4ΔV, 2Δ α			2ΔN, 3ΔV, 3Δ α			2ΔN, 2ΔV,	
	V	α	N	V	α	N	V	α
ΔV	.0598 ΔV F	.0275 ΔV E	.0034 ΔV E	.0531 ΔV F	.17 ΔV E	.0377 ΔV E	.0454 ΔV F	.17 Δ E
Δ α	.00093 Δα D	* .36 E	.0109 Δα F	.00134 Δα D	* .25 E	Negative	.00093 Δα D	* .36 E
ΔN	.0023 D	.197 A	.064 Δ F	.0023 D	.197 A	.064 ΔN F	.0023 D	.036 A

NOTES: N = 12 RPM

MASS CONFIGURATIONS:

A. Launch
B. Booms Extended
C. Cruise
D. Only Main Probe Separated
E. All Probes Separated
F. All

TABLE 4.4.3(a) - Xb. ERROR M

Cand. System / Error In / Error Caused by	I, II, III (P-5)			IV, V and VI (P-7)			VII, VIII and IX (P	
	2ΔN, 4ΔV, 2Δ α			2ΔN, 3ΔV, 3Δ α			2ΔN, 2ΔV,	
	V	α	N	V	α	N	V	α
ΔV	.0598 ΔV F	.046 ΔV C	.0071 ΔV C	.0531 ΔV F	.287 ΔV C	.079 ΔV C	.0454 ΔV F	.287 Δ C
Δ α	.00048 Δα E	* .25 E	.0135 Δα F	.00072 Δα E	* .170 E	Negative	.00048 Δα E	* .25 E
ΔN	.0029 D	.125 A	.064 ΔN F	.0029 D	.125 A	.064 ΔN F	.0029 D	.032 A

NOTES: N = 15 RPM

MASS CONFIGURATION:

A. Launch
B. Booms Erected
C. Cruise

D. Orb
E. Fue
F. All

* This error is actually a resolution based u
engines normally used to perform a Δ α m

Xa. ERROR MATRIX - PROBE MISSION

III and IX (P-10)			X, XI and XII (P-12)			XIII, XIV and XV (P-13)		
2ΔN, 2ΔV, 2Δα			2ΔN, 1ΔV, 2Δα			2ΔN, 2ΔV, 2Δα		
	α	N	V	α	N	V	α	N
ΔV	.17 ΔV E	.0377 ΔV E	.0362 ΔV F	.17 ΔV E	.0377 ΔV E	.0454 ΔV F	.17 ΔV E	.0377 ΔV E
Δα	* .36 E	Negative	.00093 Δα D	* .36 E	.0109 Δα F	.00093 Δα D	.36 E	.0109 Δα F
	.036 A	.064 ΔN F	.099 D	2.32 A	.064 ΔN F	.099 D	2.25 A	.064 ΔN F

UNITS: ΔV - Meters/second
Δα - Degrees
ΔN - RPM

b. ERROR MATRIX - ORBITER MISSION

II and IX (P-10)			X, XI and XII (P-12)			XIII, XIV and XV (P-13)		
2ΔN, 2ΔV, 2Δα			2ΔN, 1ΔV, 2Δα			2ΔN, 2ΔV, 2Δα		
	α	N	V	α	N	V	α	N
V	.287 ΔV C	.079 ΔV C	.0362 ΔV F	.287 ΔV C	.079 ΔV C	.0454 ΔV F	.287 ΔV C	.079 ΔV C
Δα	*.25 E	Negative	.00048 Δα E	* .25 E	.0135 Δα F	.00048 Δα E	.25 E	.0135 Δα F
	.032 A	.064 ΔN F	.12 D	.58 A	.064 ΔN F	.12 D	.577 A	.064 ΔN F

D. Orbit
E. Fuel Expended
F. All

ion based upon a .039 second firing of all the Δα
m a Δα maneuver.

4.4.3(b) ENGINE OPERATING LIFE MARGIN

The performance analysis studies conducted for the Planetary Explorer Study utilized the performance data of the IDCSP/A 5 lb_f engine manufactured by Hamilton Standard. This engine was originally qualified at 20,000 pulses with a total of 36 minutes on time. For subsequent applications the IDCSP/A engine has demonstrated 70,000 pulses with a total of 101.5 minutes on time. The results of the performance analysis study relative to the total number of engine pulses and "on" times required for each engine, for each placement concept studied, are presented in Tables 4.4.3(b)-I. This table illustrates the life margin for each candidate subsystem from the standpoint of both number of pulses and total "on" time. Each of the engine numbers in the table is related to the engine identification numbers on Figure 4.1-5 and 4.2-1a through 4.2-1e. Each of the engines has the life margin identified for two operating modes - primary and back-up. The primary mode assumes all engines operational for the entire Planetary Explorer mission where the spacecraft maneuvers are equally divided between the functional groups of engines, with approximately 50 percent of the operations performed by each. The back-up mode assumes all spacecraft maneuvers throughout the mission are performed with one functional group of engines only. As can be seen from the two tables the life performance margin in the primary operational mode is high for all of the engines.

This is true also for the engines operating in the back-up mode, with a few exceptions where the total "on" time exceeds the IDCSP/A engine qualification status (blocks marked with asterisk). In these cases the performance margin for "on" time is compared to demonstrated test of 101.5 minutes.

There is no significant difference between the life margin for the engines for all the candidate subsystems except in the cases where either one or two engines are used to perform the mid-course correction maneuvers. During these events, the greater number of engines used the greater will be the individual engine life margin. In all cases the IDCSP/A engine has sufficient life margin to perform the Planetary Explorer requirements.

FOLDOUT FRAME 1

TABLE 4.4.3(b) -I ENGINE OPERATING LIFE

MISSION	ENGINE		CANDIDA					
			I, II, III } (P-5) OR VII, VIII, IX } (P-10)				IV, V, VI }	
			PULSES		TIME		PULSES	
	NO.	OPERATION MODE	NO.	MARGIN (%)	TOTAL (MIN.)	MARGIN (%)	NO.	MARGIN (%)
ORBITER	1	PRIMARY	1623	92.	13.88	61.	1601	92.
		BACK UP	2824	86.	23.00	36.	3202	84.
	2	PR.	1207	94.	9.32	74.	1940	90.
		B/U	6	100.	0.19	100.	339	98.
	3	PR.	1207	94.	10.62	71.	1946	90.
		B/U	2408	88.	19.75	45.	3547	82.
	4	PR.	1207	94.	9.50	74.	1607	92.
		B/U	6	100.	0.38	99.	6	100.
	5	PR.	1201	94.	10.43	71.	1946	90.
		B/U	2402	88.	19.55	42.	3547	82.
	6	PR.	1617	92.	12.19	67.	1607	92.
		B/U	416	98.	3.07	92.	6	100.
	7	PR.	1201	94.	10.43	71.	—	
		B/U	2402	88.	19.55	46.	—	
	8	PR.	1201	94.	9.13	75.	—	
		B/U	0	100.	0	100.	—	
PROBE	1	PRIMARY	2460	88.	12.09	67.	1080	94.
		BACK UP	3282	84.	19.16	47.	2159	90.
	2	PR.	824	96.	9.12	75.	2158	90.
		B/U	2	100.	2.05	95.	1078	94.
	3	PR.	823	96.	9.12	75.	2158	90.
		B/U.	1645	92.	16.20	55.	3238	84.
	4	PR.	823	96.	7.31	80.	1082	94.
		B/U	1	100.	0.23	99.	2	100.
	5	PR.	821	96.	7.08	81.	2159	90.
		B/U	1643	92.	14.15	61.	3239	84.
	6	PR.	2460	88.	11.85	67.	1081	94.
		B/U	1638	92.	4.78	87.	1	100.
	7	PR	821	96.	7.08	81.	—	
		B/U	1643	92.	14.15	61.	—	
	8	PR.	822	96.	7.08	81.	—	
		B/U	0	100.	0	100.	—	

NOTES: 1) LIFE MARGINS BASED ON IDCSP/A QUALIFICATION OF 20,C
* 2) LIFE MARGINS BASED ON IDCSP/A DEMONSTRATION TEST C
3) ENGINE PLACEMENT CONCEPT P-10 CAN BE OPERATED EI
4) ENGINE NUMBERS RELATE TO ENGINE IDENTIFICATION N
5) PRIMARY OPERATION MODE ASSUMES OPERATION SPLIT
OF ENGINES. , BACK-UP MODE ASSUMES OPERATION PE
GROUP OF ENGINES

ATING LIFE MARGIN

CANDIDATE SUBSYSTEM

IV, V, VI } (P-7)				X, XI, XII } (P-12M)				XIII, XIV, XV } (P-B) OR VII, VIII, IX } (P-10)			
PULSES		TIME		PULSES		TIME		PULSES		TIME	
NO.	MARGIN (%)	TOTAL (MIN)	MARGIN (%)	NO.	MARGIN (%)	TOTAL (MIN)	MARGIN (%)	NO.	MARGIN (%)	TOTAL (MIN.)	MARGIN %
601	92.	13.04	64.	2906	86.	23.09	36.	2404	88.	19.58	46.
202	84.	26.09	28.	5813	71.	46.18	54. *	4808	76.	39.17	61. *
940	90.	15.21	58.	2906	86.	23.09	36.	2820	86.	27.96	22.
339	98.	2.16	95.	0	100.	0	100.	5254	74.	42.24	58. *
946	90.	15.70	58.	422	98.	3.45	90.	2820	86.	27.95	22.
547	82.	28.63	21.	422	98.	3.45	90	416	98.	3.07	92.
607	92.	13.24	64.	6	100.	0.19	100.	2404	88.	19.58	46.
6	100.	0.19	100.	6	100.	0.19	100.	0	100.	0	100.
946	90.	15.39	58.	6	100.	0.38	99.	6	100.	0.38	99.
547	82.	28.44	21.	6	100.	0.38	99.	6	100.	0.38	99.
607	92.	13.42	64.	422	98.	3.26	91.	6	100.	0.19	100.
6	100.	0.38	99.	422	98.	3.26	91.	6	100.	0.19	100.
—		—		—		—		6	100.	0.38	99.
—		—		—		—		6	100.	0.38	99.
—		—		—		—		6	100.	0.19	100.
—		—		—		—		6	100.	0.19	100.
080	94.	5.16	86.	2805	86.	26.20	28.	1620	92.	13.60	64.
159	90.	10.31	71.	5610	72.	52.39	48. *	3239	84.	27.20	24.
158	90.	8.14	78.	2805	86.	26.20	27.	3818	81.	23.58	36.
078	94.	2.97	92.	0	100.	0	100.	5438	73.	37.18	63. *
158	90.	16.18	55.	1629	92.	4.72	87.	3769	81.	23.58	36.
238	84.	21.33	41.	1629	92.	4.72	87.	2147	90.	9.98	72.
082	94.	7.09	81.	2	100.	1.93	95.	1620	92.	13.60	64.
2	100.	1.92	95.	2	100.	1.93	95.	0	100.	0	100.
159	90.	17.88	50.	1	100.	0.23	99.	1	100.	0.23	99.
239	84.	23.03	36.	1	100.	0.23	99.	1	100.	0.23	99.
081	94.	5.39	85.	1630	92.	6.42	82.	2	100.	1.93	95.
1	100.	0.23	99.	1630	92.	6.42	82.	2	100.	1.98	95.
—		—		—		—		1	100.	0.23	99.
—		—		—		—		1	100.	0.23	99.
—		—		—		—		2	100.	1.93	95.
—		—		—		—		2	100.	1.93	95.

ON OF 20,000 PULSES AND 36 MINUTES TOTAL "ON" TIME
TION TEST OF 70,000 PULSES AND 102 MINUTES "ON" TIME
PERATED EITHER AS P-5 OR P-B
FICATION NUMBERS ON FIGURES 4.2-1a THRU 4.2-1e
TION SPLIT APPROX. 50% BETWEEN FUNCTIONAL GROUPS
ERATION PERFORMED WITH ONE FUNCTIONAL

4.4.4 Power

A summary of the power requirements, including total mission power consumption for each of the candidate systems, is shown in Table 4.4.4-I. A comparison of total energy expended for each system shows that for a given mission, the levels are approximately equal, and the same is true for the mission average power. The energy expended versus the mission event for each of the candidate systems, where power conditioning is not used, is shown in Figure 4.4.4-1 for the Orbiter mission, and in Figure 4.4.4-2 for the Probe mission. The use of power conditioning for each of the systems was considered and these results are included in the summary table 4.4.4-I. Power conditioning, as used here, refers to a process whereby the voltage initially supplied to an engine propellant valve is reduced after the valve has opened in order to conserve power. Power conditioning makes use of the fact that it takes significantly less power to maintain a valve in the open position, once it has been opened, than it takes to actuate it from the closed to the open position.

Data from the IDCSP/A engine propellant valve (Hydraulic Research and Manufacturing valve Part No. 48000680) has shown as little as 2.0 VDC may maintain the valve open once the valve has been energized open with the normal operating voltage. A review of the IDCSP/A valve test results has shown that the maximum dropout voltage is 8.0 VDC, which could provide considerable power savings even when allowing a significant margin of safety on the voltage.

In order to provide opening force margin and fast opening response, the operating voltage range of these valves substantially exceeds the pull-in voltage (voltage required to open valve). The HR&M 48000680 valve has an actual pull-in voltage of only 8.8 VDC maximum as compared with the normal operating voltage of 18 to 30 VDC. Therefore, in addition to power conditioning, power savings can also be realized by operating at voltages across the valve coil of less than the 28 VDC supplied to the valve itself.

Several basic power saving techniques exist for pulsed rocket engine valves without steady state performance degradation. Addition of a series resistance (Figure 4.4.4-3b) is the simplest method of power reduction. This method offers modest transient degradation while reducing the power to 50%, and maintaining 14 VDC minimum across the valve coil which is substanially above the pull-in and dropout voltages. Transient performance can be restored by wave shaping the drive pulse. A full power transient "on" pulse that is dropped to save current and to sustain hold-in can be accomplished by the circuit shown in Figure 4.4.4-3c. This circuit allows the current versus time characteristics to be almost identical to the valve without the power conditioning to a point well past the opening when it will then drop the voltage across the valve coil. This is the only true power conditioning circuit where there is no compromise in opening force margin. To attain the maximum power savings with no transient performance degradation, both the transient and steady state portion of the valve drive signal can be interrupted using a solid state chopper as shown in Figure 4.4.4-3d.

4.4.4 (Continued)

However, the dynamic force for this circuit is substantially reduced. Figure 4.4.4-3a shows the standard control circuit for the torque motor. The circuit in Figure 4.4.4-3c would be recommended if power conditioning were used. This circuit offers no compromise in initial current characteristics which means dynamic opening response and opening force margins will be essentially unchanged from the qualification test results. Figure 4.4.4-4 shows the current versus time characteristics of the various power conditioning concepts.

All of the circuits are substantially the same with respect to "on" times, and the reliability of the electronic components are relatively equal especially since the steady state current is reduced in the coil, thereby reducing internal heating effects.

Figure 4.4.4-5 shows the results of power conditioning for the orbiter mission with the P-5 engine placement configuration. As the total values of Table 4.4.4-I indicate, there will be a savings of about 49% in energy expended during the mission. Figure 4.4.4-6 is a typical plot of mission (both orbital and probe) energy as a function of cruise spin rate. This figure shows that savings in energy will be realized over the full range of RPM values by power conditioning the system. The decreasing savings at higher spin rates occurs since the average "on" time of the engines during pulse firing is smaller thus the valve voltage reduction after the first 30 msec firing has less of an effect on power savings.

Power conditioning of the Planetary Explorer candidate systems was also evaluated to determine if a spacecraft weight reduction could be realized. Figure 4.4.4-7 represents a plot of weight savings corresponding to a given electrical energy savings for a typical system. This plot enables a direct reading of what the weight savings will be for a given system (which has a certain energy savings associated with it) when used in conjunction with an energy source which has a certain weight/energy rating. For example, if the configuration described per Figure 4.4.4-7 (P-5 or P-10) were to be used with a silver zinc battery (which has a weight/energy rating in the range of 10-20 lb/kw-hr) the maximum weight savings is read from the figure to be 0.16 lb. In this manner, a trade-off relating to energy source may also be included in the analysis. The figure shows that regardless of whether the energy source is solar cells or silver zinc batteries, there will be a net weight increase if power conditioning is used. This occurs because the weights of the added electronic components and associated packaging exceeds the energy weight savings.

In summary, the results of the power conditioning analysis show that there is a substantial power and energy savings, but, that other factors such as weight and added system complexity (thus decreased system reliability) offset this advantage to a degree that it is not an approach recommended.

TABLE 4.4.4-I. POWER SUMMARY

Mission	Candidate System	Total Energy Expended (watt-sec)		Peak Power (watts)		Maneuver Average Power (watts)	
		Without Power Conditioning	With Power Conditioning	Without Power Conditioning	With Power Conditioning	Without Power Conditioning	With Power Conditioning
ORBIT	I, II, III, VII, VIII & IX (P-5 & P-10)	57215	29180	44.64	22.32	5.29	2.65
	IV, V & VI (P-7)	57383	29265	33.58	16.79	3.98	1.99
	X, XI & XII (P-12)	57182	29163	22.32	11.16	2.65	1.32
	VII, VIII, IX, XIII, XIV & XV (P-13 & P-10)	57226	29185	22.32	11.16	2.65	1.32
PROBE	I, II, III, VII, VIII & IX (P-5 & P-10)	47352	24150	44.64	22.32	5.29	2.65
	IV, V & VI (P-7)	45041	22970	33.58	16.79	3.98	1.99
	X, XI & XII (P-12)	45343	23125	22.32	11.16	2.65	1.32
	VII, VIII, IX, XIII, XIV & XV (P-13 & P-10)	45354	23130	22.32	11.16	2.65	1.32

NOTES: (1) Peak power based on maximum number of simultaneous engine firings with max steady state power requirements per engine as follows:

- Without power conditioning - 11.16 watts at 28 VDC
- With power conditioning - 5.60 watts after 30 msec.

(2) Power conditioning values based on active transistor circuit (Ref. Figure 4.4.4-3C with a 50% reduction in power after 30 msec.

(3) Maneuver average power is equal to peak power for spin maneuvers since the engine is fired in the continous rather than the pulse mode.

FOLDOUT FRAME

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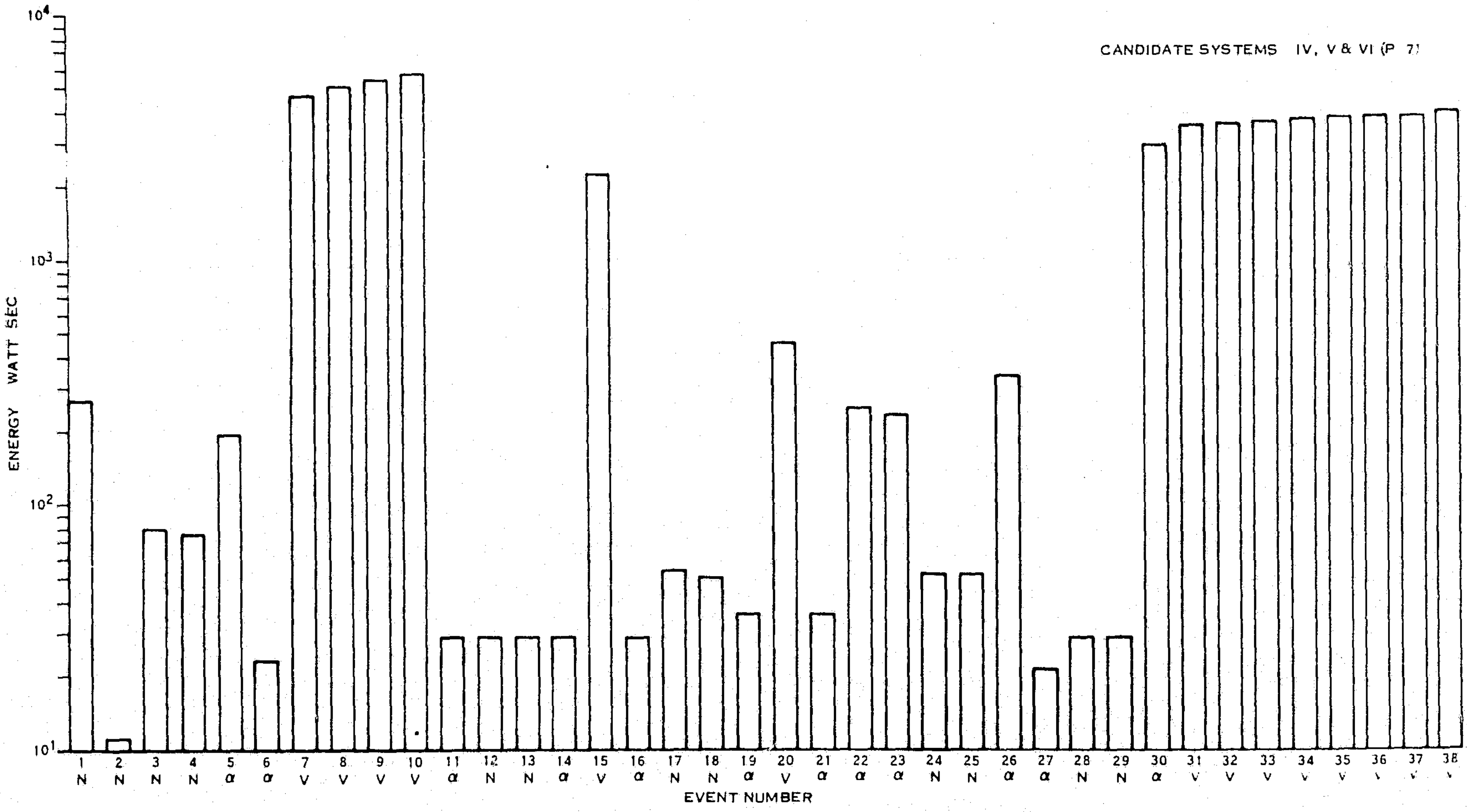
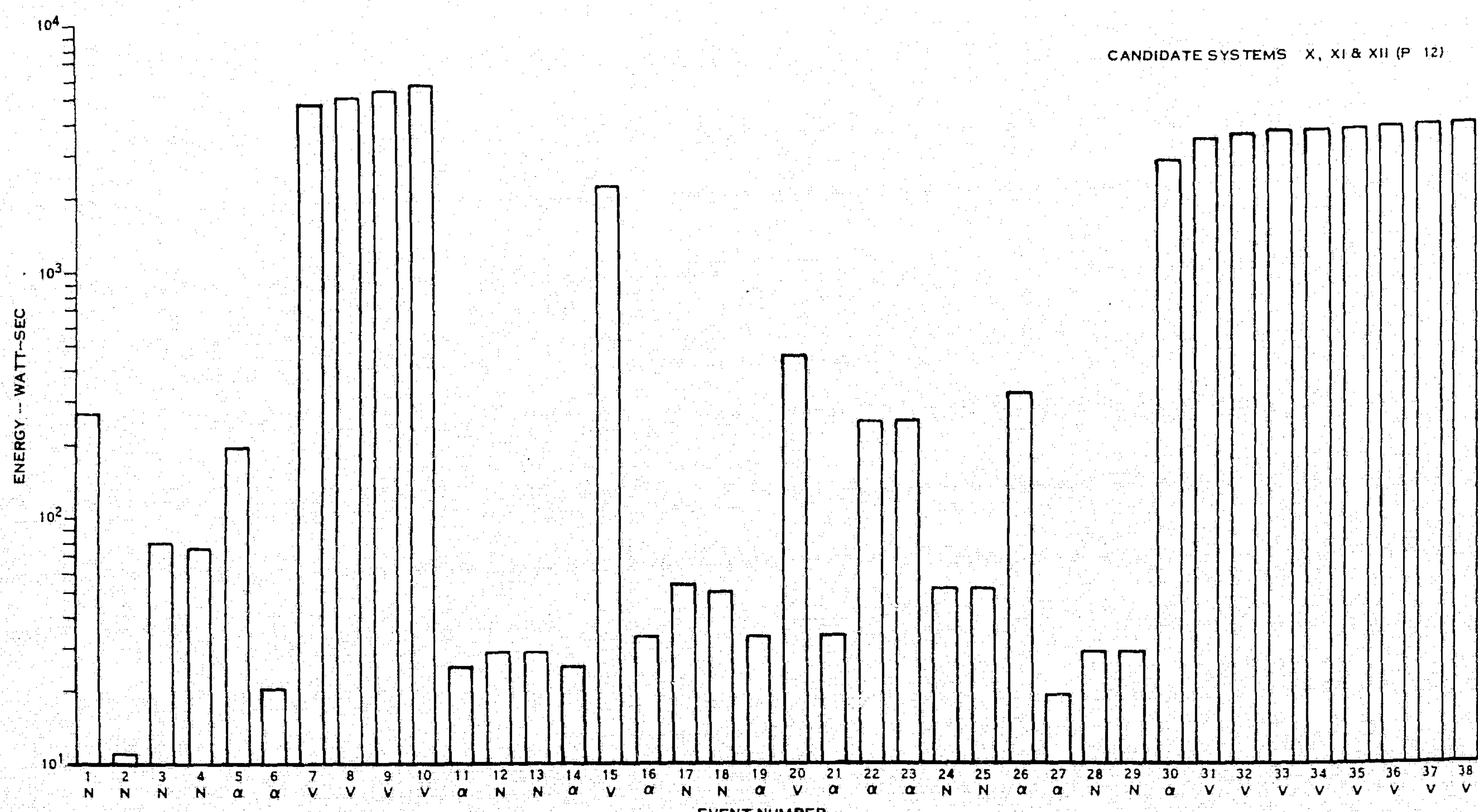
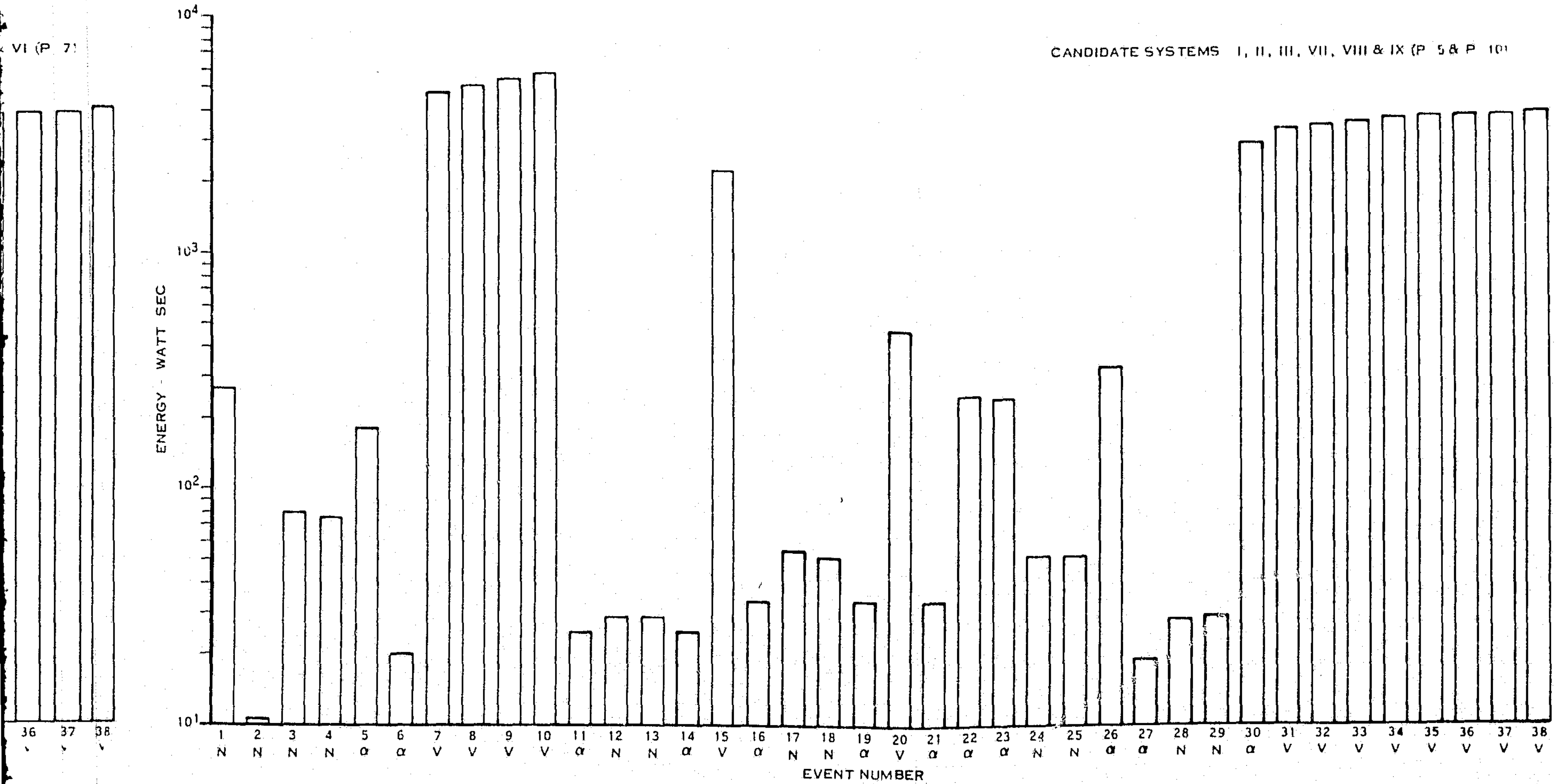






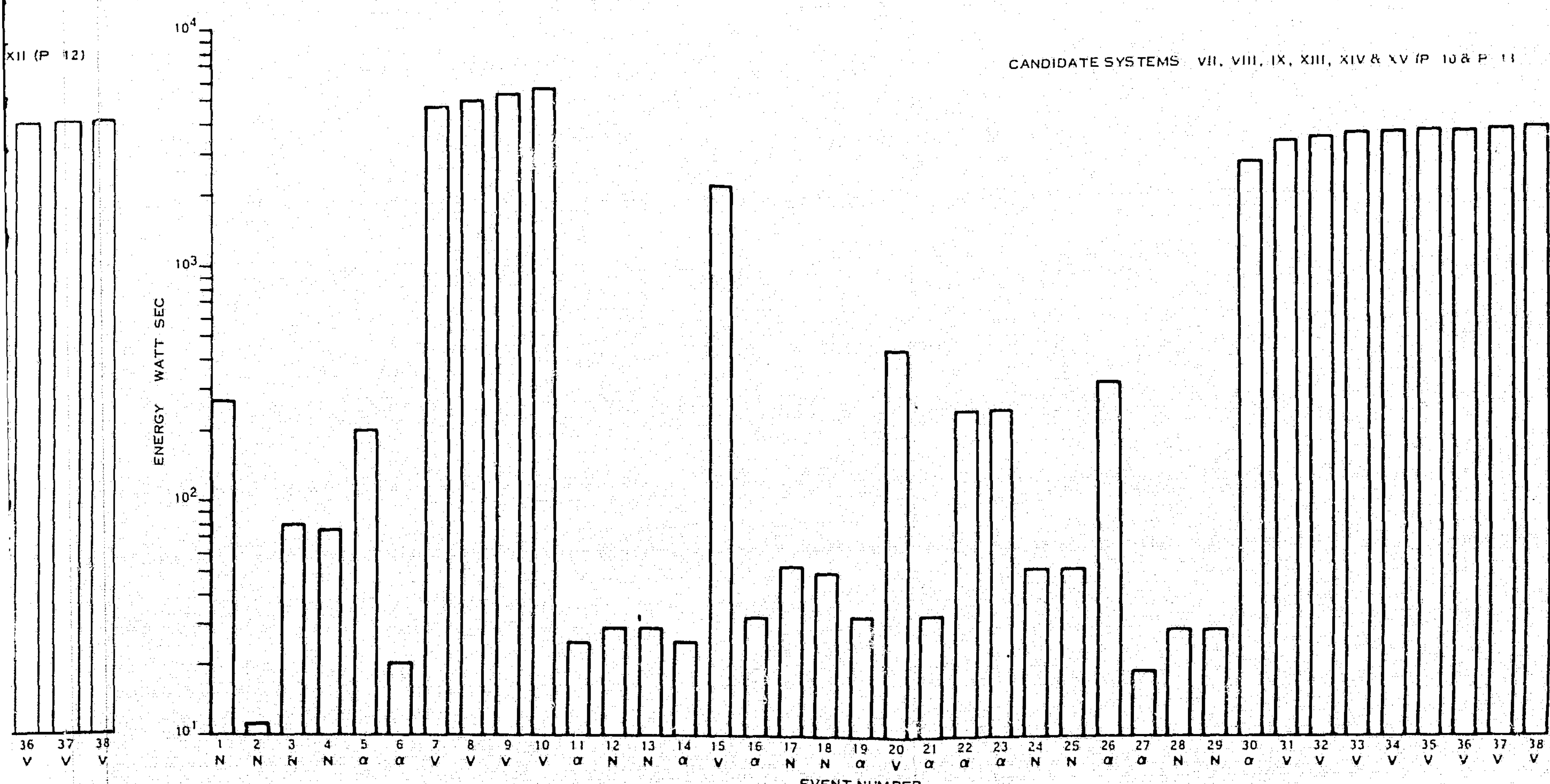
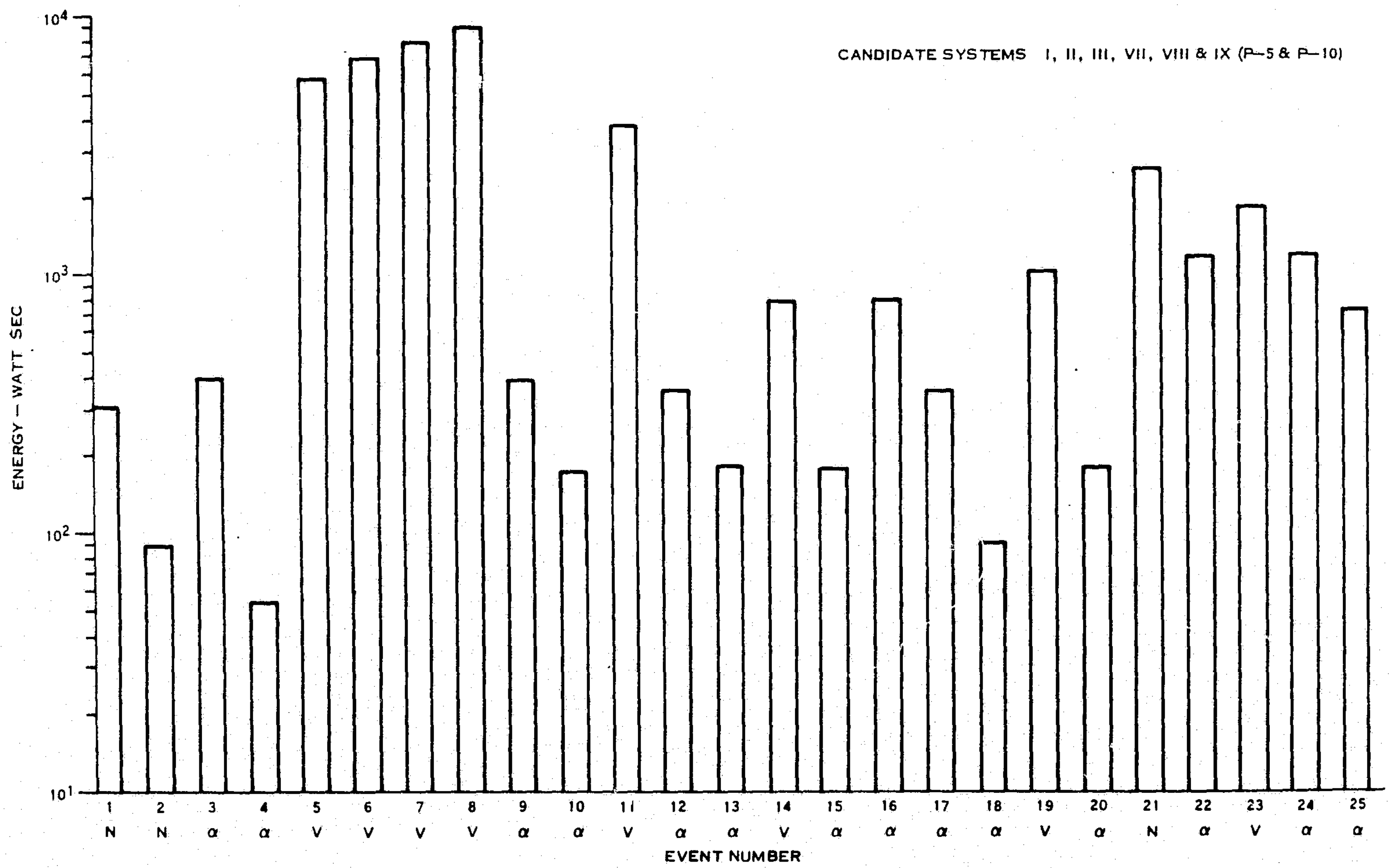
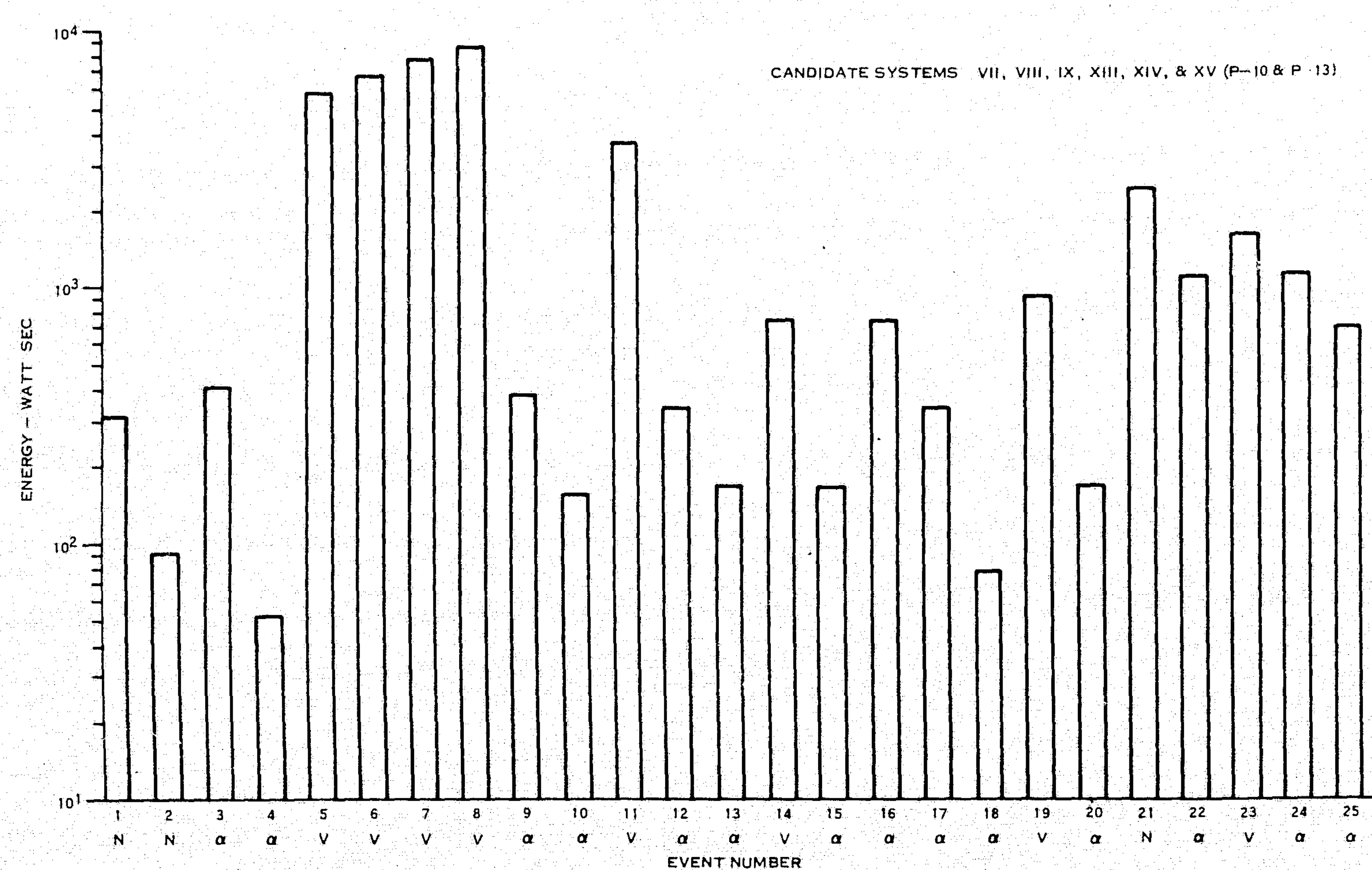


FIGURE 4.4.4-1. ELECTRICAL ENERGY VS MISSION EVENT — ORBITER MISSION

FOLDOUT FRAME

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION





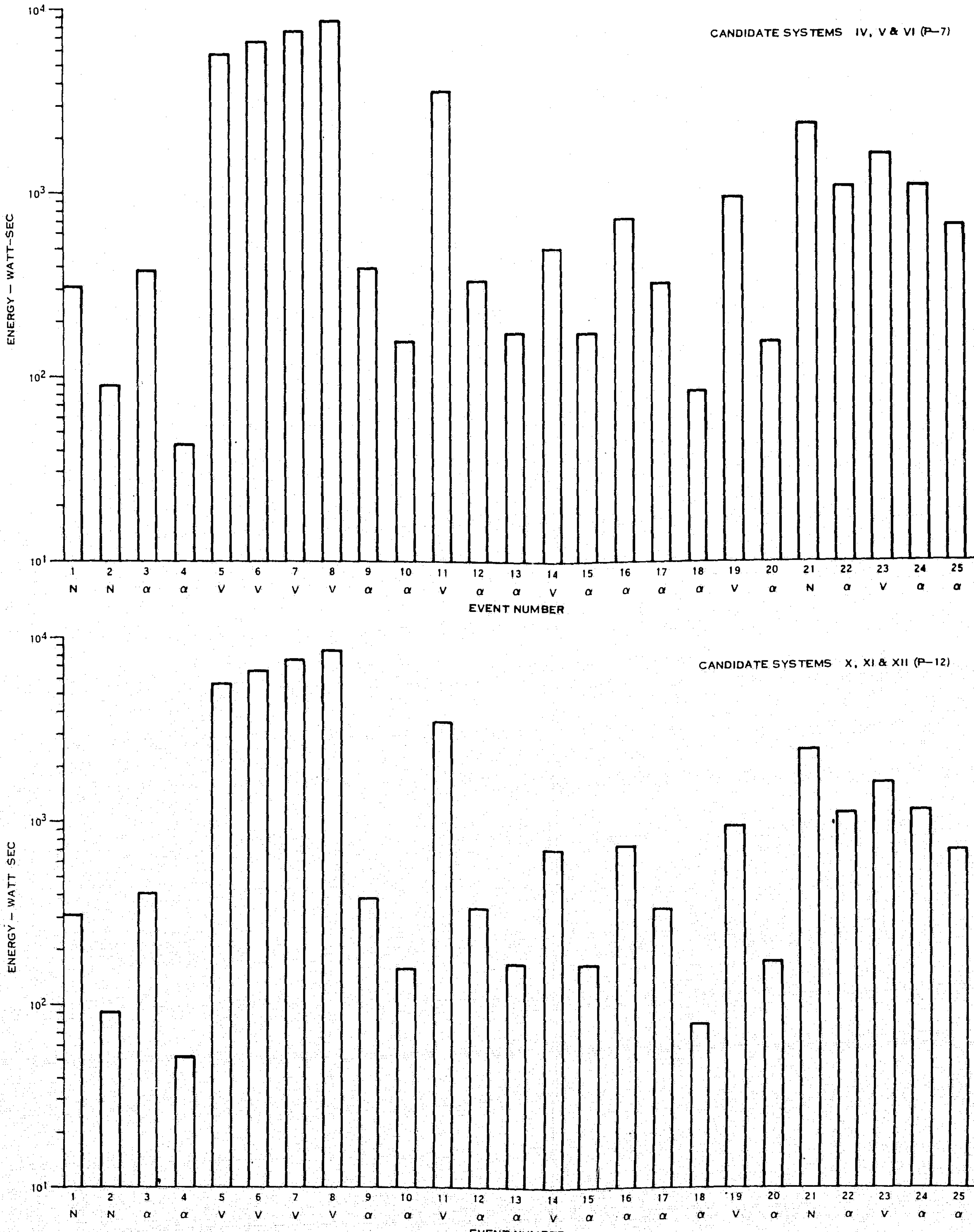


FIGURE 4.4.4–2. ELECTRICAL ENERGY VS MISSION EVENT – PROBE MISSION

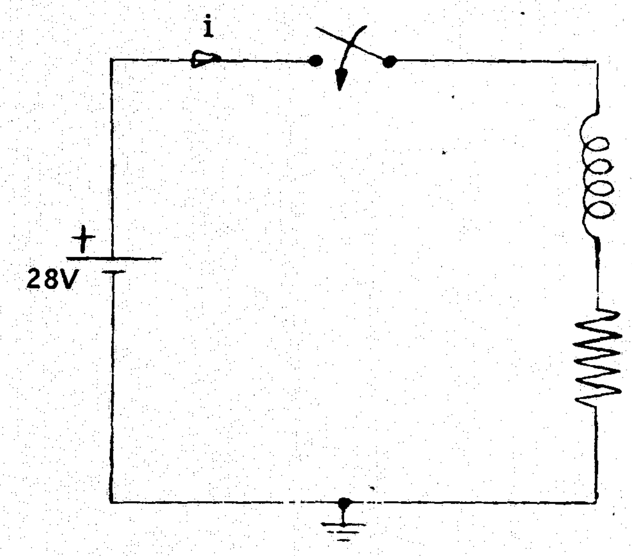


FIGURE 4.4.4–3A
STANDARD VALVE CIRCUIT

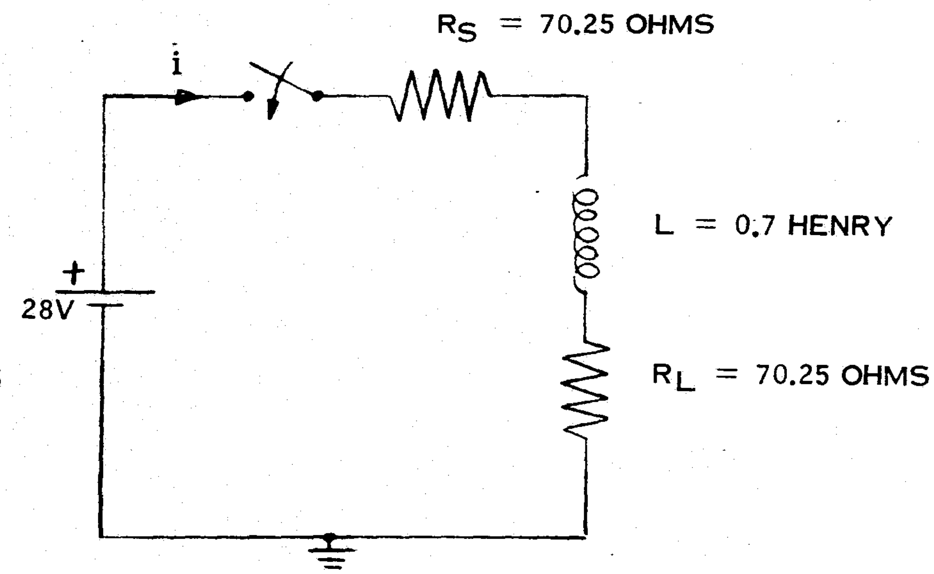


FIGURE 4.4.4–3B
PASSIVE RESISTIVE CONCEPT

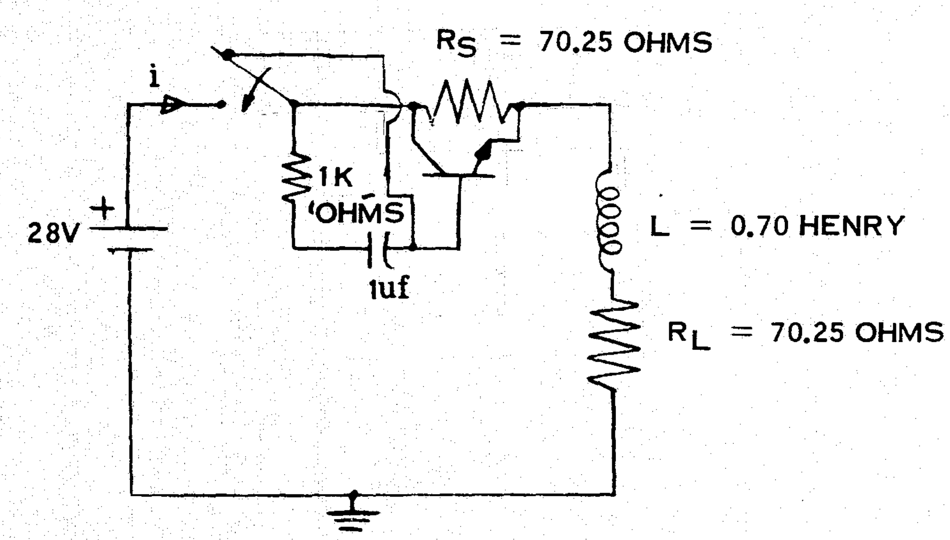


FIGURE 4.4.4–3C
ACTIVE TRANSISTOR CONCEPT

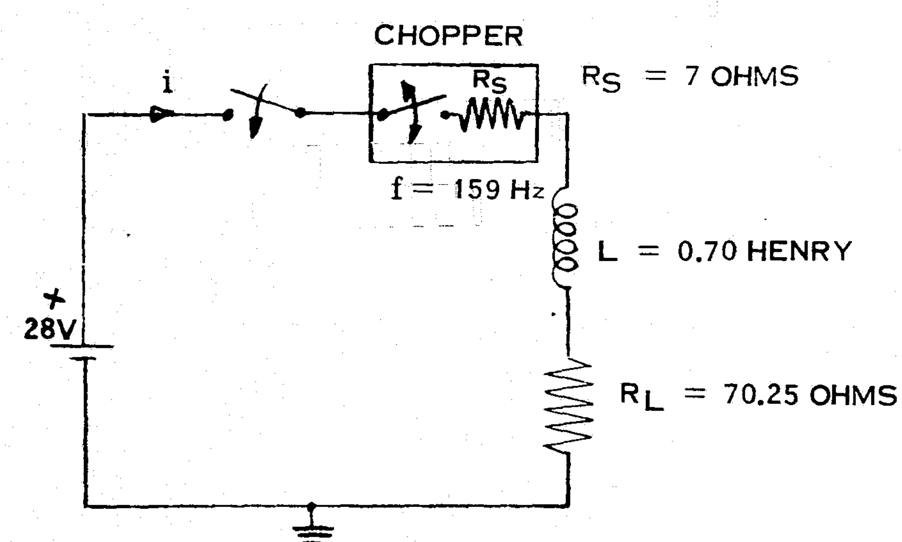


FIGURE 4.4.4–3D
CHOPPER CONCEPT

FIGURE 4.4.4–3. POWER CONDITIONING CIRCUITS

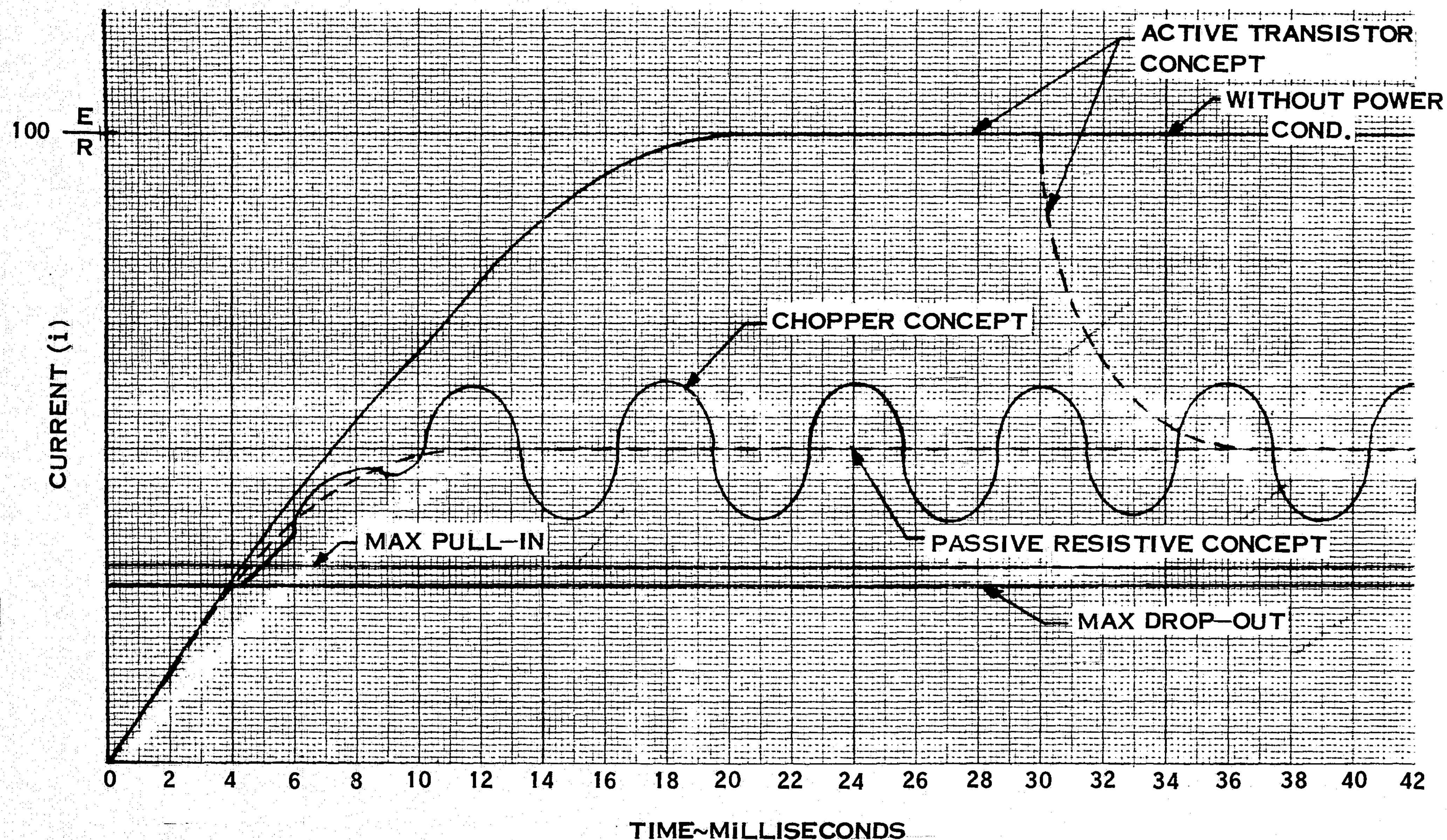


FIGURE 4.4.4-4. POWER CONDITIONING - CURRENT VS TIME

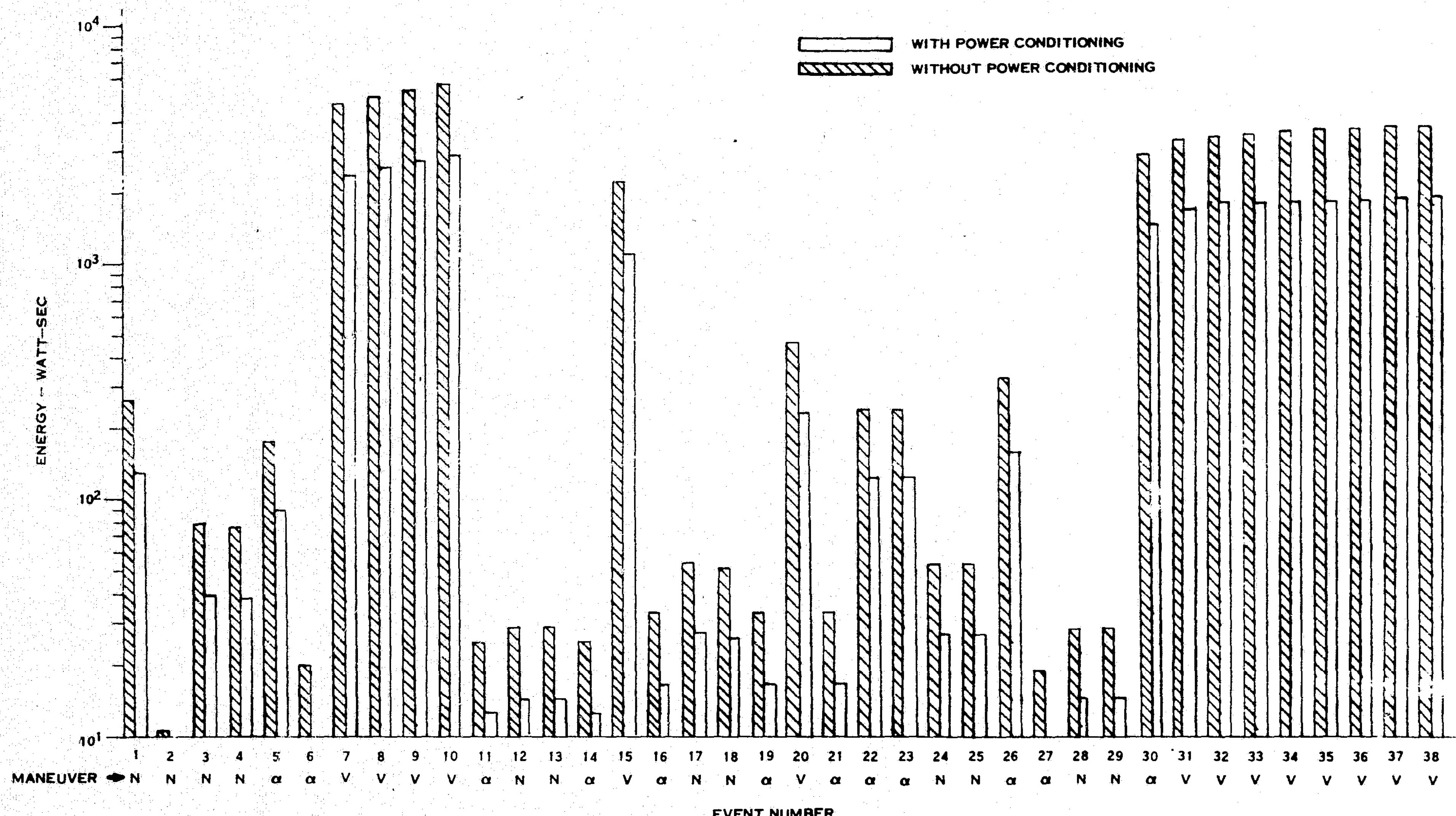


FIGURE 4.4.4–5. ELECTRICAL ENERGY EXPENDED VS EVENT–ORBITER MISSION (P5 CONFIGURATION – SYSTEMS I, II, & III)

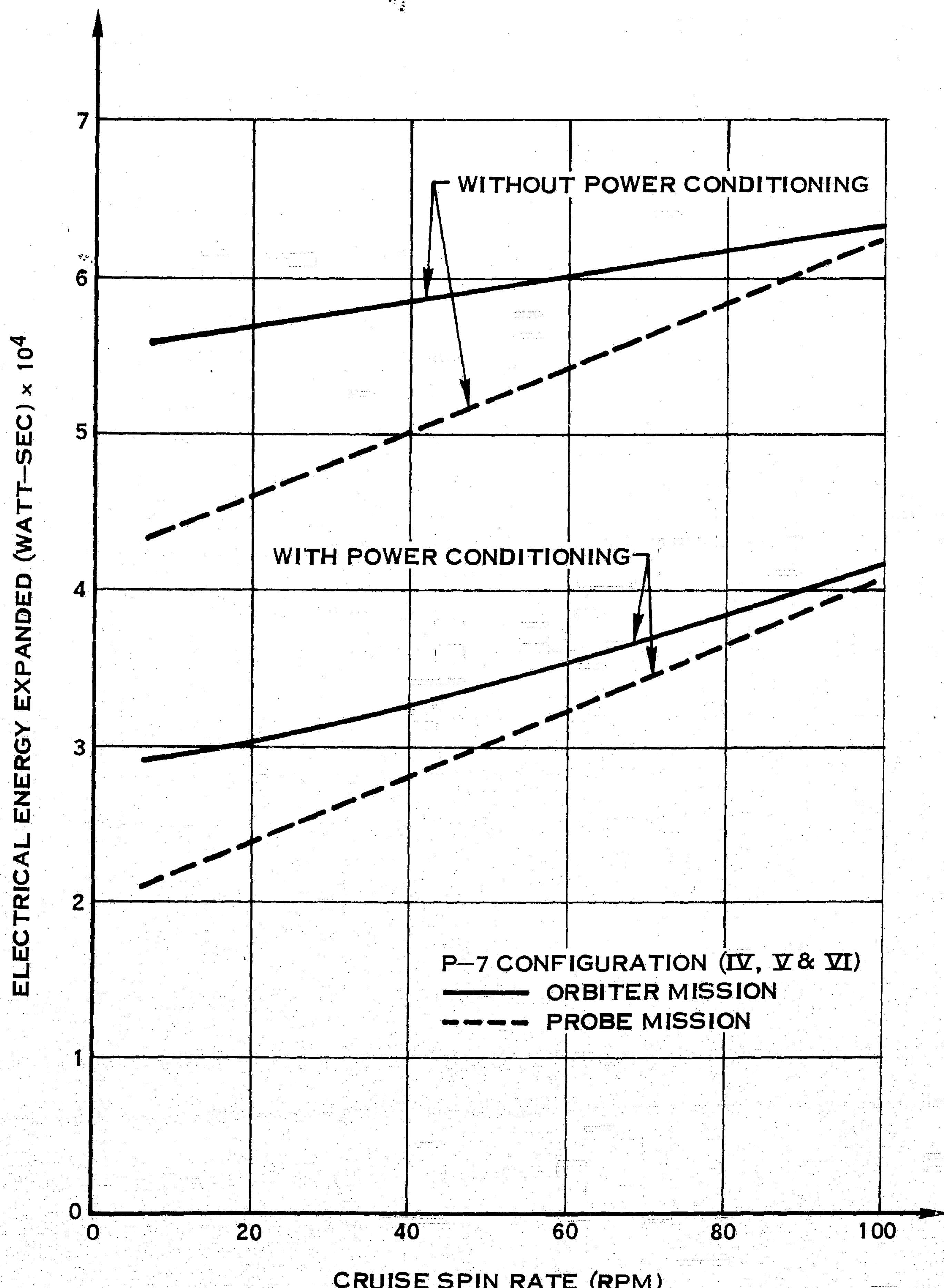


FIGURE 4.4.4–6. ELECTRICAL ENERGY EXPANDED VS CRUISE SPIN SPEED

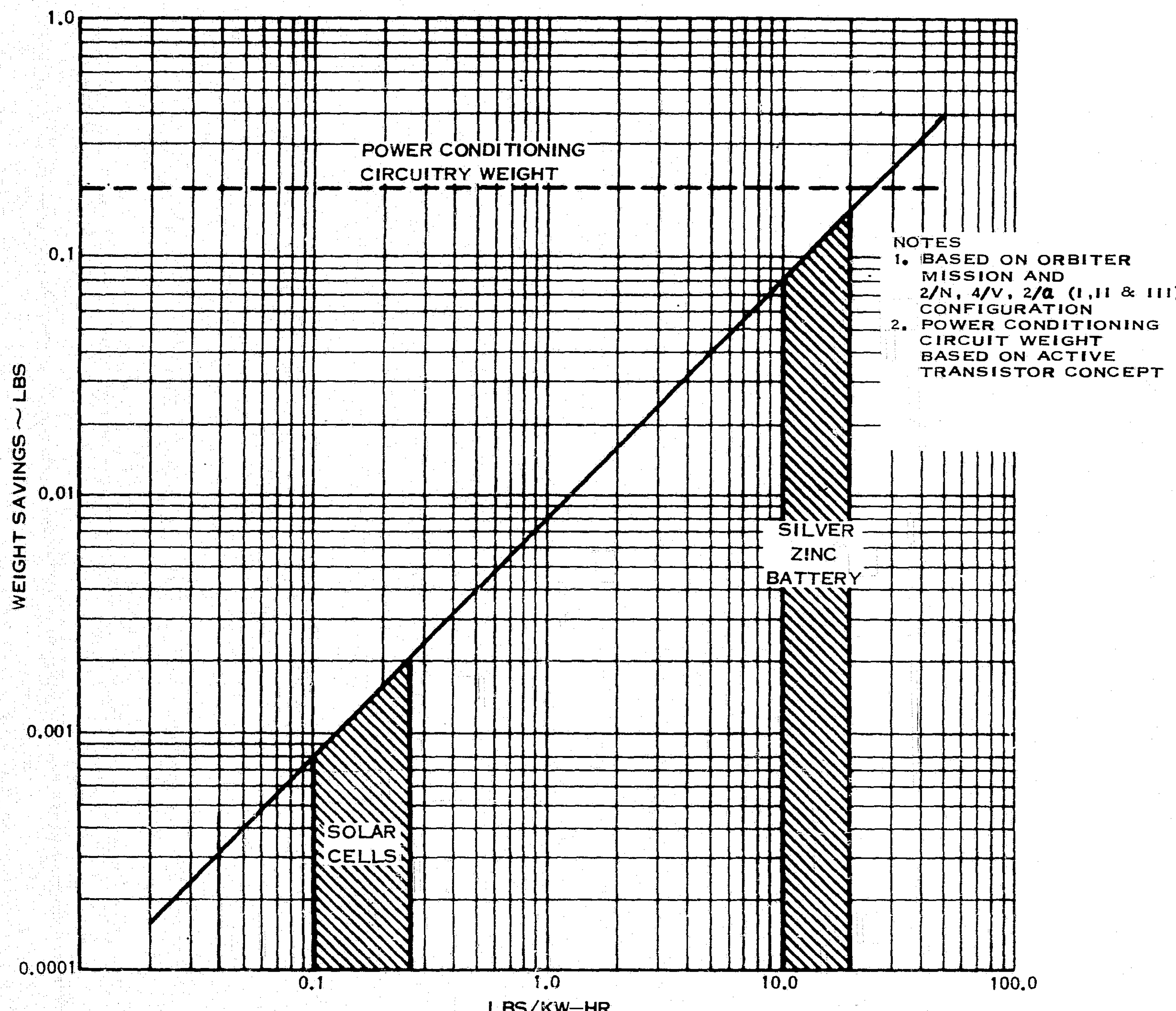


FIGURE 4.4.4–7. POWER CONDITIONING – WEIGHT SAVINGS VS WEIGHT/ELECTRICAL ENERGY

4.4.5 Cost

The candidate propulsion systems are ranked in order of their relative overall program cost in Tables 4.4.0-I and 4.4.0-II. The lowest cost systems are candidates IV and VI. Both of these candidates use engine placement concept P-7 (6 engine configuration), with candidate IV using feed system concept FS-2, and candidate VI using feed system concept FS-8.

The ranking presented in the referenced tables defines only the relative position of the candidate systems, without quantifying the differences between them in terms of dollars. To do this would have required a detailed definition of the type of program GSFC would want, as well as a much more detailed cost analysis. On the basis of cost histories of previous programs of a similar type, an estimate can be made that the cost spread between the lowest cost candidate system and the highest cost system will be approximately 8%. The rationale that went into the program cost ranking of the candidate systems included the following items, or judgments.

- The "types" of equipment required for the various candidate systems are almost identical, with the ordnance valve and the gimbal actuator the only items of equipment not common to all. As a result, the differences in equipment development/qualification costs can be considered insignificant.

- A judgment was made that lower program costs would be realized in designing the system to use the same single engine configuration as was originally qualified than it would be to cluster engines in modules. The rationale here was that even a low level design and development effort and a small development risk, associated with the thermal and dynamic aspects of a new structural package (engine module), will more than offset the cost advantages gained by packaging engines in modules.

4.4.6 Operations

Propulsion subsystem operational evaluations were divided into two major phases during the study to facilitate the engineering effort, and are presented herein in the same manner. Pre-Flight Operations are presented in Section 4.4.6.1, below, followed by a presentation of Flight Operations in Section 4.4.6.2.

4.4.6.1 PREFLIGHT OPERATIONS:

The scope and complexity of the preflight operations associated with each of the candidate subsystems are dependent upon the number of engines used in each of the systems, as well as the number of propellant feed lines, and the feed system components. These factors affect the complexity of each subsystem relative to installation and checkout procedures. All candidate subsystems utilize the same tankage arrangement and feed system to the component panel. The comparative differences exist at the component panel, and in the downstream portions of the subsystems. Candidate subsystems with the least number of components, particularly in the number of engines, propellant lines, and connections, are obviously the easiest to install into the spacecraft. Likewise, thrust alignment becomes an easier task where less engines are used.

Additional installation complexity exists where gimbal actuators are used to position the engines. Actuators give rise to complexity in the control circuits, flexible lines, and special electrical and mechanical interfaces which result in relatively complex installation, alignment and checkout procedures.

Subsystem checkout procedures, such as leakage testing, electrical checks, and proof pressure tests vary slightly between subsystems. Subsystems utilizing an ordnance valve require pressurization of two ports to perform leakage and proof pressure tests whereas a system with latching valves can be pressurized through one port with actuation of the latching valve to the open position. The servicing procedures (propellant conditioning, loading and unloading, vacuum drying, etc.) are also very similar for the candidate subsystems. However, those subsystems which require additional plumbing to accommodate the extra feed system components and engines, require extra time for draining the propellant and vacuum drying. Those systems utilizing an ordnance type propellant isolation have an advantage here in that they limit the portions of the system exposed to propellant.

In general, it may be concluded that the predominant factors which affect subsystem installation and checkout procedures are the number of engines used (and associated plumbing), and the presence of gimbal actuated engines. Although there are some varying checkout procedures with each of the feed system concepts, there are no significant advantages or disadvantages associated with any of them.

4.4.6.2 Flight Operations

Flight operations, presented in Tables 4.4.6.2-II, -III and -IV, describe the procedures for performing velocity change (ΔV), attitude control ($\Delta \alpha$) and spin control (ΔN) maneuvers, and the constraints imposed in performing these maneuvers. Information in these tables was prepared to provide a means for rating the various candidate systems in regard to flight operations, and for this reason, they are presented for only one mass configuration (cruise). However, where another condition would bring to light significant advantages or disadvantages, these conditions are also considered, and are noted. Table 4.4.6.2-I presents the Flight Operations rating of the various candidate systems.

In performing ΔV maneuvers, the major constraint is the spin perturbation, which is limited to .30 rpm. Because of the flight uncertainty (dr_{ENF}), there is an error in locating the spacecraft CG. Part of the radial component of the spacecraft's CG from where it is calculated to be (r_{EN}) results in a perturbation which necessitates dividing up ΔV maneuvers the size of the midcourse correction into many parts for candidate systems P-7, P-10, P-12 and P-13. The worst case is the Orbiter Mission, where the 108 meters/second midcourse correction would have to be done in 29 steps because of this perturbation. The P-5 candidate system is superior in this regard because of the location of its engines, and for this reason, is rated best with a Rank = I. In all systems, a calibration maneuver is required, and ΔV trim maneuvers are required when the maneuver ΔV is greater than 2. meters per second. In candidate systems using engine placement concepts, P-10, P-12 and P-13, no ΔV trim maneuver is needed for ΔV maneuvers less than 2 meters/second.

In performing $\Delta \alpha$ maneuvers with candidate systems using engine placement configurations P-5, P-12 and P-13, the major constraint is the spin perturbation which is limited to .30 rpm. In candidate systems using engine placement concepts P-7 and P-10, the spin perturbation is negligible because of the location of the $\Delta \alpha$ engines. This results in an ability to perform most $\Delta \alpha$ maneuvers in one step plus a $\Delta \alpha$ trim maneuver in order to attain the required accuracy of .20 degrees. A $\Delta \alpha$ trim maneuver is also required for candidate systems using engine placement concepts P-5, P-12 and P-13. The velocity perturbation is small compared to the size of the third midcourse correction of 2 meters/second. For all candidate systems, except those using concept P-7, it will be required to perform part of the final $\Delta \alpha$ trim maneuver with only one engine in order to meet the accuracy requirement of .20 degrees.

Performing ΔN maneuvers greater than 1.6 rpm will require a ΔN trim maneuver for all candidate systems in order to meet the accuracy requirements of ±.10 rpm. All maneuvers greater than 25 rpm will be required to be performed in three steps, including the trim maneuver, in order to meet the accuracy requirements. The velocity and precession perturbations for all the candidate systems are small; however, systems using concepts P-12 and P-13 do have precession perturbations

4.4.6.2 (continued)

considerably higher than the other systems and for this reason are rated last, with a Rank = III.

Flight Operations Rating Summary

Individual ratings for the three maneuvers are assigned the candidate systems in Table 4.4.6.2-I. An overall rating is also given on the table which shows system concept P-5 rated highest. No particular set of weighting factors was applied to draw this conclusion, but it was felt that the undesirable aspects of the other candidates with respect to spin perturbation errors during ΔV maneuvers was sufficient reason to rate the candidates in this manner.

TABLE 4.4.6.2-I. FLIGHT OPERATIONS RATINGS OF CANDIDATE SYSTEMS

Candidate System		ΔV Maneuvers	Δα Maneuvers	ΔN Maneuvers	Overall Rating
I, II & III	(P-5)	I	III	II	I
IV, V & VI	(P-7)	II	II	II	III
VII, VIII & IX	(P-10)	II	I	I	II
X, XI & XII	(P-12M)	II	III	III	IV
XIII, XIV & XV	(P-13)	II	III	III	IV

FOLDOUT FRAME 1

TABLE 4.4.6.2-II. FLIGHT OPERATIONS

System	Procedure	Maneuver Increment	Error
I, II & III (P-5)	Calibrate prior to first maneuver in order to determine $r_E\alpha$ and r_{EN}. Fire: 1, 3, 5, 7 or 2, 4, 6, 8 Modulate: 1,3 or 2,4	Probe: Max. ΔV = 95 m/sec Cal ΔV = 1 m/sec Orbiter: Max. ΔV = 42 m/sec Cal ΔV = 1 m/sec	Probe: dV = 5.7 m/sec $d\alpha$ = 2.4° dN = .30 RPM Orbiter: dV = 2.5 m/sec $d\alpha$ = 1.9° dN = .3 RPM
IV, V & VI (P-7)	Calibrate prior to first maneuver in order to determine $r_E\alpha$ and r_{EN}. Fire: 1, 3, 5 or 2, 4, 6 Modulate: 3,5 or 4,6	Probe: Max. ΔV = 8.6 m/sec. Cal ΔV = 1 m/sec Orbiter: Max. ΔV = 3.8 m/sec. Cal ΔV = 1 m/sec	Probe: dV = .46 m/sec $d\alpha$ = 1.35° dN = .30 RPM Orbiter: dV = .21 m/sec $d\alpha$ = 1.1° dN = .3 RPM
VII, VIII & IV (P-10)	Calibrate prior to first maneuver in order to determine $r_E\alpha$ and r_{EN}. Fire: 1 & 2 or 3 & 4 Modulate: 2 or 4	Probe: Max. ΔV = 8.6 m/sec Cal ΔV = 1 m/sec Orbiter: Max. ΔV = 3.8 m/sec Cal ΔV = 1 m/sec	Probe: dV = .39 m/sec $d\alpha$ = 1.35° dN = .30 RPM Orbiter: dV = .17 m/sec $d\alpha$ = 1.35° dN = .30 RPM
X, XI & XVI (P-12M)	Calibrate prior to first maneuver in order to determine $r_E\alpha$ and r_{EN}. Fire: 1 or 2 Track C, G with swivel	Probe: Max. ΔV = 8.6 m/sec Cal ΔV = 1 m/sec Orbiter: Max. ΔV = 3.8 m/sec Cal ΔV = 1 m/sec	Probe: dV = .31 m/sec $d\alpha$ = 1.35° dN = .30 RPM Orbiter: dV = .14 m/sec $d\alpha$ = 1.1° dN = .30 RPM
XIII, XIV & XV (P-13)	Calibrate prior to first maneuver in order to determine $r_E\alpha$ and r_{EN}. Fire: 1 & 2 or 3 & 4 Modulate: 2 or 4	Probe: Max. ΔV = 8.6 m/sec Cal ΔV = 1 m/sec Orbiter: Max. ΔV = 3.8 m/sec Cal ΔV = 1 m/sec	Probe: dV = .39 m/sec $d\alpha$ = 1.35° dN = .30 RPM Orbiter: dV = .17 m/sec $d\alpha$ = 1.10 dN = .30 RPM

NOTE: Spacecraft is in cruise mode unless otherwise noted.

ERATIONS - ΔV MANEUVERS

ror	Remarks	Rank
/sec PM /sec M	ΔV trim needed to reduce error $\leq$ 2 1/2% for $\Delta V >$ 2 m/sec and error $\leq$ 5% for $\Delta V <$ 2 m/sec. For probe ΔN trim needed if maneuver total $\Delta V >$ 95 m/sec. For orbiter ΔN trim needed if maneuver total $\Delta V >$ 42 m/sec.	I
/sec PM /sec M	ΔV trim needed to reduce error $\leq$ 2% for $\Delta V >$ 2 m/sec and error $\leq$ 5% for $\Delta V <$ 2 m/sec. For probe ΔN trim needed if maneuver total $\Delta V >$ 8.6 m/sec. For orbiter ΔN trim needed if maneuver total $\Delta V >$ 3.8 n/sec.	II
/sec PM /sec PM	ΔV trim needed to reduce error $\leq$ 2 1/2% for $\Delta V >$ 2 m/sec. For probe ΔN trim needed if maneuver total $\Delta V >$ 8.6 m/sec. For orbiter ΔN trim needed if maneuver total $\Delta V >$ 3.8 m/sec.	II
/sec PM /sec PM	ΔV trim needed to reduce error $\leq$ 2 1/2% for $\Delta V >$ 2 m/sec. For probe ΔN trim needed if maneuver total $\Delta V >$ 8.6 m/sec. For orbiter ΔN trim needed if maneuver total $\Delta V >$ 3.8 m/sec.	II
/sec PM sec M	ΔV trim needed to reduce error $\leq$ 2 1/2% for $\Delta V >$ 2 m/sec. For probe ΔN trim needed if maneuver total $\Delta V >$ 8.6 m/sec. For orbiter ΔN trim needed if maneuver total $\Delta V >$ 3.8 m/sec.	II

System	Procedure	Maneuver Increment	Error
I, II & III (P-5)	Fire: 1&6 or 3 & 8 or 2&5 or 4&7 $\Delta\alpha$ Trim maneuver required after all maneuvers in order to meet .20° accuracy	Probe: Max. $\Delta\alpha$=27.5° Max. $\Delta\alpha$=3.9° for N=85 RPM for events 27 & 28 Orbiter: Max. $\Delta\alpha$=22°	Probe: dv=.015 m/sec dα (Max.)=.096° f maneuvers≤2.1 dN=.30 RPM Orbiter: dv=.006 *dα (Max.)=.22° f maneuvers ≤4.8° dN=.30 RPM
IV, V & VI (P-7)	Fire: 1, 4, 6 or 2, 3, 5 $\Delta\alpha$ Trim maneuver required after all maneuvers in order to meet .20° accuracy Modulate: 4,6 or 3,5	Probe: Max. $\Delta\alpha$=71° (For events 27 and 28 also) Orbiter: Max. $\Delta\alpha$=71°	Probe: dv=.055 m/sec dα (Max.)=.066° fo maneuvers ≤ 1.25° dN=Negligible Orbiter: dv=.031 m/sec dα (Max.)=14° for maneuvers ≤ 2.6° dN=Negligible
VII, VIII & IX (P-10)	Fire: 1&4 or 2&3 $\Delta\alpha$ Trim maneuver required after all maneuvers in order to meet .20° accuracy	Probe: Max. $\Delta\alpha$=96° (For events 27 and 28 also) Orbiter: Max. $\Delta\alpha$=96°	Probe: dv=.05 m/sec dα (Max.)=.096° f maneuvers ≤ 2.1° dN=Negligible Orbiter: dv=.028 m/sec *dα (Max.)=.22° f maneuvers≤4.8° dN=Negligible
X, XI & XII (P-12)	Fire: 3&6 or 4&5 $\Delta\alpha$ Trim maneuver required after all maneuvers in order to meet .20° accuracy	Probe: Max. $\Delta\alpha$=27.5° Max. $\Delta\alpha$=3.9° for N=85 RPM for events 27&28 Orbiter: Max. $\Delta\alpha$=22°	Probe: dv=.015 m/sec dα (Max.)=.096° f maneuvers≤2.1° dN-.30 RPM Orbiter: dv=.0064 m/sec *dα (Max)=.22° for maneuvers≤4.8° dN=.30 RPM
XIII, XIV & IV (P-13)	Fire: 6&7 or 5&8 $\Delta\alpha$ Trim maneuver required after all maneuvers in order to meet .20° accuracy	Probe: Max. $\Delta\alpha$=27.6° Max. $\Delta\alpha$=3.9° for N=85 RPM for events 27&28 Orbiter: Max. $\Delta\alpha$=22°	Probe: dv=.015 m/sec dα (Max.=.096° for maneuvers ≤ 2.1° dN=.30 RPM Orbiter: dv=.0064 m/sec *dα (Max.)=.22° f maneuvers ≤4.8° dN= .30 RPM

Note: Spacecraft is in cruise mode unless otherwise noted.

ble 4.4.6.2-III

FOLDOUT FRAME 2

Error	Remarks	Rank
/sec =.096° for trim ≤2.1 M dv=.0064 m/sec)=.22° for trim ≤4.8° M	ΔN Trim maneuver needed if maneuver total > 27.5° for probe, and Δα>22° for orbiter (cruise condition). During events 27&28 in probe mission ΔN trim is needed for each Δα=3.9° increment because of high spin speed. *dα (Max.) is ½ this value for single engine operation.	III
/sec =.066° for trim ≤1.25° ible /sec =14° for trim ≤2.6° ible	Max. Δα increment based upon performing only one trim maneuver	II
/sec =.096° for trim ≤2.1° ible /sec)=.22° for trim ≤4.8° ible	Max. Δα increment based upon performing only one trim maneuver. *dα (Max.) is ½ this value for single engine operation.	I
/sec =.096° for trim ≤2.1° M n/sec =.22° for trim ≤4.8° M	ΔN Trim maneuver needed if maneuver total > 27.5° for probe, and Δα>22° for orbiter (cruise condition). During events 27&28 in probe mission N trim is needed for each Δα=3.9° increment because of high spin speed. * dα (Max.) is ½ this value for single engine operation.	III
/sec .096° for trim ≤2.1° M n/sec)=.22° for trim ≤4.8° M	ΔN Trim maneuver needed if maneuver total > 27.5° for probe, and Δα>22° for orbiter (cruise condition). During events 27 &28 in probe mission ΔN trim is needed for each Δα=3.9° increment because of high spin speed. * dα (Max.) is ½ this value for single engine operation	III

FOLDOUT FRAME

TABLE 4.4.6.2-IV

System	Procedure	Maneuver Increment	Error
I, II & III (P-5)	Fire: 2&3 or 6&7 or 5&8 or 1&4 ΔN trim $\leq$ 1.6 RPM needed after all maneuvers in order to meet ± .10 RPM accuracy	Probe: Max. ΔN = 25 RPM Orbiter: Max. ΔN = 25 RPM	Probe: dv (Max.) = .0011 m/sec $d\alpha$ (Max.) = .040° @ Nav. = 5 dN = 1.60 RPM Orbiter: dv (Max.) = .0015 m/sec $d\alpha$ (Max.) = .125° @ N = 15 R dN = 1.6 RPM
IV, V & VI (P-7)	Fire: 3&6 or 4&5 ΔN trim $\leq$ 1.6 RPM needed after all maneuvers in order to meet ± .10 RPM accuracy	Probe: Max. ΔN = 25 RPM Orbiter: Max. ΔN = 25 RPM	Probe: dv (Max.) = .0011 m/sec $d\alpha$ (Max.) = .040° @ Navg = 5 dN = 1.60 RPM Orbiter: dv (Max.) = .0015 m/sec $d\alpha$ (Max.) = .125° @ N = 15 R dN = 1.6 RPM
VII, VIII & IX (P-10)	Fire: 5&8 or 6&7 ΔN trim $\leq$ 1.6 RPM needed after all maneuvers in order to meet ± .10 RPM accuracy	Probe: Max. ΔN = 25 RPM Orbiter: Max. ΔN = 25 RPM	Probe: dv (Max) = .0011 m/sec $d\alpha$ (Max) = .017 @ N_{AVG} = 50 dN = 1.6 RPM Orbiter: dv (Max.) = .0015 m/sec $d\alpha$ (Max.) = .032° @ N = 15 RPM dN = 1.6 RPM
X, XI & XII (P-12)	Fire: 4&6 or 5&3 ΔN trim $\leq$ 1.6 RPM needed after all maneuvers in order to meet ± .10 RPM accuracy	Probe: Max. ΔN = 25 RPM Orbiter: Max. ΔN = 25 RPM	Probe: dv (Max.) = .045 m/sec $d\alpha$ (Max.) = .320° @ Navg = 50 RPM Orbiter: dv (Max.) = .060 m/sec $d\alpha$ (Max.) = .58° @ N = 15 RP dN = 1.6 RPM
XIII, XIV & XV (P-13)	Fire: 5&7 or 6&8 ΔN trim $\leq$ 1.6 RPM needed after all maneuvers in order to meet ± .10 RPM accuracy	Probe: Max. ΔN = 25 RPM Orbiter: Max. ΔN = 25 RPM	Probe: dv (Max.) = .045 m/sec $d\alpha$ (Max. = .31° @Navg = 50 RPM dN = 1.6 RPM Orbiter: dv (Max.) = .06 m/sec $d\alpha$ (Max.) = 5.8° $d\alpha$ (max.) 58° @ N = 15 RPM dN = 1.6 RPM

Note: Probe is in launch condition and orbiter in cruise mode for above errors.

FOLDOUT FRAME

Error	Remarks	Rank
=.0011 m/sec =.010° @ Navg.=50 RPM =.0015 m/sec =.125° @ N=15 RPM PM	Probe mission errors are for despin maneuver conditions	II
=.0011 m/sec =.040° @ Navg=50 RPM)=.0015 m/sec)=.125° @ N=15 RPM PM	Probe Mission errors are for despin maneuver conditions.	II
=.0011 m/sec =.317 @ N_{Avg}= 50 RPM PM)=.0015 m/sec)=.032° @ N.= PM	Probe mission errors are for despin maneuver conditions.	I
)=.045 m/sec)=.320° @ Navg=)=.060 m/sec)=.58° @ N=15 RPM PM	Probe mission errors are for despin maneuver conditions.	III
)=.045 m/sec =.31° @Navg= PM)=.06 m/sec)=5.8° dα (max.) = 15 RPM	Probe mission errors are for despin maneuver conditions.	III

4.4.7 Components

The degree of complexity of the candidate subsystems is a function of the types and quantities of components used in the subsystems and thus provides a basis for performing a technical evaluation and comparison. Table 4.4.7-I is a list of the types of components, and the quantities of each type, used for each subsystem; and a summary of the relative complexity of each of the subsystems. Because of the complexity (logic circuitry, position accuracy control, moving and sliding fits) associated with gimbal actuators for engine positioning, the systems utilizing this component are more complex than systems utilizing a fixed position engine. Likewise, the number of engines used in a system, and the use of latching valves represent more complexity, in terms of operation and checkout, than any of the other types of components (squib valves, test parts, etc.) that may be used on a candidate subsystem. Thus, using relative component complexity as a criteria, the subsystems are evaluated and categorized. The candidate subsystems utilizing gimbal actuators (X, XI, XII) represent the most complex configurations with candidate XI having the highest degree of complexity because, in addition to the actuators, the system also utilizes 4 latching solenoid valves. The systems having only 6 engines, and no gimbal actuator (IV, V, VI), represent the least complex of all the candidates with subsystem VI being the simplest since it utilizes a squib valve for isolation as opposed to the latching valves. The remaining candidates, representing the medium complexity range, use 8 engines and are rated in this category according to the type of isolation valve used. The subsystems utilizing squib valves (candidates III, IX, XV) are the simplest in this category with the remaining subsystems in this range considered more complex because of the use of latching valves.

TABLE 4.4.7-I

ORBITER SUBSYSTEM COMPONENT COMPLEXITY

Candidate Subsystem Component	QUANTITY OF COMPONENTS														
	I	II	III	IV	V	VI	VII	VIII	IX	X	XI	XII	XIII	XIV	XV
	P-5			P-7			P-10			P-12			P-13		
	FS-2	FS4	FS8	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8	FS-2	FS-4	FS-8
							(1)				(2)				
Tank	9	9	9	9	9	9	9	9	9	9	9	9	9	9	9
Engine	8	8	8	6	6	6	8	8	8	6	6	6	8	8	8
Fill & Drain	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2
Filter	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1
Transducer	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1
Squib Valve	-	-	1	-	-	1	-	-	1	-	-	1	-	-	1
Latching Valve	2	4	-	2	4		2	4	-	2	4	-	2	4	-
Actuator	-	-	-	-	-	-	-	-	-	2	2	2	-	-	-
Test Port	2	1	1	2	1	1	2	1	1	2	1	1	2	1	1
Relative Complexity Range	MEDIUM			LOWEST			MEDIUM			(2) HIGHEST			MEDIUM		

(1) Least complex of all candidate subsystems
(2) Most complex of all candidate subsystems

4.4.8. Structural/Physical Considerations

Since each of the candidate subsystems are similar except for the number of engine modules, and the number of components on the component panel, there are no significant structural or physical aspects that can be used to rank the candidate subsystems. The structural (vibration and shock) characteristics for each of systems vary slightly only because there are differences in engine locations and quantity as well as plumbing and feed system components. Subsystem interfaces, with the exception of plumbing and engine quantity and locations are also similar for all of the candidates. The component panel, although containing different components according to the feed system utilized, is designed to minimize spacecraft interfaces.

The subsystem using gimbal actuators to position engines, presents the most outstanding difference in installation and interface because of the added complexity associated with alignment and checkout. However, other considerations related to the candidate subsystems, such as thermal effects, plumbing effects, contamination control and center of mass tolerances, do not provide any significant physical advantages or disadvantages that can be used as a basis for ranking the subsystems. The inability to rank the candidate with respect to these considerations is due to the basic similarities present in the subsystems.

SECTION 5.0
PARAMETRIC AND DESIGN STUDIES

5.0 PARAMETRIC AND DESIGN STUDIES

During the course of the Study Program, a series of analytical and design studies was conducted. In some cases, these studies developed data that supported the effort concerned with establishing a comparative evaluation of the individual propulsion system candidates, but, in most cases, the information developed was general in nature, and applied to all candidates. In addition, some of the information was developed in parametric form so as to be useful to GSFC at some later date in their conduct of trade-offs at the spacecraft level. These studies are outlined below and are discussed in detail in the subsequent sections. Each of the studies is common to all candidate systems, with the exception of the section on "Flight Operations" (Section 5.4), which used candidate system I (P-5/FS-2) for both the Orbiter and Probe missions.

- Mission Analysis
- Reliability
- Pre-Flight Operations
- Flight Operations
- Components
- Test Plans
- Thermal Analysis
- Plume Study
- Leakage/Feed System Dynamics
- Environmental Effects
- Contamination Control
- CG Tolerances

5.1 Mission Analysis

To facilitate the determination of propellant weight allocations for the candidate propulsion systems under consideration and the two different missions, a computer program was written. This program uses an engine model to determine values of specific impulse and rotational efficiency for each maneuver. It also calculates weight of propellant consumed, the resultant tank pressure, the optimum electrical on time, the pulse train length, and updates the vehicle weight, spin inertia and spin rate. The computer program logic is illustrated in Figure 5.1.0-1.

The program requires the following inputs:

a) The number of tanks used, their diameter and initial pressure, the distance between the vehicle and tank center of mass, and the initial weight of propellant loaded.

b) The number of spin, velocity and attitude control engines, and their respective distances from the nozzle centerline to the vehicle center of mass.

c) The initial vehicle weight, initial vehicle spin inertia (including propellant) and the initial vehicle spin rate.

d) The engine minimum allowable on-time and the initial rotational efficiency for calculations.

e) The maneuver type and magnitude and any update on vehicle weight or spin inertia

For each maneuver, the program prints out the values of the parameters listed in Table 5.1.0-I.

In analyzing each system for the required propellant weight allocations, the program assumes that the engines used for each type of maneuver are equally distant from the vehicle center of mass, although different "types" of engines may be located at different distances. This permits the use of the same electrical on-time and pulse train length for all engines performing a particular maneuver.

Final tank pressures for each maneuver are computed assuming an isothermal blowdown. The thrust level used for the maneuver is that value of thrust corresponding to the pressure at the start of the maneuver.

There are certain maneuvers in each mission for which the number of engines, and the number of steps used to perform the maneuver, was changed. This was done for two reasons:

FOLDOUT FRAME 1

TABLE 5.1.0-I. COMPUTER PRINTOUT NOMENCLATURE

Units	Symbol	Nomenclature
lb-sec	IVEC	the value of the impulse required to perform the maneuver
lb-sec	IENG	the impulse each engine must deliver
lbf	F	the thrust level during the maneuver
	RATE	the rate at which the maneuver is performed using a single engine
	RES	the magnitude of the maneuver corresponding to the shortest on-time
	EF	the rotational efficiency of the maneuver
sec	ISP	the specific impulse for the maneuver
sec	TON	pulse electrical on time (optimum)
	PTL	pulse train length required for the maneuver
sec	TM	time required to perform the maneuver with one engine
lbs	DELM	the weight of propellant consumed during the maneuver
$lb\text{-}in^2$	IZZ	the vehicle spin moment of inertia after the maneuver
$lb\text{-}in^2$	DIZ	the change in vehicle spin moment
lbs	WTVEC	the vehicle weight after the maneuver
psia	PT	tank pressure after the maneuver
lbs	PROP	weight of propellant remaining in tanks after the maneuver
rpm	SPIN RATE	vehicle spin rate after maneuver

FOLDOUT FRAME 2

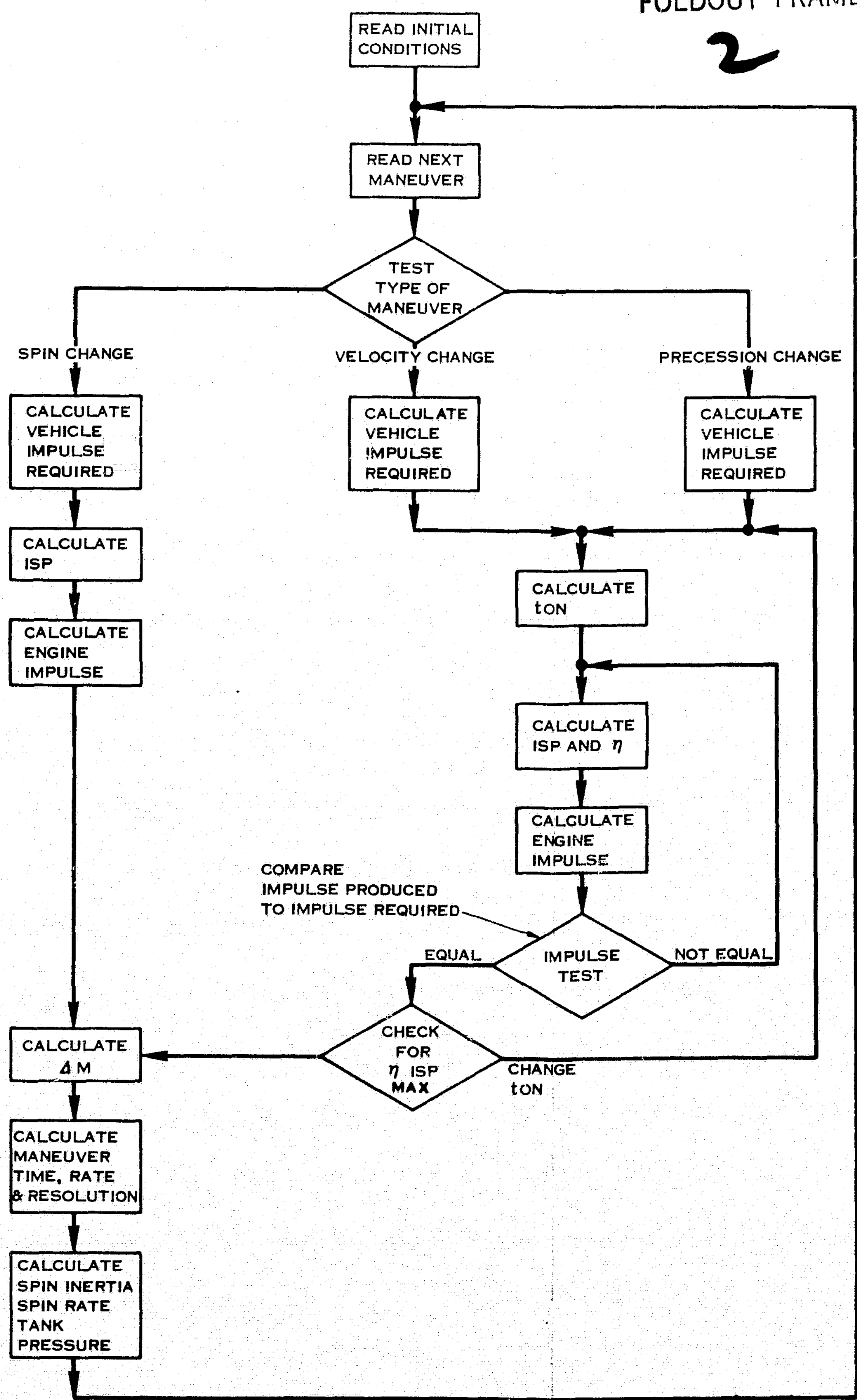


FIGURE 5.1.0-1. MISSION ANALYSIS – FLOW CHART

5.1 (continued)

1) To include the effect of tank pressure blow-down during a maneuver, and

2) To allow the coalescing of repeated maneuvers of smaller magnitude.

The logic used in making changes is that the fraction of the total impulse that each engine must deliver should be kept constant so that a large one-step maneuver utilizing four engines may be divided into four small maneuvers utilizing one engine so that one engine is still required to deliver only one-quarter of the total required impulse, or ten small maneuvers utilizing two engines each may be calculated effectively as one large maneuver using twenty engines.

The candidate systems can be represented by four different configurations based only on the number of each type of engine (Table 5.1.0-II). These four configurations were run for the two different missions, Orbiter and Probe. The results have been compiled and are presented in Tables 5.1.0-III through Table 5.1.0-X.

5.1.1 EFFECT OF CRUISE SPIN RATE ON PROPELLANT LOAD:

The variation in the total mission propellant requirements due to changes in the cruise spin rate may be determined by ratio-ing the total mass consumed by precession changes during cruise to the new cruise spin rates, and adding any additional spin-up mass consumption. This was done for the case of 48 rpm cruise spin rate and the value obtained was compared to a computer run for the same case. The difference between the computer run and the calculated value was less than .3% indicating that the propellant loading required for different spin rates could be confidently calculated and is shown here in Figure 5.1.1-1. The slope differences between the curve for the Probe mission and the one for the Orbiter mission is due to the difference in the number of precession maneuvers performed during cruise. The final cruise spin rate will have to be determined from an analysis of the maneuver errors.

5.1.2 EFFECT OF CONSTANT PULSE ELECTRICAL "ON-TIME" ON PROPELLANT LOADING:

The propellant loading for each mission and placement configuration was determined using the "optimum" electrical on-time for each maneuver; that is, the on-time for which the product of the rotational efficiency and specific impulse is maximum. In order to determine the impact of a fixed electrical on-time, the Probe mission was run using an initial propellant load of 74 lbs and fixed on-times which varied from 0.100 to 0.500 sec. The resultant propellant consumption was then plotted against on-time and the results are presented in Figure 5.1.2-1.

5.1.2 (continued)

This analysis shows that for the considered mission, with a fixed electrical on-time of approximately 0.310 sec, a propellant load of 64.2 lbs is required. This represents an increase of 1.8 lbs of propellant over a variable optimum on-time case. This margin may be reduced if the Probe mission sequence were altered to de-spin the vehicle after the release of the mini-probes. The high spin rate (85 rpm) results in an optimum on-time of approximately 0.100 sec, whereas the majority of the mission uses an on-time of between 0.500 and 0.600 sec, so that a spin-down after release should shift the minimum fixed on-time to a higher value, and the decreased on-time variance would result in a lower mission propellant load penalty for the fixed on-time case.

FOLDOUT FRAME

TABLE 5.1.0-III

ORBITER MISSION PROFILE

SYSTEMS I, II, III, VII, VIII, IX (Configurations P5 & P10)

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After ($lb\text{-}in^2$)	Tank Pressure After Event (psia)
1. Lift Off							
2. 3rd Stage Burnout					70		
3. De-Spin by Hydrazine System	-49 rpm	1	2	915.5	20.98	234833	248.5
4. Erect Booms							
5. Spin-Up to Cruise Value	+1 rpm	1	2	915.44	14.98	345415	248.4
6. S/C Separation from 3rd Stage							
7. Spin Control	+10 rpm	1	2	744.26	24.95	313102	247.88
8. Spin Control	-9.7 rpm	1	2	744.09	15.33	313046	247.40
9. Orient for Cruise	90 deg	1	2	743.71	15.32	312918	246.29
0. Orient for 1st M/C	10 deg	1	2	743.65	15.32	312899	246.13
1. 1st Midcourse Correction	27 m/sec		1	734.27	15.21	309799	222.27
	27 m/sec		1	724.96	15.09	306721	202.73
	27 m/sec		1	715.69	14.98	303664	186.43
	27 m/sec		1	706.50	14.87	300616	172.67
2. Re-Orient to Cruise	10 deg	1	2	706.44	14.87	300598	172.59
3. Spin Control	+2.5 rpm	1	2	706.38	17.37	300578	172.51
4. Spin Control	-2.5 rpm	1	2	706.33	14.87	300559	172.43
5. Orient for 2nd M/C	10 deg	1	2	706.27	14.87	300540	172.35
6. 2nd Midcourse Correction	10 m/sec	1	4	702.88	14.83	299424	167.79
7. Re-Orient to Cruise	10 deg	1	2	702.83	14.83	299406	167.72
8. Spin Control	5 rpm	1	2	702.73	19.83	299373	167.59
9. Spin Control	-4.7 rpm	1	2	702.63	15.12	299342	167.46
0. Orient for 3rd M/C	10 deg	1	2	702.58	15.12	299324	167.39
1. 3rd Midcourse Correction	2 m/sec	1	4	701.88	15.11	299086	166.49
2. Re-Orient to Cruise	10 deg	1	2	701.83	15.11	299067	166.42
3. Orient for Retrofire	90 deg	1	2	701.45	15.11	298940	165.93
4. Retrofire for Orbit Transfer	90 deg	1	2	424.08	15.10	284996	165.47
5. Spin Control	+ 5 rpm	1	2	423.99	20.09	284965	165.34
6. Spin Control	-5 rpm	1	2	423.89	15.09	284933	165.22
7. Attitude Control Maintenance	120 deg	1	2	423.41	15.09	284773	164.62
8. Orientation Trim	6 deg	1	2	423.38	15.09	284762	164.57
9. Spin Control	+2.5 rpm	1	2	423.32	17.59	284742	164.50
30. Spin Control	-2.5 rpm	1	2	423.26	15.08	284724	164.43
31. Orient for Periapsis Reduction	1050 deg	1	100	418.24	14.99	283058	158.34
32. Periapsis Reduction	24.3 m/sec	8	9	413.36	14.93	281433	152.84
	24.3 m/sec	8	9	408.52	14.86	279827	147.76
	24.3 m/sec	8	9	403.74	14.81	278221	143.05
	24.3 m/sec	8	9	399.00	14.75	276635	138.68
	24.3 m/sec	8	9	394.31	14.69	275061	134.60
	24.3 m/sec	8	9	389.66	14.63	273497	130.80
	24.3 m/sec	8	9	385.07	14.57	271948	127.24
	24.9 m/sec	8	9	380.40	14.51	270362	123.82

2

FOLDOUT FRAME

Initial Press. 250 psi
No. of Tanks 10
Tank Dia. .832
Initial Prop. 95 lbs

Tank Pressure After Event (psia)	Total Impulse Required (lbf-sec)	ENGINE Mode	Engine On-Time (sec)	Total No. of Pulses	Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lbf-sec)	Engine ISP (sec)	Propellant Used For Event (lbf)	Remaining Propellant (lbf)
248.5	120.3	SS	12.01	1	12.01	--	59.8	226.4	.53	94.47
248.4	4.94	SS	.495	1	.49	--	1.80	153.7	.03	94.44
247.88	35.59	SS	3.569	1	3.57	--	16.38	205.5	.17	94.26
247.40	34.49	SS	3.459	1	3.46	--	15.80	204.6	.17	94.09
246.29	82.77	P	.489	18	70.50	.951	43.73	223.3	.39	93.71
246.13	9.18	P	.306	3	11.75	.953	4.12	170.4	.06	93.65
222.27	2033	P	.428	1007	3751	.954	2133	227.3	9.38	84.27
202.73	2007	P	.432	1064	3981	.952	2112	226.3	9.32	74.96
186.43	1982	P	.497	979	3677	.948	2094	225.6	9.27	65.69
172.67	1956	P	.501	1023	3861	.946	2071	224.9	9.19	56.50
172.59	8.62	P	.378	3	11.84	.946	3.86	166.0	.06	56.44
172.51	9.94	SS	1.298	1	1.3	--	3.91	166.8	.06	56.38
172.43	9.94	SS	1.299	1	1.3	--	3.91	166.8	.06	56.33
172.35	8.62	P	.378	3	11.84	.946	3.86	166.0	.06	56.27
167.79	717.96	P	.504	99	375.8	.945	190.3	224.3	3.38	52.88
167.72	8.57	P	.379	4	12	.945	3.87	165.9	.06	52.83
167.59	18.25	SS	2.436	1	2.4	--	7.75	183.3	.10	52.73
167.46	17.4	SS	2.324	1	2.3	--	7.35	181.9	.096	52.63
167.39	8.74	P	.372	4	12.25	.945	4.06	166.9	.055	52.58
166.49	143.12	P	.496	21	76.6	.944	37.74	218.2	.695	51.88
166.42	8.73	P	.372	4	13.3	.945	4.06	166.9	.055	51.83
165.93	78.59	P	.496	22	83.4	.944	40.5	219.0	.380	51.45
165.47	74.90	P	.496	22	79.3	.944	39.03	218.5	.363	51.08
165.34	17.45	P	2.35	1	1.1	--	7.37	181.9	.096	50.99
165.22	17.45	P	2.35	1	1.1	--	7.37	181.9	.096	50.89
164.62	99.74	P	.497	29	105.9	.944	52.59	221.6	.477	50.41
164.57	5.08	P	.217	4	12.12	.926	2.20	153.4	.036	50.38
164.50	9.47	SS	1.282	1	1.28	--	3.70	165.3	.057	50.32
164.43	9.47	SS	1.283	1	1.28	--	3.70	165.3	.057	50.26
158.34	870.35	P	.435	6	21.2 ea	.945	7.94	183.8	5.025	45.24
152.84	1029.58	P	.500	67	255.2	.943	120.4	223.7	4.882	40.36
147.76	1017.55	P	.502	69	259.0	.942	120.08	223.4	4.835	35.52
143.05	1005.6	P	.504	69	262.6	.941	118	223	4.79	30.74
138.68	993.8	P	.506	70	266.1	.940	117	223	4.74	26.0
134.60	982.2	P	.509	70	269.2	.940	115	222.8	4.69	21.31
130.80	970.6	P	.511	71	272.2	.939	115	222.6	4.64	16.66
127.24	959.2	P	.513	72	274.8	.938	114	222.4	4.59	12.07
123.82	971.1	P	.515	73	284.2	.937	115	222.3	4.66	7.40

FOLDOUT FRAME

TABLE 5.1.0-IV
ORBITER MISSION PROFILE
SYSTEMS IV, V, & VI (Configuration P7)

				SATELLITE			
Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After ($lb-in^2$)	Tank Pressure After Event (psia)
1. Lift Off				WTVEC	Spin Rate 70	I_{zz}	P_T
2. 3rd Stage Burnout							
3. De-Spin by Hydrazine System	-40 rpm	1	2	915.47	20.98	234833	248.46
4. Erect Booms							
5. Spin-Up to Cruise Value	+ 1 rpm	1	2	915.44	14.98	345415	248.37
6. S/C Separation from 3rd Stage							
7. Spin Control	+10 rpm	1	2	744.26	24.98	313102	247.88
8. Spin Control	-9.7 rpm	1	2	744.09	15.33	313046	247.40
9. Orient for Cruise	90 deg	1	3	743.69	15.32	312915	246.26
10. Orient for 1st M/C	10 deg	1	3	843.63	15.32	312894	246.08
11. 1st Midcourse Correction	27 m/sec		1	734.26	15.21	309794	222.23
	27 m/sec	4	1	724.95	15.09	306716	202.70
	27 m/sec		1	715.67	14.98	303658	186.41
	27 m/sec		1	706.48	14.87	300610	172.64
12. Re-Orient to Cruise	10 deg	1	3	706.42	14.87	300590	172.56
13. Spin Control	+2.5 rpm	1	2	706.36	17.37	300571	172.48
14. Spin Control	-2.5 rpm	1	2	706.30	14.87	300552	172.40
15. Orient for 2nd M/C	10 deg	1	3	706.24	14.87	300532	172.31
16. 2nd Midcourse Correction	10 m/sec	1	3	702.86	14.83	299416	167.76
17. Re-Orient to Cruise	10 deg	1	3	702.80	14.83	299396	167.68
18. Spin Control	+5 rpm	1	2	702.70	19.82	299363	167.55
19. Spin Control	-4.7 rpm	1	2	702.60	15.12	299332	167.42
20. Orient for 3rd M/C	10 deg	1	3	702.54	15.12	299312	167.34
21. 3rd Midcourse Correction	2 m/sec	1	3	701.86	15.13	299077	166.46
22. Re-Orient to Cruise	10 deg	1	3	701.80	15.11	299057	166.38
23. Orient for Retrofire	90 deg	1	3	701.40	15.11	298925	165.87
24. Retro Fire for Orbit Transfer	90 deg	1	3	424.03	15.10	284978	165.39
25. Spin Control	+5 rpm	1	2	423.93	20.10	284946	165.27
26. Spin Control	-5 rpm	1	2	423.83	15.10	284914	165.15
27. Attitude Control Maintenance	120 deg	1	3	423.35	15.09	284752	164.54
28. Orientation Trim	6 deg	1	3	423.31	15.09	284739	164.49
29. Spin Control	+2.5 rpm	1	2	423.25	17.58	284720	164.42
30. Spin Control	-2.5 rpm	1	2	423.20	15.08	284702	164.34
31. Orient for Periapsis Reduction	1050 deg	1	150	417.85	14.99	282931	157.89
32. Periapsis Reduction	24.3 m/sec	8	6	412.98	14.92	281305	152.43
	24.3 m/sec	8	6	408.15	14.86	279703	147.38
33. Periapsis Reduction	24.3 m/sec		6	403.36	14.80	278099	142.70
34.	24.3 m/sec		6	398.63	70	276513	138.35
35.	24.3 m/sec		6	393.94	14.68	274939	134.30
36.	24.3 m/sec		6	398.30	14.62	273377	130.52
37.	24.3 m/sec		6	384.71	14.56	271832	126.98
38.	24.9 m/sec		6	380.06	14.50	270247	123.58

Initial Pressure	250 psia
Initial Propellant	95#
No. of Tanks	10
Tank Dia.	.823 ft.

n tia fter h^2)	Tank Pressure After Event (psia)	Total Impulse Required (lb_f-sec)	ENGINE			Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb_f-sec)	Engine ISP (sec)	Propellant Used For Event (lb_f)	Remaining Propellant (lb_f)
			Mode	Engine On-Time (sec)	Total No. of Pulses						
	P_T	IVEC		TON	PTL	TM	Eff	IENG	ISP	DELM	PROP
33	248.46	120.33	SS	12.01	1	12.01	--	59.82	226.4	.531	94.47
.5	248.37	4.94	SS	.495	1	.49	--	1.80	153.7	.032	94.44
02	247.88	35.59	SS	3.569	1	3.57	--	16.38	205.5	.173	94.26
46	247.40	34.45	SS	3.459	1	3.46	--	15.80	204.6	.168	94.09
.5	246.26	82.64	P	.459	12	46.91	.953	27.51	216.4	.401	93.69
24	246.08	9.23	P	.245	3	9.79	.947	2.32	157.9	.062	93.63
24	222.23	2033	P	.428	1007	3752	.954	2133.	227.3	9.375	84.26
6	202.70	2007	P	.432	1063	3982	.952	2110	226.3	9.316	74.94
8	186.41	1982	P	.497	979	3677	.948	2094	225.6	9.267	65.67
.0	172.64	1956	P	.501	1023	3861	.946	2072	224.9	9.192	56.48
20	172.56	8.79	P	.221	4	13.66	.928	2.71	157.9	.060	56.42
1	172.48	9.94	SS	1.299	1	1	--	3.91	166.8	.060	56.36
2	172.40	9.94	SS	1.299	1	1.3	--	3.91	166.8	.060	56.30
2	172.31	8.78	P	.221	4		.928	2.71	157.9	.060	56.24
6	167.76	717.9	P	.501	132		.945	253.7	224.3	3.388	52.86
6	167.68	8.74	P	.221	4	13.5	.927	2.71	157.7	.060	52.80
3	167.55	18.25	SS	2.437	1	2.44	--	7.75	183.3	.100	52.70
2	167.42	17.40	SS	2.324	1	2.32	--	7.35	181.9	.096	52.60
2	167.34	8.91	P	.217	5	14.0	.927	2.83	158.6	.061	52.54
7	166.46	143.11	P	.496	27	102	.944	50.26	221.3	.685	51.86
7	166.38	8.9	P	.217	5	14.1	.926	2.38	158.6	.061	51.80
5	165.87	78.71	P	.527	14	52.1	.942	26.64	212.4	.393	51.40
8	165.39	74.89	P	.496	14	52.8	.944	25.36	211.4	.375	51.03
6	165.27	17.45	SS	2.354	1	2.35	--	7.37	181.9	.096	50.93
4	165.15	17.45	SS	2.355	1	2.36	--	7.37	181.9	.096	50.83
2	164.54	99.73	P	.497	20	70.9	.944	35.1	217.0	.487	50.35
9	164.49	5.08	P	.217	3	8.01	.926	1.61	147.6	.037	50.31
0	164.42	9.47	SS	1.283	1	1.28	--	3.7	165.3	.057	50.25
2	164.34	9.47	SS	1.283	1	1.28	--	3.7	165.3	.057	50.20
1	157.89	870.82	P	.373	5	16.5 ea.	.944	5.2	172.6	5.342	44.85
5	152.43	1028.63	P	.500	101	383	.943	180.58	223.7	4.878	39.98
3	147.38	1016.61	P	.502	103	389	.942	180.09	223.5	4.830	35.15
9	142.70	1004.71	P	.505	104	394	.941	177.70	223.2	4.782	30.36
3	138.35	992.93	P	.507	105	399.5	.940	175.23	223.0	4.734	25.63
9	134.30	981.26	P	.509	106	404.2	.940	174.31	222.8	4.687	20.94
7	130.52	969.72	P	.511	106	408.6	.939	171.65	222.7	4.639	16.30
2	126.98	958.29	P	.513	108	412.7	.938	170.51	222.5	4.591	11.71
7	123.58	970.22	P	.515	110	426.6	.937	171.61	222.3	4.656	7.06

FOLDOUT FRAME 1

TABLE 5.1.0-V
ORBITER MISSION PROFILE
SYSTEMS X, XI, & XII (Configuration P-12M)

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After ($lb\text{-}in^2$)	Tank Press After Event (psia
1. Lift Off							
2. 3rd Stage Burnout							
3. De-Spin by Hydrazine System	-49 rpm	1	2	915.47	20.98	234833	248.4
4. Erect Booms							
5. Spin-Up to Cruise Value	+ 1 rpm	1	2	915.44	14.98	345415	248.3
6. S/C Separation from 3rd Stage							
7. Spin Control	+10 rpm	1	2	744.26	24.98	313102	247.8
8. Spin Control	-9.7 rpm	1	2	744.09	15.32	313046	247.4
9. Orient for Cruise	90 deg	1	2	743.71	15.32	312918	246.2
10. Orient for 1st M/C	10 deg	1	2	743.65	15.32	312900	246.1
11. 1st Midcourse Correction	27 m/sec		1	734.27	15.21	309799	222.2
	27 m/sec	4	1	724.96	15.09	306721	202.7
	27 m/sec		1	715.69	14.98	303664	186.4
	27 m/sec		1	706.50	14.87	300616	172.6
12. Re-orient to Cruise	10 deg	1	2	706.44	14.87	300598	172.5
13. Spin Control	+2.5 rpm	1	2	706.38	17.37	300578	172.5
14. Spin Control	-2.5 rpm	1	2	706.33	14.87	300559	172.4
15. Orient for 2nd M/C	10 deg	1	2	706.27	14.87	300540	172.3
16. 2nd Midcourse Correction	10 m/sec	1	1	702.88	14.83	299424	167.7
17. Re-orient to Cruise	10 deg	1	2	702.83	14.83	299406	167.7
18. Spin Control	+ 5 rpm	1	2	702.73	19.82	299374	167.5
19. Spin Control	+4.7 rpm	1	2	702.63	15.12	299342	167.4
20. Orient for 3rd M/C	10 deg	1	2	702.58	15.12	299324	167.3
21. 3rd Midcourse Correction	2 m/sec	1	1	701.90	15.11	299092	166.5
22. Re-orient to Cruise	10 deg	1	2	701.85	15.11	299074	166.4
23. Orient for Retrofire	90 deg	1	2	701.46	15.11	298946	165.9
24. Retrofire for Orbit Transfer	90 deg	1	2	424.10	15.10	285002	165.4
25. Spin Control	+ 5 rpm	1	2	424.01	20.10	284971	165.3
26. Spin Control	- 5 rpm	1	2	423.91	15.10	284939	165.2
27. Attitude Control Maintenance	120 deg	1	2	423.43	15.09	284780	164.6
28. Orientation Trim	6 deg	1	2	423.40	15.09	284768	164.6
29. Spin Control	+2.5 rpm	1	2	423.34	17.59	284749	164.5
30. Spin Control	-2.5 rpm	1	2	423.28	15.08	284730	164.4
31. Orient for Periapsis Reduction	1050 deg	1	100	418.26	14.99	283064	158.3
32. Periapsis Reduction	24.3 m/sec	8	2	413.38	14.93	281439	152.8
	24.3 m/sec		2	408.54	14.87	279834	147.7
	24.3 m/sec		2	403.76	14.81	278228	143.0
	24.3 m/sec		2	399.02	70	276641	138.7
	24.3 m/sec		2	394.33	14.75	275067	134.6
	24.3 m/sec		2	389.68	14.69	273504	130.8
	24.3 m/sec		2	385.09	14.63	271955	127.2
	24.9 m/sec		2	380.43	14.57	270369	123.8

FOLDOUT FRAME 2

SP 07R70 - F

Initial Pressure 250 psia
Initial Propellant 95 #
No. of Tanks 10
Tank Dia. .823

ELLITE			ENGINE								
ia ter 2)	Tank Pressure After Event (psia)	Total Impulse Required (lb_f-sec)	Mode	Engine On-Time (sec)	Total No. of Pulses	Event Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb_f-sec)	Engine ISP (sec)	Propellant Used For Event (lb_f)	Remaining Propellant (lb_f)
3	248.46	120.33	SS	12.01	1	12.01	-	59.82	226.4	.531	94.47
5	248.37	4.94	SS	.495	1	.49	-	1.80	153.7	.032	94.44
2	247.88	35.59	SS	3.569	1	3.57	-	16.38	205.5	.173	94.26
6	247.40	34.45	SS	3.459	1	3.46	-	15.80	204.6	.168	94.09
8	246.29	82.77	P	.489	18	65.9	.951	43.73	223.3	.390	93.71
0	246.13	9.18	P	.306	3	11.7	.953	4.12	170.4	.057	93.65
9	222.27	2033.08	P	.428	1006	3751.2	.954	2132.81	227.3	9.375	84.27
	202.73	2007.36	P	.432	1064	3981.5	.952	2112.04	226.3	9.316	74.96
	186.43	1981.80	P	.497	979	3676.9	.948	2093.73	225.6	9.267	65.69
	172.67	1956.41	P	.501	1023	3861.2	.946	2071.47	224.9	9.192	56.50
	172.59	8.62	P	.378	3	11.8	.946	3.86	166.0	.055	56.55
	172.51	9.94	SS	1.298	1	1.3	-	3.91	166.8	.060	56.38
	172.43	9.94	SS	1.299	1	1.3	-	3.91	166.8	.060	56.33
	172.35	8.62	P	.378	3	11.8	.946	3.86	166.0	.055	56.27
	167.79	717.96	P	.504	396	1502.6	.945	761.04	224.3	3.388	52.88
	167.72	8.57	P	.379	4	12.0	.945	3.87	165.9	.055	52.83
	167.59	18.[illegible]	SS	2.436	1	2.44	-	7.75	183.3	.100	52.73
	167.46	17.40	SS	2.324	1	2.32	-	7.35	181.9	.096	52.63
	167.39	8.74	P	.372	4	12.2	.945	4.06	166.9	.055	52.58
	166.51	143.12	P	.496	81	306.2	.944	150.41	224.1	.676	51.90
	166.44	8.73	P	.372	4	12.3	.945	4.06	166.9	.055	51.85
	165.95	78.60	P	.496	22	83.0	.944	40.49	219.0	.380	51.46
	165.49	74.90	P	.496	22	79.3	.944	39.03	218.5	.363	51.10
	165.37	17.45	SS	2.353	1	2.35	-	7.37	181.9	.096	51.01
	165.25	17.45	SS	2.354	1	2.35	-	7.37	181.9	.096	50.91
	164.64	99.75	P	.497	29	105.9	.944	52.59	221.6	.477	50.43
	164.60	5.08	P	.217	4	12.1	.926	2.20	153.4	.036	50.40
	164.52	9.47	SS	1.282	1	1.28	-	3.70	165.3	.057	50.34
	164.45	9.47	SS	1.283	1	1.28	-	3.70	165.3	.057	50.28
	158.36	870.38	P	.435	6	21.2 ea	.945	7.94	183.3	5.025	45.26
	152.86	1029.62	P	.500	76	1148.3	.943	135.46	223.7	4.882	40.38
	147.78	1017.59	P	.502	77	1165.6	.942	135.11	223.5	4.834	35.54
	143.07	1005.38	P	.504	78	1181.4	.941	132.90	223.2	4.786	30.76
	138.70	993.89	P	.506	79	1197.2	.940	132.32	223.0	4.739	26.02
	134.62	982.21	P	.509	79	1211.4	.940	129.98	222.8	4.691	21.33
	130.82	970.66	P	.511	80	1224.6	.939	129.21	222.7	4.643	16.68
	127.26	959.22	P	.513	80	1237.0	.938	126.77	222.5	4.596	12.09
	123.84	971.16	P	.515	83	1279.0	.937	129.57	222.3	4.660	7.43

FOLDOUT FRAME

TABLE 5.1.0-VI

ORBITER MISSION PROFILE

SYSTEMS VII, VIII, IX, XIII, XIV & XV (Configurations P-10 & P-13)

Ini
Ini
No.
Tan

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After $(lb\text{-}in^2)$	Tan Press After Event (psia
1. Lift Off							
2. 3rd Stage Burnout							
3. De-Spin by Hydrazine System	- 49 rpm	1	2	915.47	20.98	234833	248.4
4. Erect Booms							
5. Spin-Up to Cruse Value	+ 1 rpm	1	2	915.44	14.98	345415	248.3
6. S/C Separation from 3rd State							
7. Spin Control	+ 10 rpm	1	2	744.26	24.98	313102	247.8
8. Spin Control	-9.7 rpm	1	2	744.09	15.32	313046	247.4
9. Orient for Cruise	90 deg	1	2	743.71	15.32	312918	246.2
10. Orient for 1st M/C	10 deg	1	2	743.65	15.32	312900	246.1
11. 1st Midcourse Correction	27 m/sec		1	734.27	15.21	309799	222.2
	27 m/sec	4	1	724.96	15.09	306721	202.7
	27 m/sec		1	715.69	14.98	303664	186.4
	27 m/sec		1	706.50	14.87	300616	172.6
12. Re-orient to Cruise	10 deg	1	2	706.44	14.87	300598	172.5
13. Spin Control	+2.5 rpm	1	2	706.38	17.37	300578	172.5
14. Spin Control	-2.5 rpm	1	2	706.33	14.87	300559	172.4
15. Orient for 2nd M/C	10 deg	1	2	706.27	14.87	300540	172.3
16. 2nd Midcourse Correction	10 m/sec	1	2	702.88	14.83	299424	167.7
17. Re-Orient to Cruise	10 deg	1	2	702.83	14.83	299406	167.7
18. Spin Control	+ 5 rpm	1	2	702.73	19.83	299374	167.5
19. Spin Control	-4.7 rpm	1	2	702.63	15.12	299342	167.4
20. Orient for 3rd M/C	10 deg	1	2	702.58	15.12	299324	167.3
21. 3rd Midcourse Correction	2 m/sec	1	2	701.90	15.11	299091	166.5
22. Re-Orient to Cruise	10 deg	1	2	701.84	15.11	299073	166.4
23. Orient for Retrofire	90 deg	1	2	701.46	15.11	298945	165.9
24. Retrofire for Orbit Transfer	90 deg	1	2	424.10	15.10	285002	165.4
25. Spin Control	+5 rpm	1	2	424.00	20.10	284970	165.3
26. Spin Control	-5 rpm	1	2	423.91	15.10	284938	165.2
27. Attitude Control Maintenance	120 deg	1	2	423.43	15.09	284779	164.6
28. Orientation Trim	6 deg	1	2	423.40	15.09	284767	164.5
29. Spin Control	+2.5 rpm	1	2	423.34	17.59	284748	164.5
30. Spin Control	-2.5 rpm	1	2	423.28	15.08	284729	164.4
31. Orient for Periapsis Reduction	1050 deg	1	100	418.26	14.99	283063	158.3
32. Periapsis Reduction	24.3 m/sec	8	4	413.37	14.93	281438	152.8
	24.3 m/sec		4	408.54	14.87	279833	147.7
	24.3 m/sec		4	403.75	14.81	278227	143.0
	24.3 m/sec		4	399.02	14.75	276641	138.7
	24.3 m/sec		4	394.33	14.69	275068	134.6
	24.3 m/sec		4	389.68	14.63	273504	130.8
	24.3 m/sec		4	385.09	14.57	271955	127.2
	24.9 m/sec		4	380.43	14.51	270370	123.8

Initial Pressure 250 psia
Initial Propellant 95 #
No. of Tanks 10
Tank Dia. .823

Tank Pressure After Event (psia)	Total Impulse Required (lb_f-sec)	ENGINE Mode	ENGINE Engine On-Time (sec)	ENGINE No. of Pulses	Event Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb_f-sec)	Engine ISP (sec)	Propellant Used For Event (lb_f)	Remaining Propellant (lb_f)
248.46	120.33	SS	12.010	1	12.01	-	59.82	226.4	.531	94.47
248.37	4.94	SS	.495	1	.495	-	1.80	153.7	.032	94.44
247.88	35.59	SS	3.569	1	3.57	-	16.38	205.5	.173	94.26
247.40	34.45	SS	3.459	1	3.46	-	15.80	204.6	.168	94.07
246.29	82.77	P	.489	18	65.9	.951	43.73	223.3	.390	93.71
246.13	9.18	P	.306	3	11.7	.953	4.12	170.4	.057	93.65
222.27	2033.08	P	.428	1006	3751.2	.954	2132.81	227.3	9.375	84.27
202.73	2007.36	P	.432	1064	3981.5	.952	2112.04	226.3	9.316	74.96
186.43	1981.80	P	.497	979	3676.9	.948	2093.73	225.6	9.267	65.69
172.67	1956.41	P	.501	1023	3861.2	.946	2071.47	224.9	9.192	56.50
172.59	8.62	P	.378	3	11.8	.946	3.86	166.0	.055	56.44
172.51	9.94	SS	1.298	1	1.3	-	3.91	166.8	.060	56.38
172.43	9.94	SS	1.299	1	1.3	-	3.91	166.8	.060	56.33
172.35	8.62	P	.378	3	11.8	.946	3.86	166.0	.055	56.27
167.79	717.96	P	.504	198	751.2	.945	380.52	224.3	3.388	52.88
167.72	8.57	P	.379	4	12.0	.945	3.87	165.9	.055	52.83
167.59	18.25	SS	2.436	1	2.44	-	7.75	183.3	.100	52.73
167.46	17.40	SS	2.324	1	2.32	-	7.35	181.9	.096	52.63
167.39	8.74	P	.372	4	12.2	.945	4.06	166.9	.055	52.58
166.51	143.12	P	.496	41	153.0	.944	75.04	223.5	.678	51.90
166.44	8.73	P	.372	4	12.3	.945	4.06	166.9	.055	51.84
165.95	78.60	P	.496	22	83.0	.944	40.49	219.0	.380	51.46
165.49	74.90	P	.496	22	79.3	.944	39.03	218.5	.363	51.10
165.37	17.45	SS	2.353	1	2.35	-	7.37	181.9	.096	51.00
165.24	17.45	SS	2.354	1	2.35	-	7.37	181.9	.096	50.91
164.64	99.75	P	.497	29	105.9	.944	52.59	221.6	.477	50.43
164.59	5.08	P	.217	4	12.1	.926	2.20	153.4	.036	50.40
164.52	9.47	SS	1.282	1	1.28	-	3.70	165.3	.057	50.34
164.45	9.47	SS	1.283	1	1.28	-	3.70	165.3	.057	50.28
158.36	870.37	P	.435	6	21.2 ea	.945	7.94	183.3	5.025	45.26
152.86	1029.62	P	.500	152	574.1	.943	272.73	223.7	4.881	40.37
147.78	1017.59	P	.502	154	582.8	.942	270.23	223.5	4.834	35.54
143.07	1005.68	P	.504	156	590.9	.941	267.53	223.3	4.786	30.75
138.70	933.89	P	.506	158	598.6	.940	264.65	223.0	4.738	26.02
134.62	982.21	P	.509	159	605.7	.940	261.62	222.9	4.691	21.33
130.82	970.66	P	.511	160	612.3	.939	258.44	222.7	4.643	16.68
127.26	959.22	P	.513	161	618.6	.938	255.15	222.5	4.595	12.09
123.84	971.16	P	.515	166	639.5	.937	259.16	222.4	4.660	7.43

FOLDOUT FRAME 1

TABLE 5.1.0-VII
PROBE MISSION PROFILE
SYSTEMS I, II, III, VII, VIII, & IX (Configurations P-5 & P-10)

Initial
No. of
Tank Di
Initial

	Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After (lb-in^2)	Tank Pressure After Event (psia)
1.	Lift off							
2.	3rd Stage Burnout					70		
3.	De-spin - Hydrazine	-40 rpm	1	2	1051.4	29.98	335796	245.61
4.	Erect Booms					8.0		
5.	Spin Up to Cruise Valve	+ 4 rpm	1	2	1051.2	11.98	898292	244.27
6.	S/C Separation from 3rd Stage							
7.	Orient for Cruise	90 deg	1	2	879.4	11.97	833513	238.9
8.	Orient for 1st M/C	10 deg	1	2	879.3	11.97	833478	238.2
9.	1st Midcourse Correction	27 m/sec		1	868.2	11.94	829802	182.3
	Total 108 m/sec	27 m/sec	4	1	857.0	11.89	826121	147.6
		27 m/sec		1	845.8	11.16	822437	123.9
		27 m/sec		1	834.7	11.12	818748	106.9
10.	Re-orient to Cruise	50 deg	1	2	834.3	11.82	818594	106.3
11.	Orient for 2nd M/C	20 deg	1	2	834.0	11.82	818528	106.1
12.	2nd Midcourse Correction	10 m/sec	1	4	829.9	11.80	817159	100.9
13.	Re-orient to Cruise	40 deg	1	2	829.6	11.79	817034	100.5
14.	Orient for 3rd M/C	20 deg	1	2	829.4	11.79	816968	100.3
15.	3rd Midcourse Correction	2 m/sec	1	4	828.5	11.79	816691	99.3
16.	Attitude Control Maintenance	20 deg	1	2	828.4	11.79	816625	99.1
17.	Orient Spin Axis Parallel to Ecliptic	90 deg	1	2	827.5	11.79	816344	98.2
18.	Re-target Maxi Probe	40 deg	1	2	827.1	11.78	816219	97.7
19.	Separate Maxi Probe							
20.	Attitude Control Maintenance	10 deg	1	2	427.0	11.78	765495	97.6
21.	Re-target Mini Probe	5 m/sec	1	4	425.9	11.78	765139	96.5
22.	Attitude Control Maintenance	20 deg	1	2	425.8	11.78	765075	96.3
23.	Spin Up Mini Probes	+73 rpm	1	2	423.2	84.68	764195	93.5
24.	Separate Mini Probe							
25.	Attitude Control Maintenance	20 deg	1	2	193.82	84.63	201920	92.4
26.	Re-target S/C Bus	18 m/sec	1	4	192.1	84.58	201333	90.7
27.	Correct Sun Angle Drift	20 deg	1	2	191.7	84.54	201221	89.6
28.	Orient Spin to Velocity Vector	12 deg	1	2	191.5	84.51	201152	88.9

Initial Press. 250 psia
No. of Tanks 6
Tank Dia. .823 ft.
Initial Drop 74 lbs.

TELLITE			ENGINE								
...n ...tia ...fter ...n²)	Tank Pressure After Event (psia)	Total Impulse Required (lbf-sec)	Mode	Engine On-Time (sec)	Total No. of Pulses	Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lbf-sec)	Engine ISP (sec)	Propellant Used For Event (lbf)	Remaining Propellant (lbf)
...96	245.61	140.4	SS	14.02	1	14		70	227.1	.62	73.38
...92	244.27	40.3	SS	4.08	1	4		19	208.7	.19	73.19
...3	238.9	172.1	P	.587	31	148	.953	89	227	.79	72.39
...78	238.2	19.1	P	.509	4	19.5	.954	8.3	187.4	.11	72.29
...02	182.3	2404	P	.587	890	4236	.952	2526	227	11.11	61.17
...1	147.6	2373	P	.589	1071	5079	.947	2509	225	11.15	50.02
...7	123.9	2343	P	.670	1099	5187	.940	2498	223	11.16	38.85
...8	106.9	2312	P	.672	1242	5846	.936	2478	222	11.12	27.74
...4	106.3	94.72	P	.674	28	130	.932	49.7	219	.47	27.27
...8	106.1	37.8	P	.595	13	59	.932	18.9	203	.20	27.07
...9	100.9	847.8	P	.674	128	604	.931	227	222	4.11	22.96
...4	100.5	75.7	P	.675	24	108.5	.930	40.6	216	.38	22.58
...8	100.3	37.8	P	.596	14	61.5	.930	19.5	203	.20	22.38
...1	99.3	168.9	P	.755	24	112	.927	45.8	218	.84	21.55
...5	99.1	37.8	P	.596	14	62	.930	19.6	203	.20	21.35
...4	98.2	170.1	P	.676	54	247	.929	90.4	221	.83	20.52
...9	97.7	75.6	P	.676	24	110.5	.929	39.5	216	.38	20.14
...5	97.6	17.7	P	.517	8	34	.928	8.4	182	.11	20.04
...9	96.5	217.3	P	.756	31	147.5	.926	57.4	219	1.10	18.97
...5	96.3	35.4	P	.636	13	55.5	.929	18.4	202	.19	18.78
...5	93.5	583.6	SS	118.8	1	119	.929	291.8	222	2.63	16.15
...0	92.4	67.45	P	.094	152.7	108.2	.924	36.6	221	.339	15.81
...3	90.7	352.72	P	.105	385	273.1	.921	96	221	1.73	14.08
...1	89.6	67.25	P	.105	154.7	109.7	.920	36.66	221	.337	13.74
...2	88.9	40.26	P	.105	92.9	65.9	.920	21.9	220	.211	13.53

TABLE 5.1.0-VIII

PROBE MISSION PROFILE

SYSTEMS IV, V & VI (Configuration P-7)

Initi
No.
Tank
Initi

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After $(lb\text{-}in^2)$	Tank Pressure After Event (psia)	T In Req (lb
1. Lift Off								
2. 3rd Stage Burnout					70			
3. De-spin - Hydrazine	-40 rpm	1	2	1051.4	29.98	335796	246.15	1
4. Erect Booms								
5. Spin-Up to Cruise Valve	+ 4 rpm	1	2	1051.2	11.98	898291	244.9	
6. S/C Separation from 3rd Stage								
7. Orient for Cruise	90 deg	1	3	879.4	11.98	833516	240.2	1
8. Orient for 1st M/C	10 deg	1	3	879.3	11.98	833479	239.56	
9. 1st Midcourse Correction	27 m/sec	4	1	868.2	11.94	829811	188.8	24
Total 108 m/sec	27 m/sec		1	857.0	11.90	826132	155.7	23
	27 m/sec		1	845.9	11.86	822451	132.5	23
	27 m/sec		1	834.8	11.82	818771	115.4	23
10. Re-orient to Cruise	50 deg	1	3	834.4	11.82	818613	114.7	
11. Orient for 2nd M/C	20 deg	1	3	834.1	11.82	818543	114.5	
12. 2nd Midcourse Correction	10 m/sec	1	3	830.0	11.80	817176	109.3	8
13. Re-orient to Cruise	40 deg	1	3	829.6	11.80	817048	108.8	
14. Orient for 3rd M/C	20 deg	1	3	829.4	11.80	816979	108.6	
15. 3rd Midcourse Correction	2 m/sec	1	3	828.6	11.80	816707	107.6	1
16. Attitude Control Maintenance	20 deg	1	3	828.4	11.79	816629	107.4	
17. Orient Spin Axis Parallel to Exliptic	90 deg	1	3	827.6	11.79	816351	106.4	1
18. Re-target Maxi Probe	40 deg	1	3	827.2	11.79	816222	105.9	
19. Separate Maxi Probe								
20. Attitude Control Maintenance	10 deg	1	3	427.1	11.79	765497	105.8	
21. Re-target Mini Probe	5 m/sec	1	3	426.0	11.78	765146	104.7	2
22. Attitude Control Maintenance	20 deg	1	3	425.8	11.78	765080	104.4	
23. Spin Up Mini Probes	+73 rpm	1	2	423.2	84.69	764196	101.6	5
24. Separate Mini Probes								
25. Attitude Control Maintenance	20 deg	1	3	193.9	84.64	201957	100.4	
26. Re-target S/C Bus	18 m/sec	1	3	192.2	84.59	201370	98.7	3
27. Correct Sun Angle drift	20 deg	1	3	191.9	84.54	201258	97.6	
28. Orient Spin to Velocity Vector	12 deg	1	3	191.7	84.52	201189	96.9	

Initial Press. 250 psia
No. of Tanks 6 tanks
Tank Dia. .832 ft.
Initial Prop 69 lbs.

E		ENGINE								
Tank ressure After Event (psia)	Total Impulse Required (lb_f-sec)	Mode	Engine On-Time (sec)	Total No. of Pulses	Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb_f-sec)	Engine ISP (sec)	Propellent Used For Event (lb_f)	Remaining Propellant (lb_f)
246.15	140.4	SS	14.02	1	14.02	-	70.0	227.1	.62	68.38
244.9	40.3	SS	4.07	1	4.07	-	18.8	208.7	.19	68.19
240.2	172	P	.587	20	98.3	.953	58.2	225.6	.80	67.40
239.56	19.1	P	.431	3	15	.953	5.96	179	.11	67.28
188.8	2403.8	P	.587	887	4219	.952	2527	227	11.11	56.16
155.7	2373	P	.589	1043	4950	.948	2507	225	11.13	45.03
132.5	2343	P	.670	1054	4981	.941	2494	224	11.13	33.90
115.4	2312	P	.672	1178	5558	.937	2473	223	11.08	22.82
114.7	94.8	P	.754	16	73.2	.931	33.04	214	.48	22.4
114.5	37.8	P	.674	7	32.9	.933	11.99	192	.21	22.1
109.3	848	P	.674	160	759.7	.934	302.2	222	4.10	18.04
108.8	75.4	P	.635	16	72.3	.933	25.71	209	.39	17.65
108.6	37.7	P	.675	8	34	.932	12.3	192	.21	17.44
107.6	168.9	P	.675	34	157.7	.932	60.4	220	.82	16.62
107.4	37.7	P	.676	8	34.3	.932	12.4	192	.21	16.41
106.4	169.7	P	.676	33	155	.932	59.3	220	.83	15.58
105.9	75.3	P	.596	17	78.3	.932	25.6	209	.39	15.20
105.8	17.7	P	.517	5	21.3	.931	5.5	171	.11	15.08
104.7	217.3	P	.676	43	206.7	.931	76.4	221	1.06	14.03
104.4	35.3	P	.676	7	33	.931	11.2	190	.20	13.83
101.6	583.6	SS	111.7	1	112	-	291.8	222	2.62	11.20
100.4	67.4	P	.094	101.3	71.8	.926	24.8	221	.339	10.86
98.7	352.5	P	.094	534	378	.926	127.2	221	1.72	9.14
97.6	67.13	P	.094	102.2	72.5	.925	24.2	221	.337	8.80
96.9	40.2	P	.094	61.4	43.6	.925	14.48	218	.211	8.59

FOLDOUT FRAME

TABLE 5.1.0-IX
PROBE MISSION PROFILE
SYSTEMS X, XI & XII (Configurations P-12M)

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After ($lb\text{-}in^2$)	Tank Pressu After Event (psia
1. Lift Off							
2. 3rd Stage Burnout					70		
3. De-spin - Hydrazine	-40 rpm	1	2	1051.4	29.98	335796	246.2
4. Erect Booms							
5. Spin Up to Cruise Valve	+ 4 rpm	1	2	1051.2	11.98	898291	245.0
6. S/C Separation from 3rd Stage							
7. Orient for Cruise	90 deg	1	2	879.4	11.98	833518	240.2
8. Orient for 1st M/C	10 deg	1	2	879.3	11.98	833483	239.6
9. 1st Midcourse Correction	27 m/sec	4	1	868.2	11.94	829815	188.8
Total 108 M/SEC	27 m/sec		1	857.0	11.90	826136	155.7
	27 m/sec		1	846.0	11.86	822454	132.5
	27 m/sec		1	834.8	11.82	818774	115.4
10. Re-orient to Cruise	50 deg	1	2	834.4	11.82	818621	114.8
11. Orient for 2nd M/C	20 deg	1	2	834.2	11.82	818555	114.5
12. 2nd Midcourse Correction	10 m/sec	1	2	830.1	11.80	817188	109.3
13. Re-orient to Cruise	40 deg	1	2	829.7	11.80	817065	108.9
14. Orient for 3rd M/C	20 deg	1	2	829.5	11.80	816998	108.6
15. 3rd Midcourse Correction	2 m/sec	1	1	828.7	11.80	816728	107.7
16. Attitude Control Maintenance	20 deg	1	2	828.5	11.80	816655	107.4
17. Orient Spin Axis Parallel to Ecliptic	90 deg	1	2	827.7	11.79	816378	106.5
18. Retarget Maxi Probe	40 deg	1	2	827.3	11.79	816253	106.1
19. Separate Maxi Probe							
20. Attitude Control Maintenance	10 deg	1	2	427.2	11.79	765530	105.9
21. Retarget Mini Probe	5 m/sec	1	2	426.1	11.79	765181	104.8
22. Attitude Control Maintenance	20 deg	1	2	425.9	11.79	765119	104.6
23. Spin Up Mini Probes	+73 rpm	1	2	423.3	84.69	764234	101.7
24. Separate Mini Probes							
25. Attitude Control Maintenance	20 deg	1	2	194	84.64	201957	100.5
26. Retarget S/C Bus	18 m/sec	1	2	192.2	84.59	201370	98.8
27. Correct Sun Angle Drift	20 deg	1	2	191.9	84.55	201258	97.7
28. Orient Spin to Velocity Vector	12 deg	1	2	191.7	84.52	201189	97.0

Initial Press. 250 psia
No. of Tanks 6
Tank Dia. .823 ft.
Initial Prop. 69 lbs.

n tia fter n^2)	Tank Pressure After Event (psia)	ENGINE Total Impulse Required (lb$_f$-sec)	Mode	Engine On-Time (sec)	Total No. of Pulses	Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb$_f$-sec)	Engine ISP (sec)	Propellant Used For Event (lb$_f$)	Remaining Propellant (lb$_f$)
96	246.2	140.4	SS	14.02	1	14.02	-	70.0	227.1	.62	68.38
91	245.0	40.34	SS	4.07	1	4.07	-	18.8	208.7	.19	68.19
18	240.2	172.07	P	.587	31	147.6	.953	89.2	227	.79	67.39
83	239.6	19.08	P	.509	4	19.2	.954	8.2	187.3	.11	67.29
15	188.8	2403	P	.587	887	4218	.952	2527	227.1	11.11	56.17
36	155.7	2373	P	.589	1042	4949	.948	2507	225	11.13	45.04
54	132.5	2343	P	.670	1054	4981	.941	2494	223.6	11.13	33.91
74	115.4	2318	P	.672	1178	5558	.937	2473	227.6	11.08	22.83
21	114.8	94.4	P	.635	28	130.5	.935	49.6	218.9	.46	22.37
55	114.5	37.8	P	..555	13	59.9	.935	18.9	203	.19	22.17
88	109.3	847.9	P	.674	483	1139	.934	455	222	4.1	18.08
65	108.9	75.5	P	.675	23	102	.932	40.5	217	.37	17.70
98	108.6	37.7	P	.516	15	67	.932	19.1	203	.20	17.5
28	107.7	168.9	P	.675	99	473	.932	90.0	221	.82	16.7
55	107.4	37.7	P	.636	12	55	.932	19.4	203	.20	16.5
78	106.5	169.7	P	.676	51	232	.932	91.1	221	.82	15.7
53	106.1	75.3	P	.557	28	126	.932	39.8	216	.37	15.3
30	105.9	17.7	P	.596	6	27.7	.932	8.4	182	.10	15.19
31	104.8	217.4	P	.676	65	620	.931	115.7	221	1.05	14.13
19	104.6	35.3	P	.597	13	56	.932	18.3	202	.19	13.94
34	101.7	584	SS	111.6	1	112	-	292	222	2.63	11.32
57	100.5	67.4	P	.094	154.4	109	.926	36.5	221	.339	10.89
70	98.81	352.7	P	.094	802	568.8	.926	191	221	1.73	9.25
58	97.7	67.2	P	.094	154	109.2	.925	36.49	221	.337	8.91
39	97.02	40.2	P	.094	92.8	65.8	.925	21.9	221	.211	8.70

TABLE 5.1.0-X
PROBE MISSION PROFILE
SYSTEMS VII, VIII, IX, XIII, XIV & XV (Configurations P-10 & P-13)

Event	Event Magnitude	No. of Steps Used	Equivalent No. of Engines	SATELLITE Mass After Event	Spin Rate After Event (RPM)	Spin Inertia I_{zz} After ($lb\text{-}in^2$)	Tank Pressure After Event (psia)
1. Lift Off							
2. 3rd Stage Burnout							
3. De-spin - Hydrazine	-40 rpm	1	2	1051.4	29.98	335796	246.1
4. Erect Booms							
5. Spin-Up to Cruise Valve	+4 rpm	1	2	1051.2	11.98	898291	244.9
6. S/C Separation from 3rd Stage							
7. Orient for Cruise	90 deg	1	2	879.4	11.98	833518	240.2
8. Orient for 1st M/C	10 deg	1	2	879.3	11.98	833483	239.6
9. 1st Midcourse Correction	27 m/sec	4	1	868.2	11.94	829815	188.8
Total 108 m/sec	27 m/sec		1	857.0	11.90	826136	155.7
	27 m/sec		1	845.9	11.86	822454	132.5
	27 m/sec		1	834.8	11.82	818775	115.4
10. Re-orient to Cruise	50 deg	1	2	834.4	11.82	818621	114.8
11. Orient for 2nd M/C	20 deg	1	2	834.2	11.82	818555	114.5
12. 2nd Midcourse Correction	10 m/sec	1	2	830.1	11.81	817188	109.3
13. Re-orient to Cruise	40 deg	1	2	829.7	11.80	817065	108.9
14. Orient for 3rd M/C	20 deg	1	2	829.5	11.80	816728	108.6
15. 3rd Midcourse Correction	2 m/sec	1	2	828.7	11.80	816655	107.7
16. Attitude Control Maintenance	20 deg	1	2	828.5	11.80	816378	107.4
17. Orient Spin Axis Parallel to Ecliptic	90 deg	1	2	827.7	11.79	816378	106.5
18. Re-target Maxi Probe	40 deg	1	2	827.3	11.79	816253	106.1
19. Separate Maxi Probe							
20. Attitude Control Maintenance	10 deg	1	2	427.2	11.79	765530	105.9
21. Retarget Mini Probe	5 m/sec	1	2	426.1	11.78	765180	104.8
22. Attitude Control Maintenance	20 deg	1	2	425.9	11.78	765118	104.6
23. Spin Up Mini Probes	+73 rpm	1	2	423.3	84.69	764235	101.7
24. Separate Mini Probes							
25. Attitude Control Maintenance	20 deg	1	2	194.0	84.64	201957	100.5
26. Retarget S/C Bus	18 m/sec	1	2	192.2	84.59	201370	98.81
27. Correct Sun Angle Drift	20 deg	1	2	191.9	84.55	201258	97.7
28. Orient Spin to Velocity Vector	12 deg	1	2	191.7	84.52	201189	97.02

FOLDOUT FRAME

Initial Press. 250 psia
No. of Tanks 6
Tank Dia. .823 ft.
Initial Prop 69 lbs.

SATELLITE			ENGINE								
Spin nertia z After b-in^2)	Tank Pressure After Event (psia)	Total Impulse Required (lb_f-sec)	Mode	Engine On-Time (sec)	Total No. of Pulses	Correction Elapsed Time (sec)	Engine Rotational Efficiency	Engine Impulse Required (lb_f-sec)	Engine ISP (sec)	Propellant Used For Event (lb_f)	Remaining Propellant (lb_f)
35796	246.1	140.4	SS	14.02	1	14.02	-	70	227	.62	68.38
98291	244.9	40.3	SS	4.07	1	4.07	-	18.8	209	.19	68.19
33518	240.2	172.1	P	.587	31	147.5	.953	89.2	227	.79	67.39
33483	239.6	19.1	P	.509	4	19.2	.954	8.2	187	.11	67.29
29815	188.8	2404	P	.587	887	4218	.952	2527	227	11.11	56.17
26136	155.7	2373	P	.589	1042	4949	.948	2507	225	11.13	45.04
22454	132.5	2343	P	.670	1053	4981	.941	2494	224	11.13	33.91
18775	115.4	2312	P	.672	1178	5558	.937	2473	223	11.08	22.83
18621	114.8	94.42	P	.635	261	130.5	.935	49.6	219	.46	22.37
18555	114.5	37.8	P	.555	120	60	.935	18.9	203	.19	22.17
17188	109.3	847.9	P	.674	242	1139	.934	455	222	4.10	18.08
17065	108.9	75.5	P	.675	204	102	.932	40.5	217	.374	17.7
16728	108.6	37.7	P	.516	15	67	.932	19.1	203	.20	17.5
6655	107.7	168.9	P	.675	50	236.5	.932	90.0	221	.82	16.68
6378	107.4	37.7	P	.636	12	55	.932	19.4	203	.20	16.49
6378	106.5	169.7	P	.676	51	232.5	.932	91.1	221	.82	15.66
6253	106.1	75.3	P	.557	28	126	.932	39.8	216	.37	15.29
5530	105.9	17.7	P	.596	6	27.5	.932	8.4	182	.10	15.19
5180	104.8	217.4	P	.676	65	310	.931	115.7	221.4	1.05	14.13
5118	104.6	35.3	P	.597	13	56	.932	18.3	202	.19	13.94
4235	101.7	583.6	SS	111.6	1	112	-	292	222	2.63	11.32
1957	100.5	67.4	P	.094	152.4	108	.926	36.54	221	.339	10.89
1370	98.81	352.7	P	.094	802	568.8	.926	191	221	1.73	9.25
1258	97.7	67.2	P	.094	154.06	109	.925	36.49	221	.337	8.91
1189	97.02	40.2	P	.094	93.26	66	.925	21.79	220.5	.211	8.70

FOLDOUT FRAME

2

FOLDOUT FRAME 1

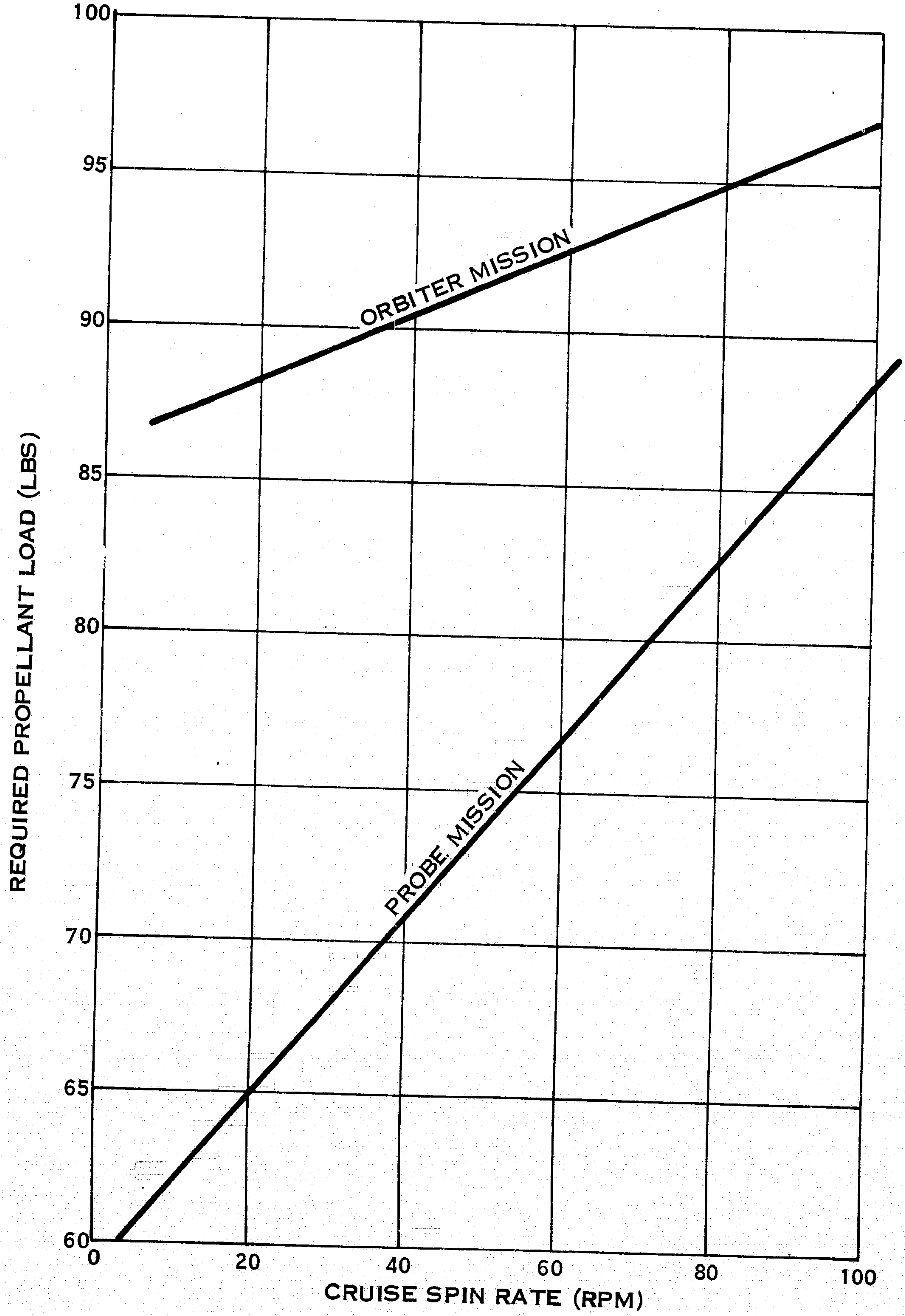


FIGURE 5.1.1–1. REQUIRED PROPELLANT LOAD VS CRUISE SPIN SPEED

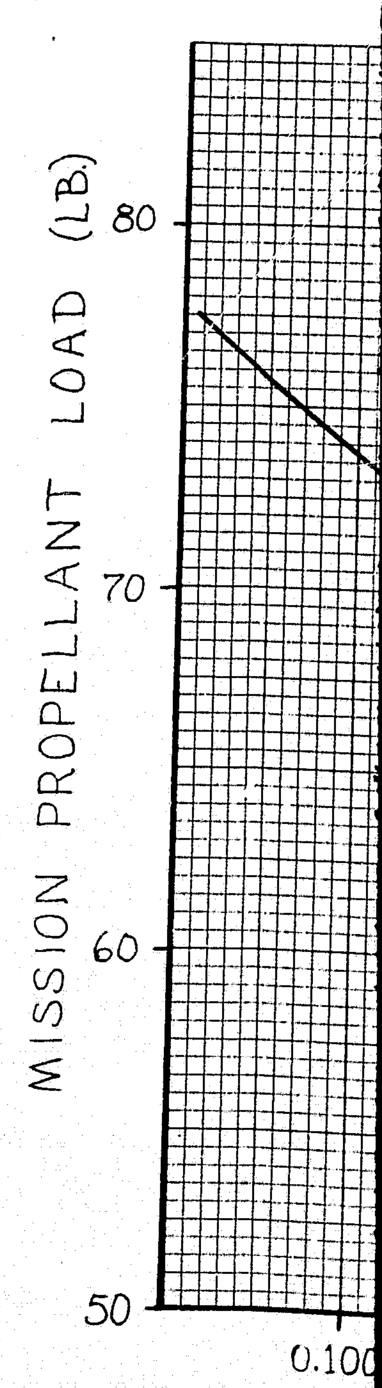


FIGURE 5

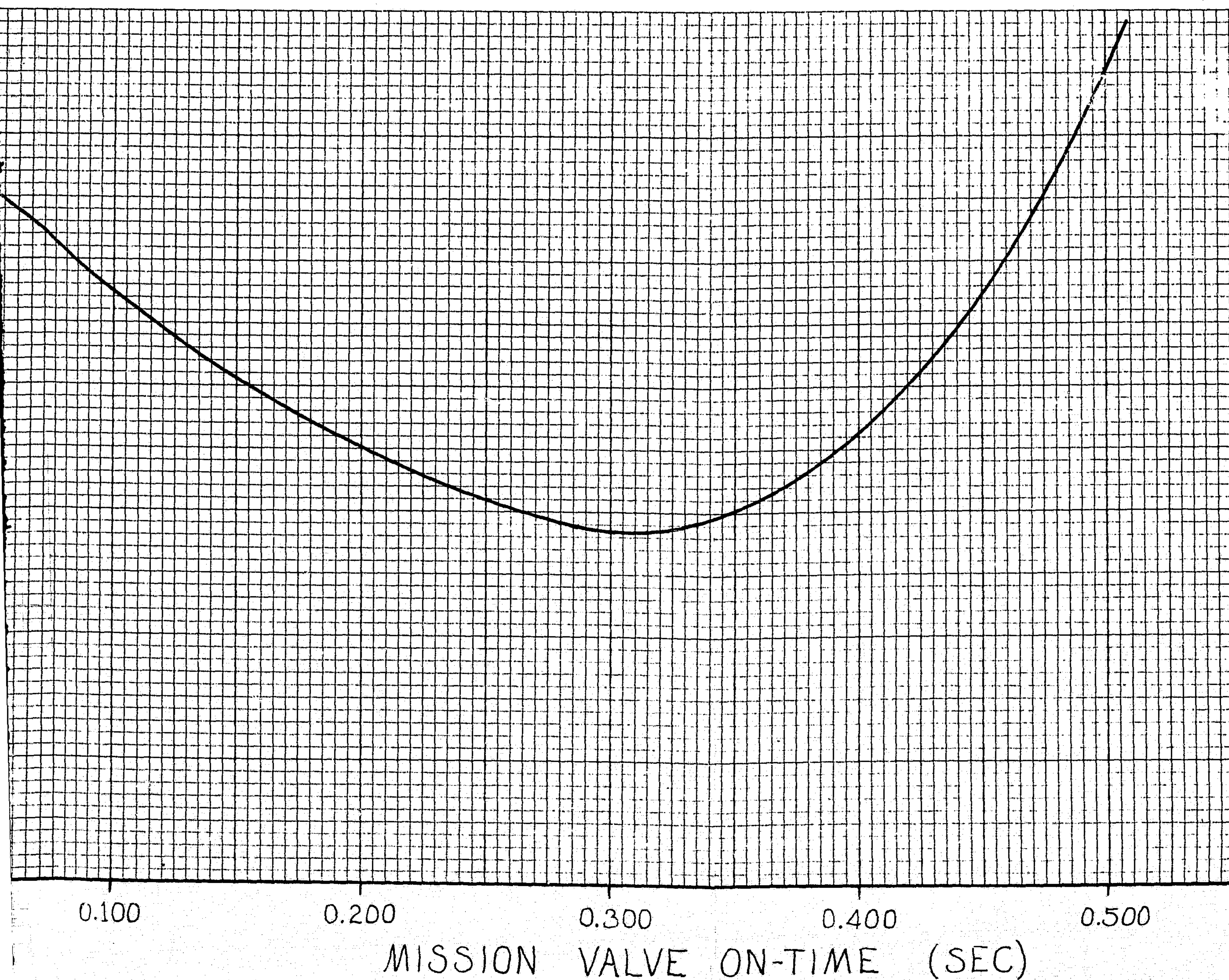


FIGURE 5.1.2-1. MISSION PROPELLANT LOAD VS VARIOUS FIXED ELECTRICAL ON-TIMES

5.2 Reliability

The Feed System concepts and Engine Placement concepts selected in the previous phases of the study have been analyzed for their reliability characteristics. The results of the analysis, presented in this section, include a tabulation of failure rates and their sources, a Failure Mode and Effects Analysis, a description of the methods used for calculating reliability of the candidate subsystems, and the results of the calculation. The ground rules used in the analysis are listed below.

a. The mission will be a success with respect to a candidate subsystem if it does not fail before or between uses and if it accomplishes the following:

1.	Non-operating flight.	1 year
2.	Spin or de-spin, delta velocity, and attitude control.	Maximum of 6000 pulses/60 minutes usage per engine.

b. No single failure having any significant probability of occurrence shall abort the mission.

c. Force couples are not required for spin or de-spin.

d. The vehicle is always spinning.

e. Force couples are required for attitude adjustment (this is accomplished by precession).

f. If an engine control valve fails open, a manifold valve will be closed, to prevent unwanted thrust.

g. Manifold valves are closed during any period when thrust is not needed.

h. Propellant is not admitted to the engine system until one hour after launch. This first hour is considered a "start" mode. Analysis is based on a one-year operating requirement, in "mission" mode.

i. In primary operating configurations, the required engines will be fired simultaneously once for each 360° of spin. A 180° firing interval will be used when necessary to achieve the required function in secondary configuration.

5.2.1 ESTIMATED FAILURE RATES:

The failure rates used in the study are tabulated in this section. In each case, the basis of selection of the rate is stated. The Bureau of Naval Weapons Failure Rate Data Handbook (FARADA) is the principal reference. Hamilton Standard product and test experience, customer-generated data, and supplier data are used if appropriate.

The principal usefulness of numerical reliability analysis is in making comparisons between concepts and subsystems. When the part and component failure rates have been carefully chosen and consistently applied, the comparative reliability values of the candidate subsystems provide a valid base for subsystem selection. The accuracy of the failure rates does not, of itself, justify the number of significant places retained in the reliability calculations. However, when the same failure rates are used for all concepts under consideration, a reliability computation carried to six or more significant places will facilitate ranking of the concepts.

Table 5.2-1-I presents the failure rates used in this study.

5.2.2 FAILURE MODE AND EFFECTS ANALYSIS:

A failure mode and effects analysis which identifies the probable failure modes of the candidate subsystems and the probable effects of such failures is presented in Table 5.2.2-I. The single analysis covers all of the candidate subsystems, because of similarity of functions of the same components in each system. Where distinctions arise because of the subsystem configuration, the effects are noted. This analysis is made within the ground rules listed above. In the event of failures beyond the scope of the established ground rules, there are two other types of remedial action possible with the candidate subsystems. These additional actions have been examined as a part of the preparatory studies, but they have not been used in the preparation of the reliability estimates of this report. They are:

a. If the "fail open" of an engine valve is a small internal leak, operation of the "failed" engine may be continued by opening its corresponding manifold valve a short interval before the thruster is needed and reclosing the manifold valve upon completion of the scheduled engine operation. Similarly, the manifold valve may be opened temporarily to operate other engines which are fed by the same manifold.

b. If the "fail open" of an engine valve results in delivery of propellant to the engine in sufficient quantity to produce thrust, operation may be obtained only when all other engines connected to the same manifold are non-operating. Individual operation of the other engines from the same

5.2.2 (Continued)

manifold may not be obtained. Operation of the "failed" engine may then be by opening and closing the manifold valve. This degraded mode of operation is of limited potential usefulness.

In the FMEA, the principal failure modes of each component have been identified, together with the probable causes of each failure mode. Failure modes which have a negligible probability of occurrence have not been considered, except as they appear in the failure rate list. The probable effect of each failure mode during the mission is noted, together with the methods of preventing each failure mode. The relative probability of occurrence of each failure mode is a qualitative indication of the unreliability of each component, and is entered as a non-dimensional number. For each failure mode, the symptoms of failure available to earth-based controllers and the corrective actions are suggested.

5.2.3 RELIABILITY ESTIMATES:

The reliability values of the fifteen candidate subsystems have been estimated, and are presented in Tables 4.4.0-I and 4.4.0-II. This section describes the methods used for numerical evaluation and presents outlines of the detailed calculations. The fundamental relationships used are:

$R = 1 - Q$ where: R = Reliability, or probability of successful operation

Q = Unreliability, or probability of unsuccessful operation

$R = e^{-\lambda t}$ where: $e = 2.71828$

λ = Failure rate, usually in failures per million hours - may be in failures per million cycles

t = Time, in hours

$R_t = 1 - Q_c Q_o$ where: R_t = Reliability of a redundant pair of dissimilar items, one of which must perform for success

Q_c = Unreliability of item "c"

Q_o = Unreliability of item "o"

5.2.3.1 Feed System Concepts

The three feed system analyses are summarized in Figures 5.2.3-1 through 5.2.3-3. For each concept, the reliability block diagram appears vertically for easy correlation with the failure modes considered, and failure rates for those modes. The first hour of the mission is taken as the period between launch and the first required operation of any engine. The failure rates applicable to that period are listed in the "Start" column. The "Mission" column lists the failure rates applicable for the remainder of the mission. For conservatism, all calculations are based upon the Orbiter mission and its longer duration. In each feed system, the latching solenoid valves which feed the fuel manifolds (for example, items 9 and 10 in FS-4) are considered only for failure modes which can inadvertently cut off fuel flow.

The pressure transducer failure modes which degrade its output signal have not been included in computing overall subsystem reliability. Tank pressure can be predicted accurately on the basis of both vehicle response to maneuvers or vehicle maneuver and thermal history. If a failure or apparent failure occurs in a tank pressure reading a simple spin or attitude maneuver can be used to check the value of tank pressure and determine whether a transducer or leakage failure has occurred.

5.2.3.2 Engine Placement Concepts

The five engine placement concepts are summarized in Figures 5.2.3-4 through 5.2.3-8. Each figure presents a schematic of the engine placement, a schematic of the manifolding arrangement, a table to establish which engines may be used to accomplish each required function, a reliability schematic, and mathematical models.

In the manifolding arrangement, the valves marked "M" are latching solenoid valves. Either valve may be closed if one of the engines under its control fails in such manner that it produces unwanted thrust or leakage. For convenience of illustration, there is an overlapping between the manifolding arrangement schematic and the fuel system functional schematic. In FS-4, for example, items 9 and 10 on the functional schematic are the manifold valves.

The functional arrangement table indicates those combinations of engines which can be used to perform the necessary functions, in the normal mode of operation, and in the available degraded modes. In every function, some of the degraded modes may be operated with one or the other of the manifold valves closed. This table is the basis for developing the equation for reliability of the engine array, taking advantage of all available redundancies, including the allowable degraded modes.

For each engine array, a tabulation of possible failed engines and their failure modes was prepared. The following possibilities were included:

a. All engines operate properly and both manifold valves remain open. There can be one such case for each array.

b. Any one engine fails closed (no thrust) and manifold valves remain open. In a six engine array, there can be six such cases.

5.2.3.2 (Continued)

c. Any one engine fails open (leakage or unwanted thrust) and the corresponding manifold valve is closed. In a six engine array, there can be six such cases.

d. Any two engines fail closed and both manifold valves remain open. In a six engine array, there can be fifteen such cases.

e. Any two engines fail open and one or both of the manifold valves have to be closed. In a six engine array, there can be fifteen such cases. For purposes of the present analysis, every case in which both of the manifold valves are closed is considered a failure of the system.

f. One engine fails open and one engine fails closed, and one manifold valve is closed. In a six engine array, there are thirty possible combinations.

The cases for three engines failed were examined briefly. So few of them would be successes that they were judged to have an insignificant effect upon the probability of success of the system.

From the tables of failure and failure mode combinations equations were prepared to represent the success cases. As a check, system "no-success" euqations were also prepared and added to the "success" equations to obtain a binominal expansion. The "success" equation for each engine array was then used to calculate the reliability of each array. The "success" equation for a six engine array (concept P7, for example) is obtained as follows:

The first term of the equation represents the probability that all engines will operate correctly for the entire mission. That probability is:

$$R_x = R_1 \times R_2 \times R_3 \times R_4 \times R_5 \times R_6$$

Since all of these are equal

$$R_x = R^6$$

For the second term of the equation, we take the probability that five engines will operate correctly for the entire mission and one will fail closed some time during the mission. That probability is:

$$R_y = R_1 \times R_2 \times R_3 \times R_4 \times R_5 \times Q_{c6}$$

5.2.3.2 (Continued)

where it is assumed that engine No. 6 has failed closed. Since each engine has an equal probability of failing closed and there are six such cases possible,

$$R_y = 6 R^5 Q_c$$

The third term, which represents all cases in which one engine fails open, is obtained the same way.

The fourth term of the equation describes the probability of having two engines fail open and one or both of the manifold valves moved to the closed position. There are fifteen possible combinations of failures which meet this description. One possibility is:

$$R_z = R_1 \times R_2 \times R_3 \times R_4 \times {Q_o}^5 \times {Q_o}^6$$

If all fifteen combinations result in mission success, the term becomes:

$$R_z = 15 R^4 {Q_o}^2$$

However, inspection of the table shows that only six of the cases with two engines failed open can be successes for the P-7 concept, so the term is

$$6 R^4 {Q_o}^2$$

The probability of having two engines fail closed is obtained in the same way. Inspection of the table shows that six of the fifteen cases can be successes, so the fourth term of the equation becomes:

$$R_u = 6 R^4 {Q_c}^2$$

Inspection of the table shows that there are twelve successful cases in which one engine fails closed and another fails open. Therefore the "success" equation becomes:

$$R_E = R^6 + 6 R^5 Q_c + 6 R^5 Q_c + 6 R^4 {Q_c}^2 + 6 R^4 {Q_o}^2 + 12 R^4 Q_c Q_o$$

The exact "success" equation obtained varies with the definition of success. This equation meets the following:

5.2.3.2 (Continued)

Success will be achieved if each required function can be obtained with either one, but not both, of the manifold valves closed.

This definition and equation correspond to the ground rules stated above.

5.2.3.3 Engine Fuel Connections

Each system has one connection in the feed system for each engine, which can be disassembled for engine removal. With a six engine system (for example, FS-7) there are six such connections and all other discontinuities in the fuel system are welded except for the transducer connection. For simplicity of analysis, it has been assumed that all of these connections are under pressure for the majority of the mission, starting one hour after launch. Thus for a six engine system:

$$R_F = e^{-.01 \times 10^{-6} \times 8760 \times 6} = .99947\ 45381$$

and for an eight engine system

$$R_F = e^{-.01 \times 10^{-6} \times 8760 \times 8} = .99929\ 94455$$

5.2.3.4 Engine Manifold Valve

Those failure modes of the manifold valve which are associated with opening and remaining open after the first hour of the mission are handled as part of the feed system reliability calculations. The failure modes associated with failure to close when signaled are handled as part of the engine concept calculation. The probability of successful closing of either one of the manifold valves during the mission is:

$$R_M = e^{-.2 \times 10^{-6} \times 8760 \times 2} = .99650\ 21318$$

For simplicity of analysis, this probability of success has been associated with the entire mission (after the first hour) rather than being limited to degraded-mode operations. This simplification is in the conservative direction.

5.2.3.5 Engine Gimbal Actuator (Concept P-12 only)

In engine placement concept P12, two actuators are used, to adjust the position of two engines in accordance with shifts in center of gravity of the vehicle. The actuators move through one stroke only, during the first 150 hours of the mission. Thus the probability that both actuators will perform properly is:

$$R_A = e^{-2. \times .4 \times 10^{-6} \times 150 \times 24} = .99712\ 41432$$

5.2.3.6 Summary

The reliabilities for the individual feed systems and engine placement concepts are presented in Figures 5.2.3-1 through 5.2.3-8. They are also presented in Tables 4.4.0-I and 4.4.0-II, together with the subsystem reliabilities obtained from all available combinations of fuel system and engine placement. All of the candidate subsystems exceed the reliability required of the subsystem.

Engine placement concept P-7, with six engines is shown to be more reliable than concepts P-5 and P-13, which have eight engines each. The principal reason for this is that the reliability associated with the connections of individual engines to the fuel system, R_F, is higher for the six engine concept. The reliabilities for the two cases are given in Para. 5.2.4.3 above.

The second reason is that R_E is slightly higher for concept P-7, by the nature of the controlling equations, where

$$R_{E5} = R^8 + 8R^7 Q_o + 8R^7 Q_c + 12R^6 Q_o^2 + 20R^6 Q_c^2 + 24R^6 Q_o Q_c$$

and

$$R_{E7} = R^6 + 6R^5 Q_o + 6R^5 Q_c + 6R^4 Q_o^2 + 6R^4 Q_c^2 + 12\, R^4 Q_o Q_c$$

Using				
	$R = .99880\ 07197$		$R^6 = .99282\ 58579$	
	$Q_o = .00089\ 95951$		$R^8 = .99044\ 59329$	
	$Q_c = .00000\ 00900$			

the results are

$$R_{E5} = .99997\ 99193$$
$$R_{E7} = .99998\ 86957$$

It will be noted that the probability of all engines working properly for the entire mission, R^6, is substantially higher for the six engine concept than R^8 in the eight engine concept. In each case, this is the first term of the equation and the overlapping influence upon the numerical reliability of the concept.

Concept P-7 as presented in the preceding analysis makes considerable use of ground rule "i" as stated in Para. 5.2.1 above. The second, fourth, and sixth terms of the equation are completely dependent upon a capability for firing selected engines at 180° intervals of vehicle spin. If this capability did not exist, the controlling equation would be

$$R_{E7a} = R^6 + 6R^5{}_c + 6R^4 Q^2{}_c$$

This would degrade the numerical reliability of concept P-7 to be lower then any of the others, and would suggest that alternate modes of operation or alternate manifolding arrangements might be investigated.

TABLE 5.2.1-I. ESTIMATED FAILURE RATES

Component or Part	Failure Rate	Remarks and Sources
N_2 Fill and Drain Valve	Negligible	Valve fill ports are capped; Leak checks after fill verify readiness.
Propellant Tankage	Negligible	Structural design has generous margins of strength. The material and processing is closely specified and controlled. Rigorous testing and quality control verify integrity of product.
Piping and Connections	Negligible	All piping and connections are welded, except as specifically noted below. Non-destructive testing verifies integrity of individual units.
Pressure Transducer		
• Signal Error	2.6×10^{-6}/hr	Subsystem Specification No. S-723-P-10 for MICOMACS Subsystem - Appendix C, Table 2.
• Connection Leak	$.01 \times 10^{-6}$/hr	Flared fittings, subject to careful assembly, quality control, and leakage test before launch.
N_2H_4 Fill and Drain Valve	Negligible	Same as N_2 Fill and Drain Valve.
Filter	Negligible	Based on FARADA plus experience with Hamilton Standard JFC-60 Fuel Control Filters.
Latching Solenoid Valve		
• Fail to open	$.10 \times 10^{-6}$/hr	Derived from FARADA data plus in-house data on the individual piece parts plus estimate of probability of failure modes.
• Fail to close - major opening remaining	$.01 \times 10^{-6}$/hr	
• Fail to close - result is internal leak	$.09 \times 10^{-6}$/hr	
• Fail to hold "latched" position	Negligible	
Normally Closed Ordnance Valve including One SBASI (two SBASI to be incorporated) -		
• Fail to fire	10×10^{-6}/cycle	Derived from FARADA data plus Hamilton Standard experience plus NASA reported experience with Single Bridgewire Apollo Standard Initiator (SBASI).
• Fail to open	Negligible	
• Internal leak	Negligible	
Engine-to-System Connection - Leak	$.01 \times 10^{-6}$/hr	Flared fittings, subject to careful assembly, quality control, and leakage test before launch.
Test Port	Negligible	Same as N_2 Fill and Drain Valve.
Engine Valve		Derived from In-House data on piece parts plus estimate of probability of failure modes.
• Fail to open	$.05 \times 10^{-6}$/cycle	
• Fail to close - major opening remaining	$.01 \times 10^{-6}$/cycle	
• Fail to close - result is internal leak	$.09 \times 10^{-6}$/cycle	
• External Leak	Negligible	
Engine		
• Incorrect Thrust	$.05 \times 10^{-6}$/cycle	Derived from In-House data.
• Explosion	Negligible	
Actuator	$.4 \times 10^{-6}$/cycle	Derived from FARADA.
Flex Line to Thruster	Negligible	Solid line, coiled to accept movement of thruster through arc of less than 10° for 200 cycles.

TABLE 5.2.2-I. FAILURE MODES AND EFFECTS

Component and Failure Mode	Probable Cause of Failure Mode	Probable Effect of Failure Mode	Method of Prevention of Failure Mode	Probability of Occurrence
1. N_2 Fill & Drain Valve				
a. Internal leak	a. Dirt or other contaminant. Valve spring weakens or breaks. Seal degradation.	a. Loss of propellant pressure. Loss will probably be gradual. Eventual performance degradation in engines, for lack of the required fuel pressure	a. Cap to be installed over connector at all times when not in use, for redundant sealing. Maintain cleanliness of entire system and of propellant supply. All structural parts are designed to conversative strength-stress margins.	a. 1
b. External leak	b. Same as 1.a above	b. Same as 1.a above	b. All connections to the system and external case of item are welded to seal against external leakage.	b. 1
2. N_2H_4 Fill & Drain Valve				
a. Internal leak	a. Dirt or other contaminant. Valve spring weakens or breaks. Seal degradation.	a. Loss of propellant. Loss will probably be gradual. Contamination of vehicle and equipment.	a. Cap to be installed over connector at all times when not in use, for redundant sealing. Maintain cleanliness of entire system and of propellant supply All structural parts are designed to conservative strength-stress margins.	a. 1
b. External leak	b. Same as 2.a above	b. Same as 2.a above	b. All connections to the system and external case of item are welded, to seal against external leakage.	b. 1
3. Propellant Tank				
a. Leak at Fittings or Welds	a. Cracking or other breakage	a. Loss of propellant pressure. Loss will probably be gradual. Leakage of propellant may contaminate vehicle or equipment. Possible abort of mission.	a. All connections are welded. All are verified by non-destructive test. System is pressurized before launch, and may be leak tested. Structural support is provided for all items to ensure conservative strength-stress margins.	a. 1
b. Rupture of Tank	b. Structural overload or mechanical damage	b. Loss of propellant. Leak will contaminate vehicle and equipment. Abort mission.	b. Provide thorough testing of tank and fitting design, demonstration of strength-stress margins, careful in-process and final inspection, and careful handling.	b. 1
4. Pressure Transducer				
a. Error in Signal Output	a. Electrical elements or connections change their characteristics. Failure of sensing diaphragm	a. Inconvenience in determining status of system in flight.	a. Pressure indication is advisory only. Opportunity for check is during propellant fill and pressurization before flight.	a. 200

FOLDOUT FRAME 2

Probability of Occurrence	Symptoms of Failure	Corrective Action	Remarks
a. 1	a. Abnormal loss of pressure	a. None	a. Probable slow decay of pressure
b. 1	b. Abnormal loss of pressure	b. None	b. Probable slow decay of pressure
a. 1	a. Abnormal loss of pressure	a. None	a. Probable slow decay of pressure, propellant leak, and contamination
b. 1	b. Abnormal loss of pressure	b. None	b. Probable slow decay of pressure propellant leak, and contamination
a. 1	a. Abnormal loss of pressure	a. None	a. Probable slow decay of pressure propellant leak, and contamination
b. 1	b. Abnormal loss of pressure	b. None	b. Rapid loss of pressure
a. 200	a. Pressure signal does not correspond to status of propellant usage and tank temperature	a. Estimate remaining propellant quantity by analysis of programmed maneuver and resulting performance	

FOLDOUT FRAME

TABLE 5.2.2-I. FAILURE MODES AND EFFECTS (continued)

Component and Failure Mode	Probable Cause of Failure Mode	Probable Effect of Failure Mode	Method of Prevention of Failure Mode	Probability of Occurrence	
4. (continued)					
b. Leakage	b. Seal or diaphragm degradation. Degradation of mechanical connection of sensing element to fuel system.	b. Loss of propellant pressure. Possible abort of the mission.	b. Design includes redundant diaphragms. Welded case prevents external leak from transducer. System is pressurized before launch and may be inspected for leaks, including the connection to the fuel system.	b. 5	b. Al
5. Filter					
a. Clogging	a. Contamination collected.	a. High pressure drop across filter, causing low pressure at engines, and loss of performance.	a. Control fuel and system cleanliness and size filter volume with margin.	a. 1	a. O m m ti
b. Pass contaminants	b. Crack or other failure of filter element. Deterioration of filter element, making contamination for downstream items.	b. Possibility of clogging downstream screens, with degradation of latching valves, thruster valves, etc. Probably will cause internal leaks in valves.	b. Screens in downstream valves should stop major contaminants. Filter will be subject to rigorous development testing and close quality control.	b. 1	b. Fa st In do
6. Latching Valve					
a. Failure to open on signal	a. Electrical discontinuity	a. System cannot be energized and no propellant will be delivered to thrusters. Abort mission.	a. Latching valves are redundant partially in FS-2 and fully in FS-4.	a. 50	a. Va sh Th va
b. Internal leak	b. Contamination, seal degradation, or excessive vibration or shock loads	b. Gradual pressurization of downstream items, including thruster valves	b. Test during pre-launch will indicate any serious leakage.	b. 50	b. Sy th
c. Fail to latch open	c. Spring weakening or breakage	c. Inadvertent opening of valve, with pressurization of downstream items before schedule	c. Spring is Belleville type, and very unlikely to sustain this failure mode when in a fixed position.	c. 1	c. Sa
7. Latching Valve					
a. Failure to close on signal	a. Electrical discontinuity. Sticking. Contamination.	a. System cannot be shut down partially in order to cut off malfunctioning valve or thruster. Abort may result from this unlikely double failure.	a. "Closing" signal will not be given unless there is an initial failure in a thruster or its valve. The latching valve can be tested for continuity and operation during prelaunch. Valve is screened, in addition to upstream filter.	a. 50	a. Va sh

ility irrence	Symptoms of Failure	Corrective Action	Remarks
	b. Abnormal loss of pressure	b. None	b. Probable slow decay of pressure, propellant leak, and contamination
	a. Observe low thrust for all maneuvers. Detected by the magnitude of maneuver relative to tank pressure	a. Plan maneuvers based upon a new estimated tank pressure thrust relationship accounting for restriction in line.	
	b. Failures of one or more downstream components Intermittent failures of downstream components	b. None	
	a. Valve position indicator shows "no open" Thrusters fed by failed valve do not operate	a. Perform maneuvors with alternate thrusters Repeat signal to open valve	a. Temperature sensors may be provided on thrusters to report operation of thrusters. Maneuver performance is secondary report of thruster operation
	b. Symptom obtainable only through diagnostic test	b. None	
	c. Same as 6.b above	c. Same as 6.b above	c. Same as 6.b above
	a. Valve position indicator shown "no close"	a. Repeat signal to close valve	a. If there is no failure of propellant valve or thruster downstream of the latching valve, the failure has little effect upon mission.

TABLE 5.2.2-I. FAILURE MODES AND EFFECTS (continued)

Component and Failure Mode	Probable Cause of Failure Mode	Probable Effect of Failure Mode	Method of Prevention of Failure Mode	Probability of Occurren
7. (continued)				
b. Fail to latch closed	b. Spring weakening or breakage	b. Inadvertent closing of valve, which will shut off half of the thrusters before commanded.	b. Spring maintaining the latched position is Belleville type, and very unlikely to sustain this failure mode when in a fixed position. Back-up thruster can complete mission.	b. 1
8. Test Port				
a. Internal Leak	a. Dirt or other contaminant. Valve spring weakens or breaks. Seal degradation.	a. Loss of propellant after system is pressurized. The loss will probably be gradual. Eventual performance degradation in thrusters, for lack of required propellant pressure.	a. Cap to be installed over test port at all times when not in use, for redundant sealing. Maintain cleanliness of entire system and of test gas. All structural parts are designed to conservative strength-stress margins.	a. 1
b. External Leak	b. Same as 8.a above	b. Same as 8.a above	b. All connections to the system and the external case of the item are welded to seal against external leakage.	b. 1
9. Ordnance-operated Valve				
a. Internal Leak	a. Leak at sealed end of pipe	a. Gradual pressurization of downstrem items, including thruster valves	a. Test during pre-launch will indicate any serious leakage. Leak is of consequence only during the first hour of the mission. Probability of leakage negligible, because pipe seal is low-stressed structural part, subject to rigorous inspection before and after assembling into system.	a. 1
b. Failure to Open	b. Failure of squibs to fire, caused by electrical discontinuity	b. None since parallel ordnance valve is available.	b. Redundant valves are used. Redundancy used in electrical firing circuitry. Circuits subject to low-voltage continuity check prior to launch.	b. 10
10. Thruster Valve				
a. External leak at connection to fusl system	a. Improper assembly and tightening of connection	a. Loss of propellant after system is pressurized. The loss will probably be gradual. Eventual performance degradation in thrusters, for lack of required propellant pressure.	a. Test during prelaunch will indicate leakage, and verify quality of the connection.	a. 5

Probability of Occurren !	Symptoms of Failure	Corrective Action	Remarks
b. 1	b. Same as 7.a above	b. Same as 7.a above	b. Same as 7.a above
a. 1	a. No sysptoms unless the latching valve or ordance-operated valve upstream is open. In that case, note abnormally large propellant consumption	a. None on FS-8. On FS-2 or FS-4, keep latching valve closed in all non-operating thruster modes	
b. 1	b. Same as 8.a above	b. Same as 8.a above	b. Same as 8.a above
a. 1	a. None after liftoff	a. None	
b. 10	b. Thrusters inoperative	b. Back-up valve to open.	
a. 5	a. Abnormally large propellant consumption	a. Close latching valve as appropriate	a. If leakage is gross, confine operation to the unaffected half of the thruster system

TABLE 5.2.2-I. FAILURE MODES AND EFFECTS (continued)

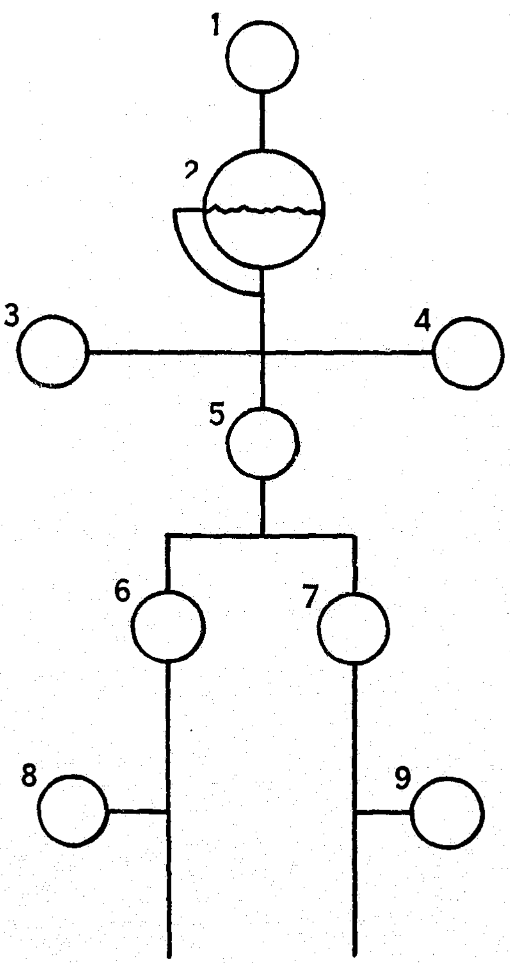
Component and Failure Mode	Probable Cause of Failure Mode	Probable Effect of Failure Mode	Method of Prevention of Failure Mode	Probability of Occurrence	
10. (continued)					
b. Fail to Open	b. Electrical discontinuity. Sticking of valve parts. Contamination.	b. No propellant will be delivered to the thruster associated with the failed valve.	b. Alternate thruster or thrusters will perform the function upon command. Maintain cleanliness of entire system and propellant supply.	b. 25	b. Ab eu
c. Fail Open (major flow)	c. Sticking of valve parts. Contamination Valve spring weakens or breaks.	c. Thruster associated with the failed valve cannot be shut off. Unscheduled thrust disrupts navigation and positioning of vehicle	c. Maintain cleanliness of entire system and propellant supply. All structural parts, including springs, are designed to convervative strength-stress margins. Manifold propellant supply may be shut off by closing latching valve (Item 6 above for FS-2, Item 7 for FS-4 and FS-8)	c. 5	c. Ab nor
d. Internal Leak (minor flow)	d. Same as 10.c above	d. Thruster associated with the failed valve may give unscheduled thrust	d. Same as 10.c above. Possible to re-open latching valve temporarily if thruster is required to operate.	d. 45	d. No of pe ch
11. Thruster					
a. Incorrect thrust	a. Improper propellant pressure. Incorrect valve opening or closing. Catalyst bed degraded.	a. Inability to perform the mission functional requirements. Disrupt navigation and positioning of vehicle.	a. Use alternate thruster or thrusters to perform the required function.	a. 25	a. Ab eu ma no th

FOLDOUT FRAME

obability Occurrence	Symptoms of Failure	Corrective Action	Remarks
25	b. Abnormal performance of maneuver.	b. Operate with alternate thrusters	b. Temperature sensors may be provided on thrusters to report operation of thrusters
5	c. Abnormal maneuver and abnormal fuel consumption	c. Close the appropriate latching valve	
45	d. Note abnormal temperature of thruster, loss of propellant, and unscheduled change in vehicle attitude	d. Close the appropriate latching valve	d. Temperature sensors may be provided on the thrusters to report operation of the thrusters
25	a. Abnormal performance of maneuver. Serious degradation may be detected through abnormal temperature of the thruster	a. For minor degradation, perform a calibration maneuver. For major degradation, operate with alternate thrusters	a. Temperature sensors may be provided on the thrusters to report operation of the thrusters

FOLDOUT FRAME 1

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

FUNCTIONAL SCHEMATIC	COMPONENTS	RELIABILITY SCHEMATIC	FAILURE MODE
	1) N_2 FILL & DRAIN VALVE	1	LEAK
	2) TANKS	2	LEAK
	3) N_2H_4 FILL & DRAIN VALVE	3	LEAK
	4) PRESSURE TRANSDUCER	4	LEAKY CASE
		4	CONNECT. LEAK
	5) FILTER	5	LEAK, CLOG
	6) LATCHING VALVE (CLOSED AT LAUNCH)	6	FAIL TO OPEN
	7) LATCHING VALVE (CLOSED AT LAUNCH)	7	FAIL TO OPEN
		6	FAIL TO LATCH
		7	FAIL TO LATCH
		6	INTERNAL LEAK
		7	INTERNAL LEAK
	8) TEST PORT	8	LEAK
	9) TEST PORT	9	LEAK

AILURE MODES	FAILURE RATES START	MISSION
EAK	NEGLIGIBLE	NEGLIGIBLE
EAK	NEGLIGIBLE	NEGLIGIBLE
EAK	NEGLIGIBLE	NEGLIGIBLE
EAKY CASE	NEGLIGIBLE	NEGLIGIBLE
ONNECT. LEAK	0.01×10^{-6}	0.01×10^{-6}
EAK, CLOG	NEGLIGIBLE	NEGLIGIBLE
AIL TO OPEN	0.10×10^{-6}	NOT APPL.
AIL TO OPEN	0.10×10^{-6}	NOT APPL.
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE
TERNAL LEAK	0.09×10^{-6}	NOT APPL.
TERNAL LEAK	0.09×10^{-6}	NOT APPL
EAK	NEGLIGIBLE	NEGLIGIBLE
EAK	NEGLIGIBLE	NEGLIGIBLE

MATHEMATICAL MODEL

START

$$R_S = e^{-\lambda t}$$
$$t = 1 \text{ HR}$$
$$\lambda = (0.10+0.10+0.09+0.09+0.01)10^{-6}$$
$$R_S = e^{-0.39 \times 10^{-6}}$$
$$= 0.9999996099$$

MISSION

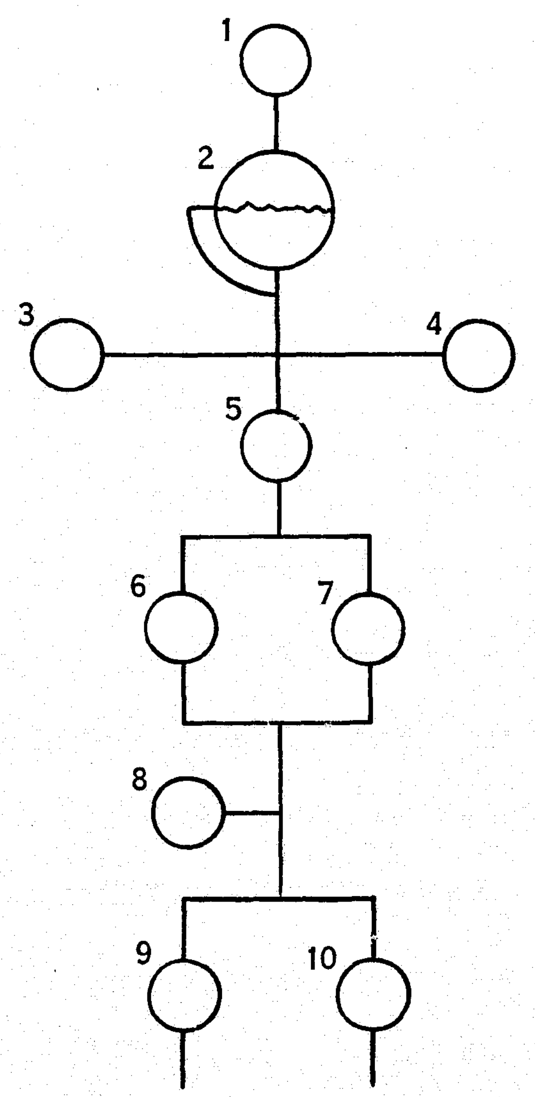
$$R_M = e^{-\lambda t}$$
$$t = 8760 \text{ HR}$$
$$\lambda = 0.01 \times 10^{-6}$$
$$R_M = e^{-0.01 \times 10^{-6} \times 8760}$$
$$= 0.9999124038$$

TOTAL FS–2

$$R_{FS-2} = R_S R_M$$
$$= 0.9999120139$$

FIGURE 5.2.3–1. RELIABILITY OF FEED SYSTEM FS–2

FOLDOUT FRAME

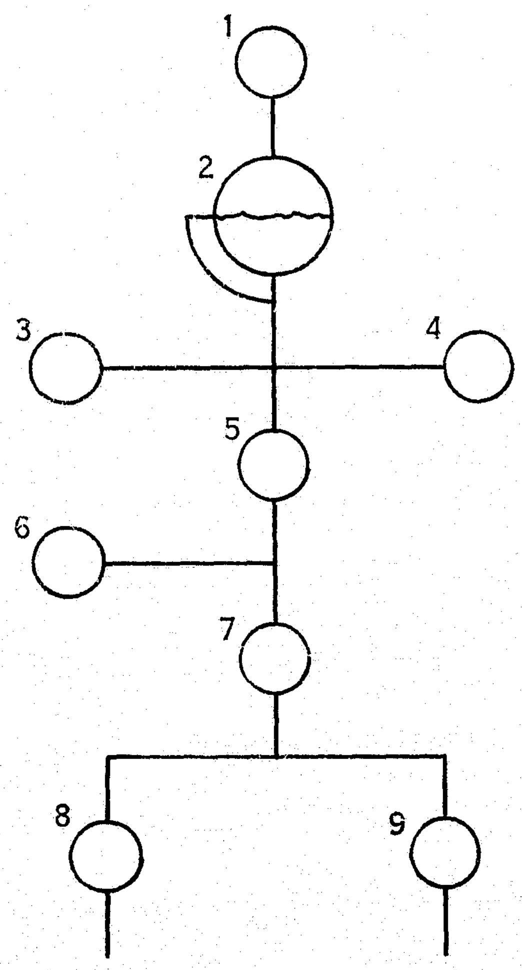
FUNCTIONAL SCHEMATIC	COMPONENTS	RELIABILITY SCHEMATIC	FAILURE MODES
	1) N_2 FILL & DRAIN VALVE	1	LEAK
	2) TANK	2	LEAK
	3) N_2H_4 FILL & DRAIN VALVE	3	LEAK
	4) PRESSURE TRANSDUCER	4	CASE LEAK
		4	CONNECT. LEAK
	5) FILTER	5	LEAK, CLOG
	6) LATCHING VALVE (CLOSED AT LAUNCH)	6	INTERNAL LEAK
	7) LATCHING VALVE (CLOSED AT LAUNCH)	7	INTERNAL LEAK
		6	FAIL TO LATCH
		7	FAIL TO LATCH
		6, 7	FAIL TO OPEN
	8) TEST PORT	8	LEAK
	9) LATCHING VALVE (OPEN AT LAUNCH)	9	FAIL TO LATCH
	10) LATCHING VALVE (OPEN AT LAUNCH)	10	FAIL TO LATCH

AILURE MODES	FAILURE RATES START	FAILURE RATES MISSION	MATHEMATICAL MODEL
EAK	NEGLIGIBLE	NEGLIGIBLE	START
EAK	NEGLIGIBLE	NEGLIGIBLE	$R_S = e^{-\lambda t}$ $t = 1$ HR $\lambda = (0.01+0.09+0.09)10^{-6}$
EAK	NEGLIGIBLE	NEGLIGIBLE	NOTE: REDUNDANCY OF ITEM 6 AND 7 MAKES FOURTH TERM NEGLIGIBLE
ASE LEAK	NEGLIGIBLE	NEGLIGIBLE	
ONNECT. LEAK	0.01×10^{-6}	0.01×10^{-6}	$R_S = e^{-0.19 \times 10^{-6}}$ $= 0.9999997999$
EAK, CLOG	NEGLIGIBLE	NEGLIGIBLE	MISSION
ITERNAL LEAK	0.09×10^{-6}	N/A	$R_M = e^{-\lambda t}$ $t = 8760$ HR $\lambda = 0.01 \times 10^{-6}$
ITERNAL LEAK	0.09×10^{-6}	N/A	$R_M = e^{-0.01 \times 10^{-6} \times 8760}$
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE	$= 0.9999124038$
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE	TOTAL FS-4
AIL TO OPEN	0.10×10^{-6}	N/A	$R_{FS4} = R_S R_M$ $= 0.9999122139$
EAK	NEGLIGIBLE	NEGLIGIBLE	
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE	
AIL TO LATCH	NEGLIGIBLE	NEGLIGIBLE	

FIGURE 5.2.3-2 RELIABILITY OF FEED SYSTEM FS-4

FOLDOUT FRAME

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

FUNCTIONAL SCHEMATIC	COMPONENT	RELIABILITY SCHEMATIC	FAILURE MC
	1) N_2 FILL & DRAIN VALVE	1	LEAK
	2) TANKS	2	LEAK
	3) N_2H_4 FILL & DRAIN VALVE	3	LEAK
	4) PRESSURE TRANSDUCER	4	CASE LEAK
		4	CONNECT. L
	5) NORMALLY CLOSED VALVE (ORDNANCE)	5	FAIL TO FIR
		5	FAIL TO OPE
		5	INTERNAL L
	6) TEST PORT	6	LEAK
	7) FILTER	7	LEAK, CLOG
	8) LATCHING VALVE (OPEN AT LAUNCH)	8	FAIL TO LAT
	9) LATCHING VALVE (OPEN AT LAUNCH)	9	FAIL TO LAT

...LURE MODES	FAILURE RATES START	FAILURE RATES MISSION
K	NEGLIGIBLE	NEGLIGIBLE
K	NEGLIGIBLE	NEGLIGIBLE
K	NEGLIGIBLE	NEGLIGIBLE
E LEAK	NEGLIGIBLE	NEGLIGIBLE
NECT. LEAK	0.01×10^{-6}	0.01×10^{-6}
. TO FIRE	10×10^{-6}	N/A
. TO OPEN	NEGLIGIBLE	N/A
RNAL LEAK	NEGLIGIBLE	N/A
K	NEGLIGIBLE	NEGLIGIBLE
K, CLOG	NEGLIGIBLE	NEGLIGIBLE
. TO LATCH	NEGLIGIBLE	NEGLIGIBLE
. TO LATCH	NEGLIGIBLE	NEGLIGIBLE

MATHEMATICAL MODEL

<u>START</u>

$$R_S = e^{-\lambda t}$$
$$t = 1 \text{ HR}$$
$$\lambda = (0.01 + 0.01)\,10^{-6}$$

NOTE REDUNDANCY OF INITIATORS IN (5) MAKES SECOND TERM NEGLIGIBLE

$$R_S = e^{-0.01 \times 10^{-6}}$$
$$= 0.9999999799$$

<u>MISSION</u>

$$R_M = e^{-\lambda t}$$
$$t = 8760 \text{ HR}$$
$$\lambda = 0.01 \times 10$$

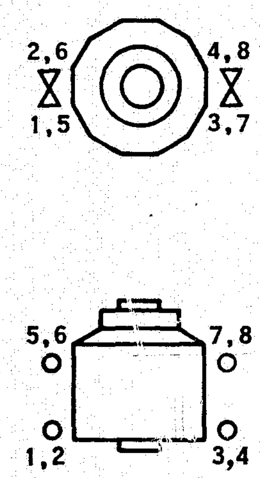
$$R_M = e^{-0.01 \times 10^{-6} \times 8760}$$
$$= 0.9999124038$$

<u>TOTAL FS-8</u>

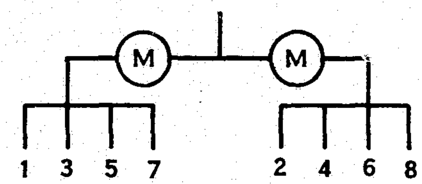
$$R_{FS8} = 0.9999123837$$

FIGURE 5.2.3-3. RELIABILITY OF FEED SYSTEM FS-8

ENGINE PLACEMENT



MANIFOLDING ARRANGEMENT



FUNCTIONAL ARRANGEMENT

MANEUVER	NORMAL MODE	DEGRADED MODE
MIDCOURSE /ORBITAL	1,3,5,7	2,4,6,8;
ACS	1,6, 2,5	4,6 2,8
+ SPIN	2,3	2 6
–SPIN	1,4	4,8

RELIABILITY SCHEMATIC

REDUNDANT ENGINE ARRAY — ENGINE FUEL CONNECTIONS — MANIFOLD VALVE

ENGINE FUEL CONNECTIONS: EXTERNAL LEAK
$\lambda = 0.01 \times 10^{-6}$
(PER ENGINE)

MANIFOLD VALVE: FAILURE TO CLOSE AND OPEN
$\lambda = 0.2 \times 10^{-6}$
(PER VALVE)

MATHEMATICAL MODELS

REDUNDANT ENGINES

$$R_E = R^8 + 8R^7Q_C + 8R^7Q_O + 20R^6Q^2_C + 12R^6Q^2_O + 24R^6Q_OQ_C$$

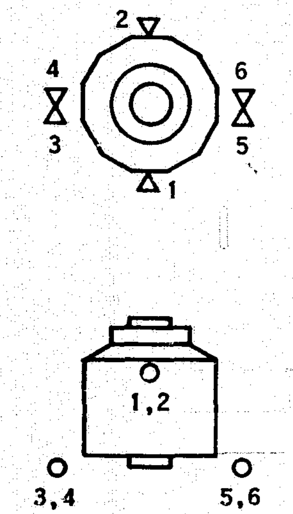
TOTAL CONCEPT

$$R_{P5} = R_E R_F R_M$$

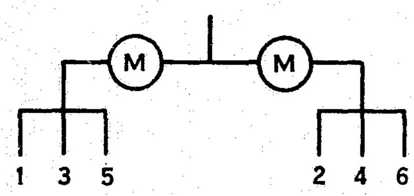
$$= 0.99578\ 40313$$

FIGURE 5.2.3–4. RELIABILITY OF ENGINE PLACEMENT CONCEPT P5

ENGINE PLACEMENT



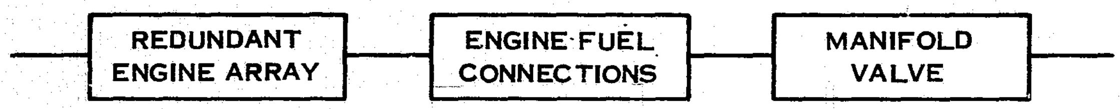
MANIFOLDING ARRANGEMENT



FUNCTIONAL ARRANGEMENT

MANEUVER	NORMAL MODE	DEGRADED MODE
MIDCOURSE /ORBITAL	1,3 & 5 2,4 & 6	1,3 & 5 2,4 & 6
ACS	1,4 & 6 2,3 & 5	1,3 & 5 2,4 & 6
+ SPIN	4 & 5	4 OR 5
−SPIN	3 & 6	3 OR 6

RELIABILITY SCHEMATIC



EXTERNAL LEAK
$\lambda = 0.01 \times 10^{-6}$
(PER ENGINE)

FAILURE TO CLOSE AND OPEN
$\lambda = 0.2 \times 10^{-6}$
(PER VALVE)

MATHEMATICAL MODELS

REDUNDANT ENGINES

$$R_E = R^6 + 6R^5Q_C + 6R^5Q_O + 6R^4Q^2{}_C + 6R^4Q^2{}_O + 12R^4Q_OQ_C$$

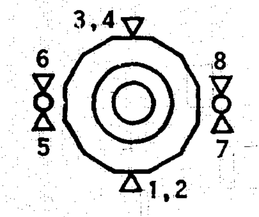
TOTAL CONCEPT

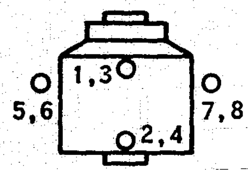
$$R_{P7} = R_E\, R_F\, R_M$$

$$= 0.99596\ 72491$$

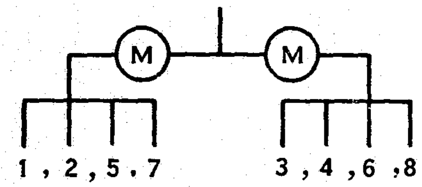
FIGURE 5.2.3–5. RELIABILITY OF ENGINE PLACEMENT CONCEPT P7

ENGINE PLACEMENT





MANIFOLDING ARRANGEMENT



FUNCTIONAL ARRANGEMENT

MANEUVER	NORMAL MODE	DEGRADED MODE
MIDCOURSE /ORBITAL	1&2 3&4	3 & 4 1 & 2
ACS	1&4 2&3	1& 2 3&4
+SPIN	6 & 7	6 OR 7
−SPIN	5 & 8	5 OR 8

RELIABILITY SCHEMATIC

EXTERNAL LEAK
$\lambda = 0.01 \times 10^{-6}$
(PER ENGINE)

FAILURE TO CLOSE AND OPEN
$\lambda = 0.2 \times 10^{-6}$
(PER VALVE)

MATHEMATICAL MODELS

REDUNDANT ENGINES

$$R_E = R^8 + 8R^7Q_C + 8R^7Q_O + 21R^6Q^2{}_C + 12R^6Q^2{}_O + 23R^6Q_OQ_C$$

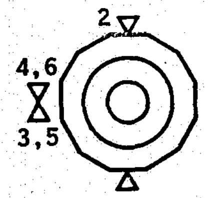
TOTAL CONCEPT

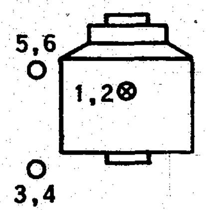
$$R_{P10} = R_E\, R_F\, R_M$$

$$= 0.99578\ 38534$$

FIGURE 5.2.3-6. RELIABILITY OF ENGINE PLACEMENT CONCEPT P10

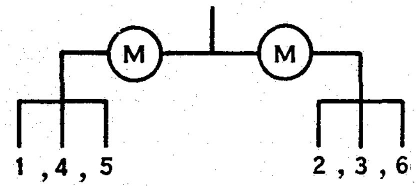
ENGINE PLACEMENT





SWIVEL ENGINES

MANIFOLDING ARRANGEMENT



FUNCTIONAL ARRANGEMENT

MANEUVER	NORMAL MODE	DEGRADED MODE
MIDCOURSE /ORBITAL	1	2
ACS	4&5	3&6
+ SPIN	4&6	6 OR 4
–SPIN	3&5	5 OR 3

RELIABILITY SCHEMATIC

REDUNDANT ENGINE ARRAY — ENGINE FUEL CONNECTIONS — MANIFOLD VALVE — PIVOT ACTUATOR

ENGINE FUEL CONNECTIONS: EXTERNAL LEAK $\lambda = 0.01 \times 10^{-6}$ (PER ENGINE)

MANIFOLD VALVE: FAILURE TO CLOSE AND OPEN $\lambda = 0.2 \times 10^{-6}$ (PER VALVE)

PIVOT ACTUATOR: INOPERATIVE $\lambda = 0.4 \times 10^{-6}$ (PER ACTUATOR)

MATHEMATICAL MODELS

REDUNDANT ENGINES

$$R_E = R^6 + 6R^5Q_C + 6R^5Q_O + 10R^4Q^2_C + 6R^4Q^2_O + 12R^4Q_OQ_C$$

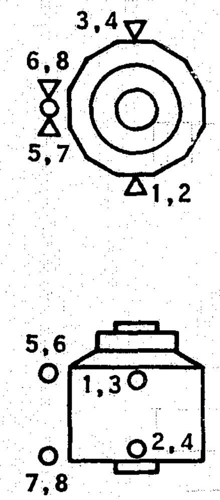
TOTAL CONCEPT

$$R_{P12} = R_E R_F R_M R_A$$

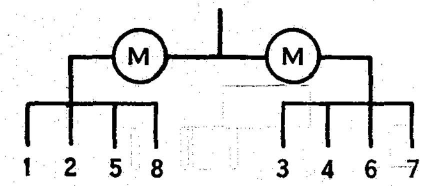
$$= 0.99310\ 33454$$

FIGURE 5.2.3–7. RELIABILITY OF ENGINE PLACEMENT CONCEPT P12

ENGINE PLACEMENT



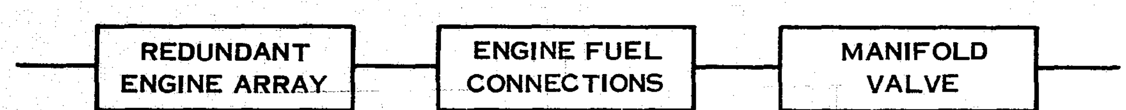
MANIFOLDING ARRANGEMENT



FUNCTIONAL ARRANGEMENT

MANEUVER	NORMAL MODE	DEGRADED MODE
MIDCOURSE /ORBITAL	1&2	3&4
ACS	5&8	6&7
+ SPIN	6&8	8 OR 6
–SPIN	5&7	5 OR 7

RELIABILITY SCHEMATIC



EXTERNAL LEAK
$\lambda = 0.01 \times 10^{-6}$
(PER ENGINE)

FAILURE TO CLOSE AND OPEN
$\lambda = 0.2 \times 10^{-6}$
(PER VALVE)

MATHEMATICAL MODELS

REDUNDANT ENGINES

$$R_E = R^8 + 8R^7Q_C + 8R^7Q_O + 20R^6Q^2{}_C + 12R^6Q^2{}_O + 24R^6Q_OQ_C$$

TOTAL CONCEPT

$$R_{P12} = R_E\, R_F\, R_M$$

$$= 0.99578\ 40313$$

FIGURE 5.2.3–8. RELIABILITY OF ENGINE PLACEMENT CONCEPT P13

5.3 Preflight Operations

Subsystem preflight operations include all of the inspections, tests and servicing operations which the propulsion subsystem is subjected to from the time the subsystem is source acceptance tested at the Propulsion System Subcontractor's Facility to the launch event. Figure 5.3.0-1 is a flow chart which illustrates the sequence of events during preflight operations. The key events are discussed below.

5.3.1 SUBSYSTEM ACCEPTANCE TESTING:

Prior to delivery, the subsystem will be subjected to tests to formally verify the satisfactory completion of the manufacturing and assembly phase. The objective here is to verify interface control, and to demonstrate that equipment performance was not degraded between the time it was acceptance tested at the component level, and then integrated into the subsystem. Examination of product includes such checks as visual and dimensional inspections, alignment of the engine thrust chambers, and dry weight measurement. Electrical checks include continuity and polarity checks as well as insulation resistance and circuit resistance checks. The subsystem is then proof pressure tested followed by internal leakage tests of the engine valves and latching solenoid valves, external leak checks on the system, and finally a calibration check of the pressure transducer. The system leakage check is performed by pressurizing the tankage and plumbing through the fill ports and test port with a nitrogen-helium mix. A helium mass spectrometer is then used to detect leakage of lines, components and connections. For an internal leak check of the propellant valves, dry nitrogen is introduced through the system test port and the leakage is measured at the engine nozzle exit by a standard liquid displacement method. A schematic for internal leakage testing of the engine propellant valves is shown in Figure 5.3.1-1. Since the propellant valve on the IDCSP/A engine utilizes dual series seats, it is desirable to verify leakage of each seat separately. The upstream seat leakage can be verified by electrically actuating the valve to a pre-determined value of current which will move the torque motor flapper sufficiently to unseat the downstream seat. With nitrogen pressure applied to the test port, leakage at the upstream seat can be measured. To measure leakage of the downstream seat, pressure is trapped between the two seats and leakage of this trapped gas is measured. This technique is presently used to verify individual seat leakage of the valves for the Intelsat IV program which are similar to the IDCSP/A valves. The calibration check of the pressure transducer is performed by pressurizing the system upstream of the isolation valves with dry nitrogen and comparing the transducer output to the monitored supply input. This test is performed at 3 different steady state pressures and the results compared with the transducer component acceptance test data.

The procedures for leakage checks of the latching solenoid valves are different for each of the 3 candidate feed system concepts. For the FS-2 feed system the fill and drain valve is used to pressurize the system upstream of the latching valves, and the system test port is used to sense leakage downstream of the valves. The normally

5.3.1 (Continued)

closed latching valves on the FS-4 feed system are tested in this same manner, however, the normally open latching valve on the FS-4 system require different procedures. First the valves are actuated to the closed position and the test port is used to pressurize the system. With the propellant valves open, and plugs inserted in each of the engine nozzles, the leakage is sensed downstream of the thrust chambers. Since the FS-8 feed system is similar to the FS-4 feed system in that they both utilize normally open latching valves located downstream of the system test port, the same procedures for testing these latching valves apply.

5.3.2 PROPULSION SUBSYSTEM INSTALLATION AND ALIGNMENT:

In order to simplify installation, and to minimize potential interface problems, the propulsion subsystem is designed to meet the following objectives:

- Simple accessible mounting features
- Non-interchangeable interface connections
- Ease of engine alignment
- Accessible fill and drain ports for ease of propellant servicing
- Minimization of mechanical interfaces

The propulsion subsystem is designed as an integral self-contained system which is shipped to GSFC as a finished product for installation into the spacecraft with no further fabrication required except for the subsystem/vehicle interface connections, and assembly of the engine modules. Subsystem to vehicle interfaces have been minimized by providing modules and panels for component mounting. All mechanical interfaces are designed to be bolted to a mating bracket or support on the vehicle.

The subsystem is assembled to a handling fixture at the Propulsion Subcontractor's Facility during the manufacturing/assembly phase. This fixture is designed to provide support at points which do not interfere with the assembly of the subsystem into the vehicle. This fixture provides for inspection of the various interfaces prior to shipment, and serves as a shipping fixture for transport of the subsystem to GSFC.

This method of subsystem support allows for "dropping" the propulsion subsystem into the vehicle, and building up the various vehicle support members if required. In the event a bulkhead or substructure separates the upper and lower areas of the vehicle, and is present prior to propulsion subsystem installation, propellant lines to engines mounted on the lower spacecraft structure can be routed through clearance holes in the bulkhead. The engines are then mounted to the spacecraft and the propellant line mechanically connected. The handling fixture considered consists of a main support column from which are extended removable radial supports to each of the separate interface stations. Each module, tank, or panel is mounted to a radial support through auxiliary mounting points which do not interfere with the vehicle mounting

5.3.2 (Continued)

interfaces. After each module or component is mounted to the appropriate interface, the radial handling fixture support to that item can be removed.

The critical interfaces during installation are the engine locations where it is necessary to align the thrust chamber nozzles relative to the vehicle center of gravity. Engines forming a module, or individually mounted engines, are geometrically aligned to the engine bracket/spacecraft interface during fabrication. Verification of engine alignment relative to the spacecraft can be performed through the utilization of a transit and reticle quadrant type discs attached to each engine nozzle. A fixture can be manufactured which will pick up the nozzle geometric centerline (defined by the throat and exit plane diameters) with the reticle disc mounted perpendicular to the fixture axis.

Alignment verification relative to the spacecraft is then achieved by sighting upon the reticle disc and adjusting the engine as required.

After the propulsion subsystem is installed into the spacecraft, electrical checks are performed to verify continuity and polarity of the electrical power circuitry, and the telemetry circuitry between the spacecraft and the propulsion subsystem. In addition, an external leakage check of the entire system, and an internal leakage test of the engine valves, as well as a calibration check on the pressure transducers, are performed. The procedures for these checks are the same as previously described.

5.3.3 LAUNCH BASE CHECKOUT AND SERVICING:

Checkout at the launch base is required to verify that the subsystem was not degraded during transport. This is performed prior to propellant and pressurant loading and includes subsystem external leakage checks, engine valve internal leakage checks and electrical continuity checks.

To load the system with propellant, the pressurant and propellant ground service lines of a propellant servicing cart are connected to the system fill and drain valves and the system is evacuated through the pressurant side. Propellant is then metered through the servicing cart until the required quantity of propellant has been loaded.

Before loading propellant, the propulsion subsystem is purged with N_2 and kept at a low pressure to exclude air and water vapor from the system. The propellant fill valve, which has the capability of being shut off with the fill line, and after the fill line is removed the fill valve inlet port is capped. The tanks are then charged to correct system pressure with gaseous nitrogen with a trace of helium to facilitate leak detection. The ground service line is then removed and the fill valve capped. Once the system has been charged with propellant and pressurant, the wet hold capability is such that minimal functional checks are required.

5.3.3, (Continued)

The frequency of these functional checks is dependent on the requirements imposed on the system for wet hold capability and the expected performance levels of the components used. The propulsion subsystem is designed to contain propellant and pressurant for periods greater than the mission flight and normal launch pad time spans, therefore, propellant off loading is unnecessary unless the mission is to be aborted for a significant time period or there is an anomoly with the propulsion subsystem. The locations of the propellant and pressurant fill lines and valves have been carefully considered to simplify the draining procedure in case of a mission abort. To drain the system, a bleed line from the ground service equipment is first connected to the pressurant vent valve. A shutoff valve in the ground service bleed system line is then opened and the tank pressure is bled to a low value. The ground service propellant drain system is then connected to the propellant drain valve and the propellant drained by gravity feed with low pressure assist. The subsystem is purged with nitrogen, flushed with water, then isopropyl alcohol, purged with nitrogen and then vacuum dried. The vacuum drying system provides for evacuating the system through a cold trap to remove propellant vapors from the line venting to the launch servicing cart or the vapors exhausting from the propulsion subsystem can be dumped, by way of service line extensions, into the launch pad underground scrubbing facilities, if available, which are normally used to eliminate other vehicle propellant vapors. If the subsystem is not immediately recharged with propellant and pressurant, a low positive pressure of GN_2 should be loaded for the storage period.

5.3.4 Ground Support Equipment

The requirements for the ground support equipment arise from the functions that must be performed on the propulsion subsystem from the point of acceptance testing through to installation into the spacecraft and finally launch base checkout and servicing. A description of these functions is covered in the previous sections of "Preflight Operations" and the requirements are listed as follows:

- Perform pressure transducer calibration
- Perform electrical checks on all electrical components
- Perform subsystem external leakage checks
- Perform internal leakage checks of thrust chamber valves (both seats)
- Propellant loading and off loading
- Pressurant loading and off loading
- GN_2 purging and vacuum drying
- Propellant vapor scrubbing
- System flushing (water and isopropyl alcohol)
- Propellant conditioning

The schematic shown in Figure 5.3.4-1 shows a typical propellant servicer cart used to perform the propellant and pressurant servicing, flushing, drying, and leak check operations. This system, in addition to all the electrical instrumentation needed to

5.3.4 (Continued)

carry out transducer calibration, electrical checks, and propellant valve seat leakage checks, will comprise the total ground support system. The servicer features straight manual operation which incorporates simplicity and reliability. The unit is semi-self contained, requiring only a high pressure nitrogen supply and external voltage source. The unit has provisions which permit hoisting the cart up the gantry so that the cart can be used on the same level as the spacecraft. Hoses and cables normally remain attached to the servicer when not in use and are stored at one end of the unit, and the controls which are most frequently used are grouped on a control panel for ease of operation. The cart consists of five tanks, hydrazine, water, alcohol, nitrogen and drain. With the exception of the nitrogen tank, each tank includes a level or weight measurement of contents, GN_2 pressurization valve, vent valve, and pressure gages. Fill valves and filters for 5 microns nominal/10 microns absolute are provided on each holding tank. The nitrogen distribution system provides GN_2 for load cell use and general pressurization. When loading the propellant, hydrazine is circulated up the spacecraft interface valve until lines are bubble-free (see return line bubble indicator), then the return line is shut off, the weight tank load noted, and the transfer started by opening the spacecraft fill-drain valve. Because of the bottom location of the propellant manifold of the propulsion subsystem, N_2 is vented to the top of the propellant tanks and out the servicing line. This scheme assures bubble free N_2H_4 loading. The propulsion subsystem can be flow flushed (H_2O and alcohol) and propellant conditioned by connecting the ground support equipment (GSE) return line to the subsystem pressurant fill valve and the GSE fill line to the subsystem fuel fill line. This completes a continuous loop into and out of the subsystem. Fuel drainage and flush fluids will be returned to a common drain tank for later disposal. The vacuum system provides for evacuating the tanks and lines through a cold trap to remove vapors and completely dry the subsystem.

FOLDOUT FRAME

SUBSYSTEM ACCEPTANCE TESTING (PROPULSION SUBCONTRACTORS FACILITY)

- EXAMINATION OF PRODUCT
- ALIGNMENT OF ENGINES
- ELECTRICAL CHECKS
- PROOF PRESSURE OF SYSTEM
- INTERNAL LEAKAGE
- EXTERNAL LEAKAGE
- TRANSDUCER CALIBRATION
- REVIEW OF COMPONENT ACCEPTANCE TEST DATA
- VIBRATION
- THERMAL VACUUM
- CONTAMINATION CHECK
- FUEL COMPATABILITY
- MASS PROPERTIES

→

INSTALLATION INTO SPACECRAFT (GSFC)

- SPACECRAFT/SUBSYSTEM INTERFACE CHECKS
  - MECHANICAL
  - ELECTRICAL
- ALIGNMENT
- ELECTRICAL CHECKS
- INTERNAL LEAKAGE
- EXTERNAL LEAKAGE
- TRANSDUCER CALIBRATION
- PROOF PRESSURE

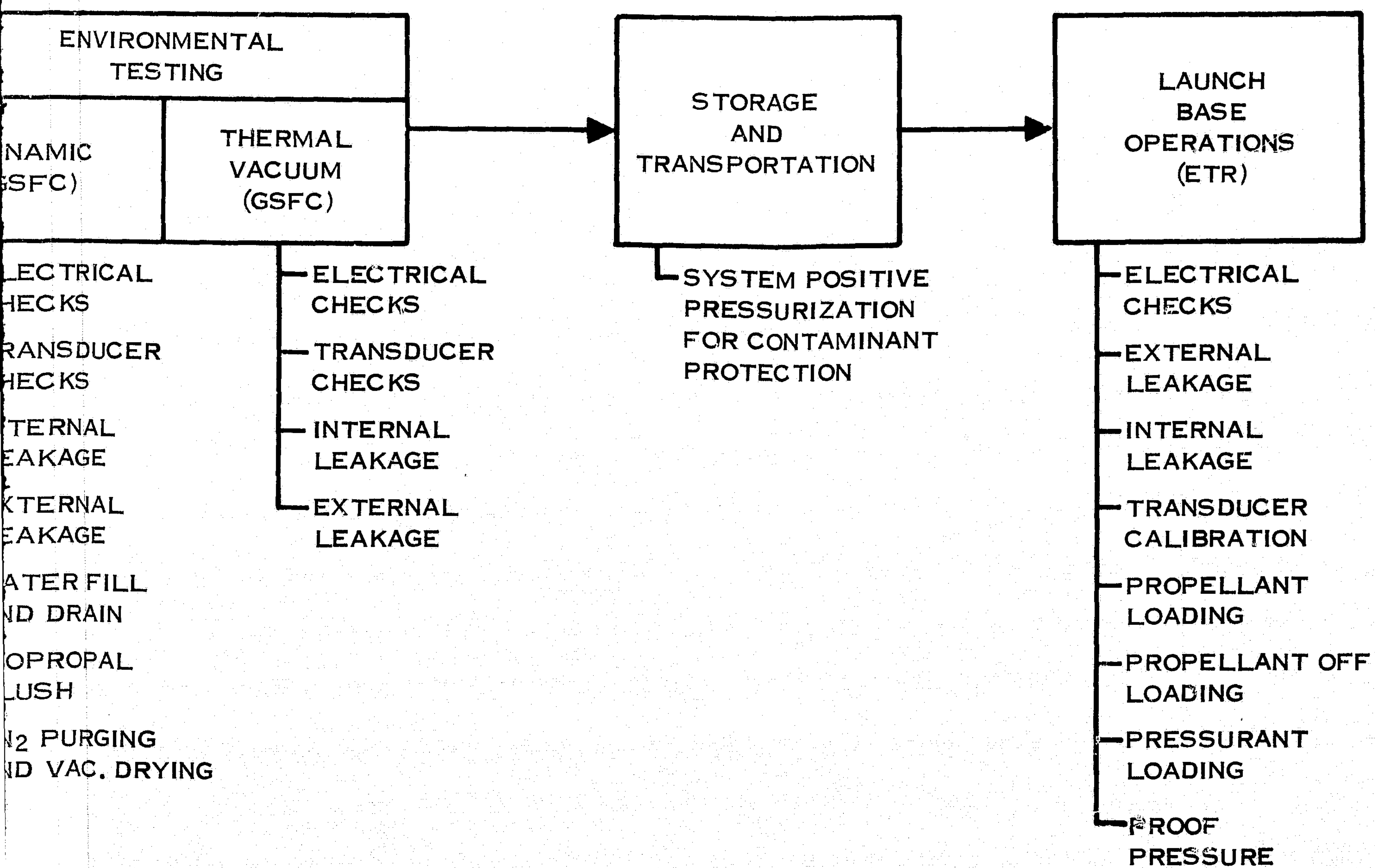
→

ENVIRONM[...] TEST[...]

DYNAMIC (GSFC)

- ELECTRICAL CHECKS
- TRANSDUCER CHECKS
- INTERNAL LEAKAGE
- EXTERNAL LEAKAGE
- WATER FILL AND DRAIN
- ISOPROPAL FLUSH
- GN_2 PURGING AND VAC. DRYI[...]

FIGURE 5.3.0–1 PROPULSION SUBSYSTEM OP[...]



3SYSTEM OPERATIONAL SEQUENCE — PRE-FLIGHT

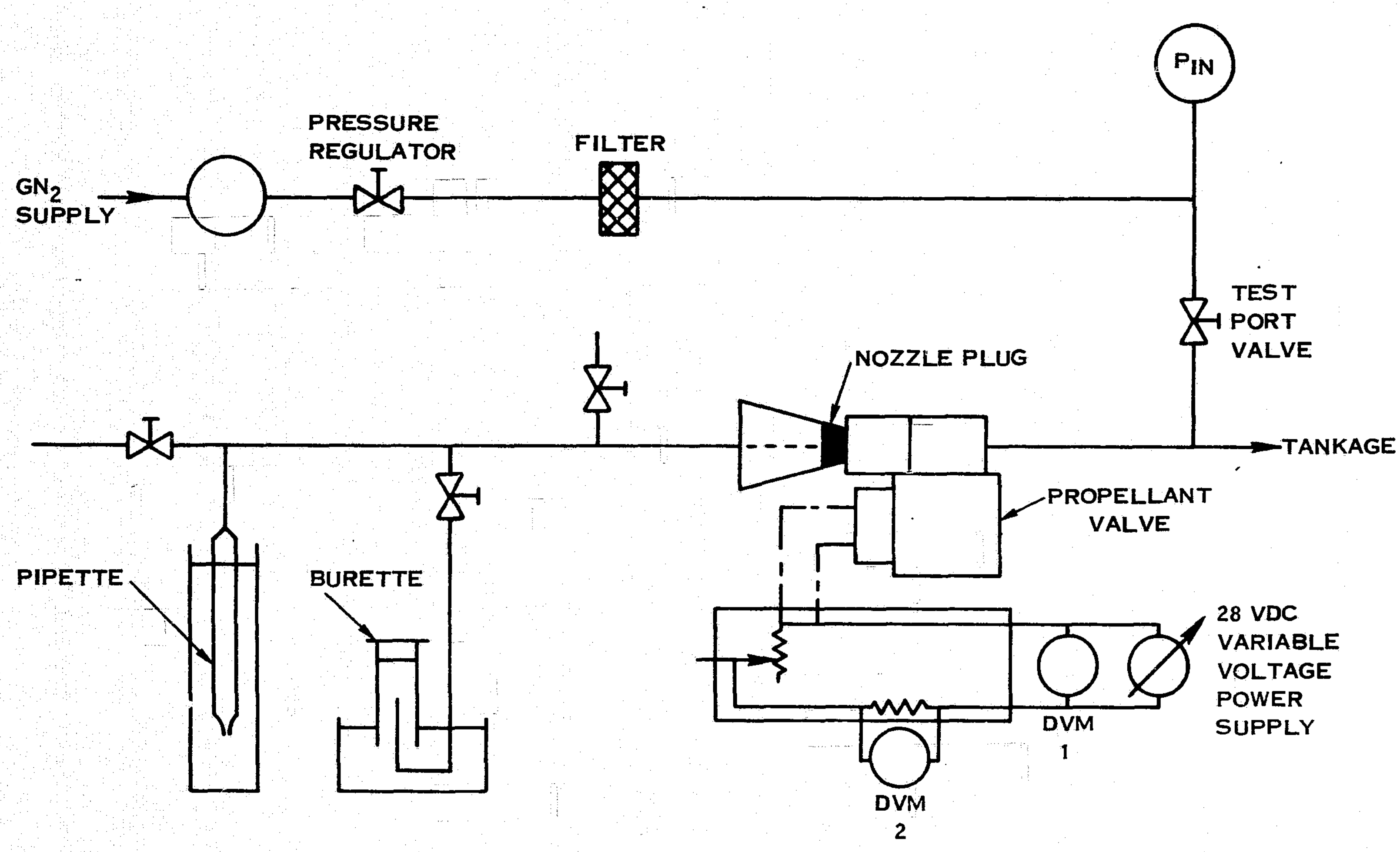
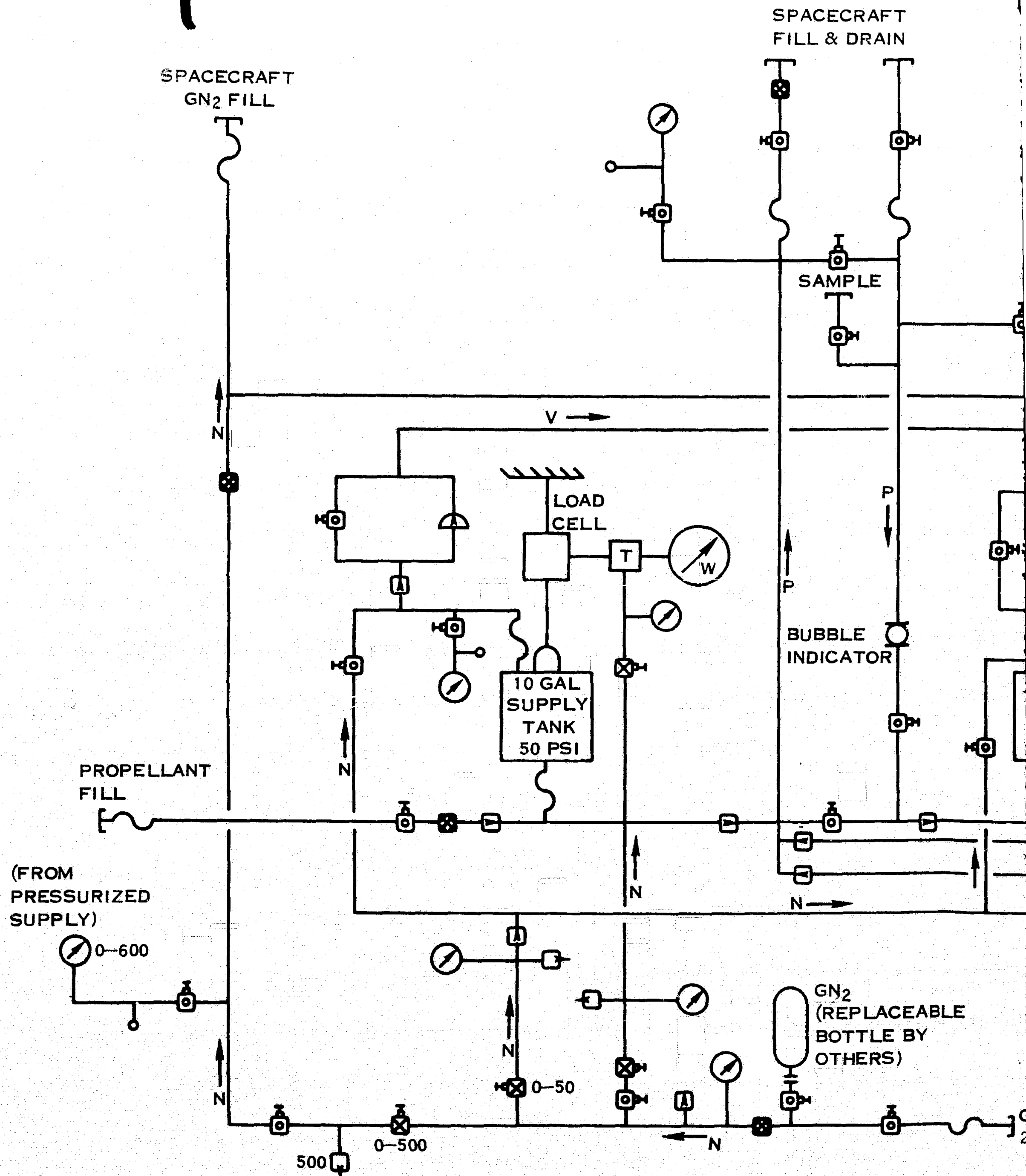


FIGURE 5.3.1–1 INTERNAL LEAKAGE TEST SCHEMATIC – ENGINE VALVE

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

FOLDOUT FRAME 1



FOLDOUT FRAME

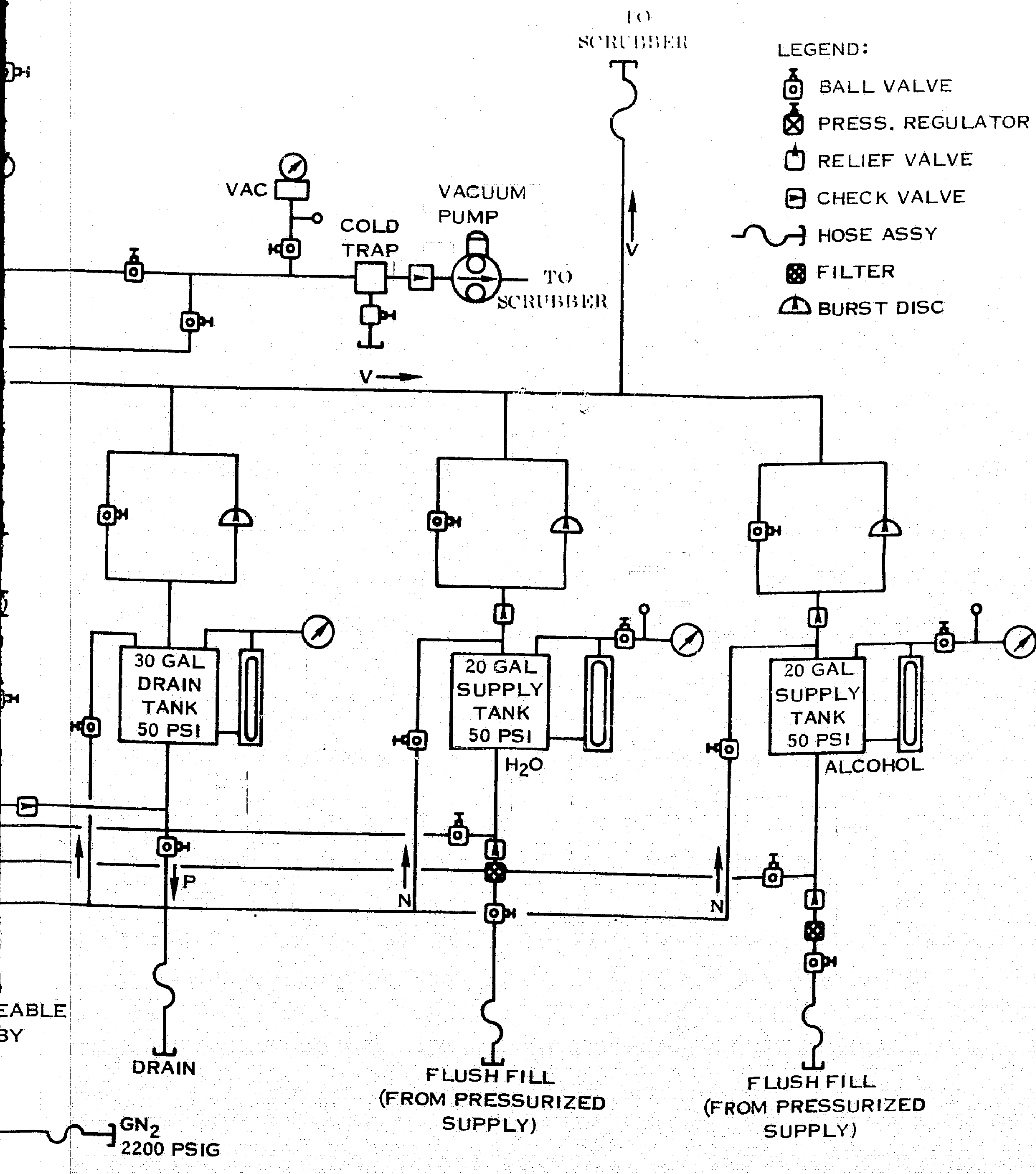


FIGURE 5.3.4-1. PROPELLANT SERVICING CART

5.4 FLIGHT OPERATIONS

In order to present a more detailed description of the flight operation procedures, and to describe the consequences of the operational constraints placed upon this system by the interaction of the error analysis and the accuracy requirements, Flight Operations Chartshave been prepared for both the Probe and Orbiter missions. These charts are included here as Table 5.4-I and 5.4-II. The P-5 candidate system with the FS-2 feed system was chosen for illustrative purposes. The basis for this chart is the Flight Operations Sheets which in effect generalize the technique required to perform certain maneuvers.

In preparing this chart it was assumed that each engine would have a bed temperature sensor and the sensor output would be telemetered to the ground station where it would be utilized for determining expected engine performance for each maneuver. In addition, it was assumed that indicators which tell the position of the latching solenoid valves would be part of the spacecraft's instrumentation. Since required thrusting to accomplish desired maneuvers will be determined at the ground command station only the positions in the spacecraft's reference system will be telemetered, therefore, resulting velocity vectors will be determined at the ground station.

During both the Probe and the Orbiter missions there are long periods of time between series of maneuvers. The shortest of these is 5 days which occurs between the event "Orient for Cruise" which occurs less than an hour after Liftoff, and the "Orient for 1st Midcourse Correction". Since the latching valves could protect against a serious gradual loss of propellant due to downstream leakage in any one engine, it is assumed that the latching solenoid valves will be closed during any time lapse equal to or greater than 5 days.

The symbols used in the Flight Operations Charts are defined below.

TABLE OF SYMBOLS

P_T	-	hydrazine tank pressure
T_T	-	hydrazine temperature
T_{Bn}	-	thruster bed temperature of engine "n"
T_{on}	-	thruster on-time
$\vec{\alpha}$	-	angular orientation in reference coordinate system
$\vec{\Delta x}$	-	displacement vector
$\vec{\Delta V}$	-	velocity vector
$\Delta\alpha$	-	precession angle change
ΔV	-	magnitude of velocity change
ΔN	-	spin speed change
$d\alpha$	-	precession angle error or perturbation
dV	-	magnitude of velocity error or perturbation
dN	-	spin speed error or perturbation
EPW	-	electrical pulse width

TABLE 5.4-I
FLIGHT OPERATIONS CHART - ORBITER MISSION
SYSTEM: I (P-5/FS-2)

EVENT	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTIONS	ERRORS
1) Liftoff	-	-	-	-	-
2) 3rd Stage Burnout	-	-	-	-	-
3) Despin $\Delta N = -49$ RPM	(P_T, T_T, T_B 5, 8, N Latch Valve Status) * *(Needed for Maneuver) Prop. System Checkout T_B (All), $\vec{\alpha}$, $\vec{x}$	ΔN, T_{on}	Open Both Latch Valves Fire 5 and 8	All telemetered measurements will be obtained and reviewed in order to determine system's status even though not all data need be known for this maneuver.	$dV = .0015$ m/sec $d\alpha = .125°$ $dN = 3.13$ RPM
4) Erect Booms	-	-	-	-	-
5) Spin Up to Cruise a) $\Delta N = +1$ RPM + Despin Error	P_T, T_T, $T_{B2,3}$, N	ΔN, T_{on}	Fire: 2 and 3	It is assumed that this will also be used to trim any despin	$dN = .265$ RPM
b) Spin Trim $\Delta N = +0-.3$ RPM	P_T, T_T, $T_{B\,1,2,3,4}$, N	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3	-	$dN = .019$ RPM
6) Separation from 3rd Stage	-	-	-	-	-
7) Spin Control a) $\Delta N = +10$ RPM	P_T, T_T, $T_{B2,3}$	ΔN, T_{on}	Fire: 2 and 3	It is assumed that the separation from the third stage caused a change in spin and the purpose of this maneuver is to trim this out.	$dN = .64$ RPM
b) Spin Trim $\Delta N = \pm 0 - .64$ RPM	P_T, T_T, $T_{B\,1,2,3,4}$, N	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3	-	$dN = .041$ RPM
8) Spin Control a) $\Delta N = -9.7$ RPM	P_T, T_T, $T_{B1,4}$, N	ΔN, T_{on}	Fire: 1 and 4	-	$dN = .62$ RPM
b) Spin Trim $\Delta N = \pm 0-.62$ RPM	P_T, T_T, $T_{B\,1,2,3,4}$, N	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3	-	$dN = .040$ RPM
9) Orient for Cruise a) $\Delta\alpha = 22°$	P_T, T_T, $T_{B\,1,6}$ N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. Pulses	Fire: 1 and 6	This maneuver will be done in five steps in order to conserve fuel while zeroing in on the desired $\Delta\alpha = 90°$. (9a), c), e), g).	$dV = .0064$ m/sec $d\alpha = 1.0°$ $dN = .30$ RPM
b) Spin Control $\Delta N = \pm 0 - .30$ RPM Repeat three more times for Events 9c, 9d, 9e, 9f, 9g, 9h	P_T, T_T, $T_{B\,1,2,3,4}$, N	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	Because of spin change due to $\Delta\alpha$ maneuver it is necessary to trim the spin rate. (9b), d), f), h).	$dN = .019$ RPM
i) $\Delta\alpha$ Course Trim $\Delta\alpha = 2°$	P_T, T_T, $T_{B\,1,6}$ N, $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. Pulses	Fire: 1 and 6 Close Latching Valves After Firing	Latching valves are closed after maneuver because next maneuver is 5 days away.	$d\alpha = .22°$ for Two Thruster Operation $d\alpha = .11°$ for One Thruster Operation
10) Orient for 1st M/C a) ΔV Calibration Engines	P_T, T_T, $T_{B\,1,3,5,7}$, N $\vec{\alpha}$, $\vec{x}$, Latch Valve	EPW, Timing, No. Pulses	Open Both Latching Valves	Before the orientation both Δv clusters will be fired in order to determine the uncertainty in the	c) $d\alpha = .41°$ $dN = .12$ RPM

Repeat three more times for Events 9c, 9d, 9e, 9f, 9g, 9h				necessary to trim the spin rate. (9b), d), f), h).	dN = .019 RPM
i) $\Delta\alpha$ Course Trim $\Delta\alpha = 2°$	P_T, T_T, T_B 1, 6 N, $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. Pulses	Fire: 1 and 6 Close Latching Valves After Firing	Latching valves are closed after maneuver because next maneuver is 5 days away.	$d\alpha = .22°$ for Two Thruster Operation $d\alpha = .11°$ for One Thruster Operation
10) Orient for 1st M/C a) ΔV Calibration Engines 1,3,5,7	P_T, T_T, T_B 1,3,5,7,N $\vec{\alpha}$, $\vec{x}$, Latch Valve Status Post Firing N, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. Pulses Post Firing ΔN, $\Delta\vec{\alpha}$, $\Delta\vec{x}$, $\Delta\vec{v}$ $r_{e\alpha c}$, r_{enc}, $r_{e\alpha}$, r_{en}	Open Both Latching Valves Fire: 1,3,5,7	Before the orientation both Δv clusters will be fired in order to determine the uncertainty in the c.g. location relative to the thrust vector. ($r_{e\alpha}$ and r_{en}). It is assumed that the c.g. location is known with ±.10" parallel to the spin axis and ±.05" normal to it for this fuel loading.	c) $d\alpha = .41°$ dN = .12 RPM d) $d\alpha = .22°$ for Two Thruster $d\alpha = .11°$ for One Thruster dN = .0135 RPM
b) ΔV Calibration Engines 2,4,6,8	P_T, T_T, T_B 2,4,6,8,N, $\vec{\alpha}$, $\vec{x}$ Post Firing N, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses Post Firing ΔN, $\Delta\vec{\alpha}$, $\Delta\vec{x}$, $\Delta\vec{v}$ $r_{e\alpha c}$, r_{enc}, $r_{e\alpha}$, r_{en}	Fire: 2,4,6,8		
c) $\Delta\alpha = 9°$ Orient for 1st M/C	P_T, T_T, T_B 2,5 N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		
d) $\Delta\alpha = 1°$ Trim Orientation for 1st M/C	P_T, T_T, T_B 2,5 N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		
11) 1st Midcourse a) Correction Δv = 42 m/sec	P_T, T_T, T_B 1,3,5,7,N $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. Pulses	Fire: 1,3,5,7	Engines 1,3,5,7 will be used for first part of maneuver. Engines 2,4,6,8 used for second 42 m/sec.	dV = 2.5 m/sec $d\alpha$ = 1.9° dN = .30 RPM dN = .019 RPM
b) Spin Control ΔN = ±0-.3 RPM	P_T, T_T, T_B 1,2,3,4,N $\vec{\alpha}$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3		
Repeat Once for Events 11 c), d). Use Engines 2,4,6,8 for c).					
e) ΔV = 24 m/sec	P_T, T_T, T_B 1,3,5,7,N $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, and No. of Pulses Taking into Account Calcu. C.G. Shift	Fire: 1,3,5,7	Engines 1, 3, 5, 7 used.	dV = 1.43 m/sec % Error = 1.3% $d\alpha$ = 1.1° dN = .17 RPM
12) Reorient to Cruise a) $\Delta\alpha = 9°$	P_T, T_T, T_B 2,5,N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	It is assumed that the $d\alpha$ from event 11 will be trimmed out here.	$d\alpha$ = .41° dN = .12 RPM
b) $\Delta\alpha = 1°$ Trim Orientation for Cruise	P_T, T_T, T_B 2,5,N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		$d\alpha$ = .22° for Two Thrusters $d\alpha$ = .11° for One Thruster dN = .0135 RPM

FOLDOUT FRAME 2

SP 70R70-F

TABLE 5.4-I (CONTINUED)
FLIGHT OPERATIONS CHART - ORBITER MISSION
SYSTEM: I (P-5/FS-2)

EVENTS	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTIONS	ERRORS
13) Spin Control a) $\Delta N = +2.5$ RPM	$P_T, T_T, T_{B2,3}, N$	ΔN, T_{on}	Fire: 2 and 3	-	$d\alpha = .096°$ $dN = .16$ RPM
b) Spin Trim $\Delta N = \pm 0 - .16$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3	-	$dN = .01$ RPM
14) Spin Control a) $\Delta N = -2.5$ RPM	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	$d\alpha = .096°$ $dN = .16$ RPM
b) Spin Control $\Delta N = \pm 0 - .16$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$ $\bar{x}, \bar{\alpha}$	ΔN, T_{on}	Fire: 2,3 for $+\Delta N$ 1,4 for $-\Delta N$ Close Latching Valves After Firing		$dN = .01$ RPM
15) Orient for 2nd Midcourse Correction a) $\Delta\alpha = 9°$	$P_T, T_T, T_{B2,5}, N$ $\vec{\alpha}, \vec{x}$, Latching Valve Status	$\Delta\alpha$, EPW, Timing, No. of Pulses	Open Both Latching Valves Fire: 2 and 5	-	$d\alpha = .41°$ $dN = .12$ RPM
b) $\Delta\alpha = 1°$ Trim for 2nd M/C	$P_T, T_T, T_{B2,5}, N$ $\vec{\alpha}, \vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		$d\alpha = .11°$ For One Thruster $dN = .0135$ RPM
16) 2nd Midcourse Correction a) $\Delta V = 9$ m/sec	$P_T, T_T, T_{B2,4,6,8}, N$ $\vec{\alpha}, \vec{x}$	$\Delta\vec{V}$, EPW, Timing and No. of Pulses	Fire: 2,4,6,8	C.G. shift from calibration prior to event no. 10 is done with sufficient accuracy to make recalibration unnecessary	$dV = .54$ m/sec $d\alpha = .41°$ $dN = .064$ RPM
b) $\Delta V = 1$ m/sec	$P_T, T_T, T_{B2,4,6,8}, N$ $\vec{\alpha}, \vec{x}$	$\Delta\vec{V}$, EPW, Timing and No. of Pulses	Fire: 2,4,6,8	-	dV .092 m/sec % Error = 1.02% $d\alpha = .046°$ $dN = .0071$ RPM
17) Reorient to Cruise a) $\Delta\alpha = 9°$	$P_T, T_T, T_{B2,5}, N$ $\vec{\alpha}, \vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	$d\alpha = .41°$ $dN = .12$ RPM
b) $\Delta\alpha = 1°$ Trim Orientation for Cruise	$P_T, T_T, T_{B2,5}, N$ $\vec{\alpha}, \vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	$d\alpha = .11°$ for One Thruster $dN = .0135$ RPM
18) Spin Control a) $\Delta N = +5$ RPM	$P_T, T_T, T_{B2,3}, N$ $\vec{\alpha}, \vec{x}$	ΔN, T_{on}	Fire: 2 and 3	-	$dN = .32$ RPM
b) Spin Trim $\Delta N = \pm 0 - .32$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	$dN = .021$ RPM
19) Spin Control a) $\Delta N = -4.7$ RPM	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	$dN = .30$ RPM
b) Spin Trim $\Delta N = \pm 0 - .3$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$ $\bar{x}, \bar{\alpha}$	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3 Close Latching Valves After Firing		$dN = .019$ RPM
20) Orient for 3rd M/C	$P_T, T_T, T_{B2,5}, N$	$\Delta\alpha$, EPW, Timing,	Open Both Latching	-	$d\alpha = .41°$ $dN = .12$ RPM

b) Spin Trim $\Delta N = \pm 0 - .32$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN= .021 RPM
19) Spin Control a) $\Delta N = -4.7$ RPM	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	dN= .30 RPM
b) Spin Trim $\Delta N = \pm 0 - .3$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$ $\bar{x}, \bar{\alpha}$	ΔN, T_{on}	Fire: 1 and 4 or 2 and 3 Close Latching Valves After Firing		dN= .019 RPM
20) Orient for 3rd M/C a) $\alpha = 9°$	$P_T, T_T, T_{B2,5}, N$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Open Both Latching Valves Fire: 2 and 5	-	dα= .41° dN= .12 RPM
b) $\Delta\alpha = 1°$ Trim for 3rd M/C	$P_T, T_T, T_{B2,5}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα= .11° for One Thruster dN= .0135 RPM
21) 3rd Midcourse Correction a) $\Delta V = 1$ m/sec	$P_T, T_T, T_{B2,4,6,8}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\vec{V}$, EPW, Timing and No. of Pulses	Fire: 2,4,6,8	If this maneuver were done in one increment a 5.98% error in ΔV could result which is > the 5% allowed. Therefore it will be done in 2 steps. Only 21 pulses (total) are required for 2 m/sec ΔV.	dV= .0598 m/sec dα= .047° dN= .0071 RPM
b) $\Delta V = 1$ m/sec	$P_T, T_T, T_{B2,4,6,8}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\vec{V}$, EPW, Timing and No. of Pulses	Fire: 2,4,6,8	Number of pulses required is a whole number.	dV= .0598 m/sec % Error = 3% dα= .047° dN= .0071 RPM
22) Reorient to Cruise a) $\Delta\alpha = 9°$	$P_T, T_T, T_{B2,5}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα= .41° dN= .12 RPM
b) Trim Reorientation for Cruise $\Delta\alpha = 1°$	$P_T, T_T, T_{B2,5}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5 Close Latching Valves After Firing	-	dα= .11° for One Thruster dN= .0135 RPM
23) Reorient for Retrofit a) $\Delta\alpha = 22°$	$P_T, T_T, T_{B2,5}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Open Both Latching Valves Fire: 2 and 5	This maneuver will be done in five steps in order to conserve fuel while zeroing in on the desired Δα = 90° and correcting the spin perturbation.	dV= .0064 m/sec dα= 1.0° dN= .30 RPM
b) Spin Control $\Delta N = \pm 0 - .30$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	Because of spin change due to Δα maneuver it is necessary to trim the spin rate.	dN= .019 RPM
Repeat Three More Times for Events 23 c), d), e), f), g), h).					
i) Δα Course Trim $\Delta\alpha = 2°$	$P_T, T_T, T_{B2,5}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα= .11° for One Thruster
24) Retrofire for Orbit Transfer and Reorient a) $\Delta\alpha = 22°$	$P_T, T_T, T_{B3,8}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dV= .0064 m/sec dα= 1.0° dN= .30 RPM
b) Spin Control $\Delta N = \pm 0 - .3$ RPM d), f), h) Repeat Three More Times for Events c), d), e), f), g), h).	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 2 and 3		dN= .019 RPM
i) Δα Course Trim $\Delta\alpha = 2°$	$P_T, T_T, T_{B3,8}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dα= .11° for One Thruster

FOLDOUT FRAME 2

SP 07R70-F

TABLE 5.4-I (CONTINUED)
FLIGHT OPERATIONS CHART - ORBITER MISSION
SYSTEM: I (P-5/FS-2)

EVENTS	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTIONS	ERRORS
25) Spin Control a) $\Delta N = +5$ RPM	$P_T, T_T, T_{B2,3}, N$	ΔN, T_{on}	Fire: 2 and 3	-	dN = .32 RPM
b) Spin Trim $\Delta N = \pm 0 - .32$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .02 RPM
26) Spin Control a) $\Delta N = -5$ RPM	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	dN = .32 RPM
b) Spin Trim $\Delta N = \pm 0 - .32$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .02 RPM
27) Attitude Control a) Maintenance $\Delta\alpha = 20°$	$P_T, T_T, T_{B4,7}, N$ $\bar{\alpha}, \bar{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 4 and 7	This maneuver will be done in a total of six even increments.	dV = .025 m/sec $d\alpha$ = .91° dN = .27 RPM
b) Spin Control $\Delta N = \pm 0 - .27$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .017 RPM
Repeat four more times for a total $\Delta\alpha = 120°$. Events 27 c), d), e), f), g), h), i), j).					
k) $\Delta\alpha = 20°$				$d\alpha$ and dN error will be trimmed out during Events 28, 29, 30.	$d\alpha$ = .91° dN = .27 RPM
28) Orientation Trim $\Delta\alpha = 6°$	$P_T, T_T, T_{B4,7}, N$ $\bar{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 4 and 7	-	$d\alpha$ = .11° for One Thruster
29) Spin Control a) $\Delta N = +2.5$ RPM	$P_T, T_T, T_{B2,3}, N$	ΔN, T_{on}	Fire: 2 and 3	-	dN = .16 RPM
b) Spin Trim $\Delta N = \pm 0 - .27$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .01 RPM
30) Spin Control a) $\Delta N = -2.5$	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	dN = .16 RPM
b) Spin Trim $\Delta N = \pm 0 - .27$ RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .01 RPM

Spin Control a) ΔN = +2.5 RPM	$P_T, T_T, T_{B2,3}, N$	ΔN, T_{on}	Fire: 2 and 3	-	dN = .16 RPM
b) Spin Trim ΔN = ±0 - .27 RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .01 RPM
30) Spin Control a) ΔN = -2.5	$P_T, T_T, T_{B1,4}, N$	ΔN, T_{on}	Fire: 1 and 4	-	dN = .16 RPM
b) Spin Trim ΔN = ±0 - .27 RPM	$P_T, T_T, T_{B1,2,3,4}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3 Close Both Latching Valves After Firing	-	dN = .01 RPM
31) Orient for Periapsis Reduction a) Δα = 27.5°	$P_T, T_T, T_{B1,6}, N$, $\vec{\alpha}$, $\vec{x}$	Δα, EPW, Timing, No. of Pulses	Open Both Latching Valves Fire: 1 and 6 Valves	-	dV = .0132 m/sec dα = 1.25° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	$P_T, T_T, T_{B1,4,6,7}, N$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 6 and 7	-	dN = .019 RPM
c) Δα Trim Δα = 7.5°	$P_T, T_T, T_{B1,6}, N$, $\vec{\alpha}$, $\vec{x}$	Δα, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	dα = .17° for One Thruster dN = .082 RPM
* 32) Periapsis Reduction a) 100 KM ΔV = 12 m/sec	$P_T, T_T, T_{B1,3,5,7}, N$, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses	Fire: 1,3,5,7	-	dV = .72 m/sec dα = .26° dN = .025 RPM
b) ΔV = 1 m/sec	$P_T, T_T, T_{B1,3,5,7}, N$, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses	Fire: 1,3,5,7	-	dV = .102 m/sec % Error = .85%
c) Reorient to Orbit Mode Δα = 27.5°	$P_T, T_T, T_{B2,5}, N$, $\vec{\alpha}$, $\vec{x}$	Δα, EPW, Timing, No. Of Pulses	Fire: 2 and 5	-	dV = .0132 m/sec dα = 1.25° dN = .30 RPM
d) Spin Control ΔN = ±0 - .3 RPM	$P_T, T_T, T_{B1,4,6,7}, N$	ΔN, T_{on} if necessary	Fire: 1 and 4 or 6 and 7	-	dN = .019 RPM
e) Δα Trim Δα = 7.5°	$P_T, T_T, T_{B2,5}, N$, $\vec{\alpha}$, $\vec{x}$	Δα, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα = .17° for One Thruster dN = .082 RPM

* Events 31 and 32 - This series of events represent a 100 KM Periapsis reduction, for a total of 1500 KM repeat 15 times. Alternate thrusters according to table.

THRUSTERS

Δ PERIAPSIS	EVENT 31 a), c)	EVENT 31 b), 32 d)	EVENT 32 a), b)	EVENT 32 c), d)
100	1, 6	1 and 4 or 6 and 7	1, 3, 5, 7	2, 5
200	3, 8		2, 4, 6, 8	4, 7
300	1, 6		1, 3, 5, 7	2, 5
400	3, 8		2, 4, 6, 8	4, 7
500	1, 6		1, 3, 5, 7	2, 5
600	3, 8		2, 4, 6, 8	4, 7
700	1, 6		1, 3, 5, 7	2, 5
800	3, 8		2, 4, 6, 8	4, 7
900	1, 6		1, 3, 5, 7	2, 5
1000	3, 8		2, 4, 6, 8	4, 7
1100	1, 6		1, 3, 5, 7	2, 5
1200	3, 8		2, 4, 6, 8	4, 7
1300	1, 6		1, 3, 5, 7	2, 5
1400	3, 8		2, 4, 6, 8	4, 7
1500	1, 6	1 and 4 or 6 and 7	1, 3, 5, 7	2, 5

TABLE 5.4-II
FLIGHT OPERATIONS CHART - PROBE MISSION
SYSTEM: I (P-5/FS-2)

EVENT	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTIONS	ERRORS
1) Liftoff	-	-	-	-	-
2) 3rd Stage Burnout	-	-	-	-	-
3) Despin by Hydrazine System $\Delta N = -40$ RPM	P_T, T_T, $T_{B1,4}$, N Latch Valve Status (Needed for Maneuver) Propulsion System Checkout: T_B (All) $\vec{\alpha}$, $\vec{x}$	ΔN, T_{on}	Fire: 1 and 4	All telemetered data will be obtained and reviewed in order to determine system's status even though not all data will be needed for this maneuver. Since the next maneuver is a spin up to cruise error will be trimmed out there and not here.	$dV = .011$ m/sec $d\alpha = .040°$ $dN = 2.56$ RPM
4) Erect Booms	-	-	-	-	-
5) Spin Up to Cruise Valve a) $\Delta N = 4$ RPM	P_T, T_T, $T_{B2,3}$, N	ΔN, T_{on}	Fire: 2 and 3	Error due to Event 3 will be trimmed out here.	$d\alpha = .074°$ $dN = .42$ RPM
b) Spin Trim $\Delta N = \pm .42$ RPM	P_T, T_T, $T_{B1,2,3,4}$ N	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	In order to meet spin accuracy of ±.15 RPM a spin trim is required.	$dN = .027$ RPM
6) S/C Separation from 3rd Stage	-	-	-	-	-
7) Orient for Cruise a) $\Delta\alpha = 27.5°$	P_T, T_T, $T_{B1, 6}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 1 and 6	This maneuver will be done in four $\Delta\alpha$ steps.	$dV = .015$ m/sec $d\alpha = 1.25°$ $dN = .30$ RPM
b) Spin Control $\Delta N = \pm 0 - .3$ RPM	P_T, T_T, $T_{B1,2,3,4}$, N	Δ N, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	Trimming spin error to keep within .30 RPM requirement.	$dN = .019$ RPM
Repeat two times for Events 7c, 7d, 7e, and 7f.					
g) $\Delta\alpha = 5.0°$	P_T, T_T, $T_{B1,6}$, N $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	$d\alpha = .23°$ $dN = .055$ RPM
h) $\Delta\alpha$ Course Trim $\Delta\alpha = 2.5°$	P_T, T_T, $T_{B1, 6}$, N $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 1 and 6 Close Latching Valve After Firing	-	$d\alpha = .11°$ $dN = .027$ RPM
8) Orient for 1st M/C a) ΔV Calibration	P_T, T_T, $T_{B1,3,5,7}$, N, $\vec{\alpha}$, $\vec{x}$ Latching	EPW, Timing, No. of Pulses	Opening Both Latching Valves	Before orientation both ΔV clusters will be fired in order to determine the uncertainty in the C.G. location relative to the thrust	

	Events 7c, 7d, 7e, and 7f.					
g)	$\Delta\alpha = 5.0°$	P_T, T_T, $T_{B1,6}$, N, $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	$d\alpha$ = .23° dN = .055 RPM
h)	$\Delta\alpha$ Course Trim $\Delta\alpha$ = 2.5°	P_T, T_T, $T_{B1,6}$, N, $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 1 and 6 Close Latching Valve After Firing	-	$d\alpha$ = .11° dN = .027 RPM
8) a)	Orient for 1st M/C ΔV Calibration Engines 1,3,5,7	P_T, T_T, $T_{B1,3,5,7}$, N, $\vec{\alpha}$, $\vec{x}$ Latching Valve Status Post Firing N, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses Post Firing ΔN, $\Delta\vec{\alpha}$, $\Delta\vec{x}$, $\Delta\vec{V}$ $r_{e\alpha c}$, r_{enc}, $r_{e\alpha}$, r_{en}	Opening Both Latching Valves Fire: 1,3,5,7	Before orientation both ΔV clusters will be fired in order to determine the uncertainty in the C. G. location relative to the thrust vector. ($r_{e\alpha}$ and r_{en}). It is assumed that the C. G. location is known within ±.10" parallel to the spin axis and ±.05" normal to it for this fuel loading and the thrusters will be modulated to put the thrust vector through the calculated C. G.	
b)	ΔV Calibration Engines 2,4,6,8	P_T, T_T, $T_{B2,4,6,8}$, N Post Firing N, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses Post Firing ΔN, $\Delta\alpha$, Δx, ΔV $r_{e\alpha c}$, r_{enc}, $r_{e\alpha}$, r_{en}	Fire: 2,4,6,8		
c)	$\Delta\alpha$ = 9° Orient for 1st M/C	P_T, T_T, $T_{B2,5}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		$d\alpha$ = .41° dN = .098 RPM
d)	$\Delta\alpha$ = 1° Trim Orientation for 1st m/c	P_T, T_T, $T_{B2,5}$, N, , x ,	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5		$d\alpha$ = .096° dN = .0109 RPM
9) a)	1st Midcourse Correction ΔV = 50 m/sec	P_T, T_T, $T_{B1,3,5,7}$, N, $\vec{\alpha}$, $\vec{x}$	EPW, Timing, No. of Pulses	Fire: 1,3,5,7	Engines 1,3,5,7 will be used for first part of maneuver.	dV = 3.0 m/sec $d\alpha$ = 1.26° dN = .157 RPM
b)	Spin Control ΔN = ±0 - .157 RPM	P_T, T_T, $T_{B1,2,3,4}$, N	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	In order to keep the spin speed within limits at the end of the maneuver it will be trimmed here.	
c)	ΔV = 50 m/sec	P_T, T_T, $T_{B2,4,6,8}$, N	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking into Account C. G. Shift due to 9a).	Fire: 2,4,6,8	Engines 2,4,6,8 will be used for second part.	dV = 3.0 m/sec $d\alpha$ = 1.26° dN = .157 RPM
d)	ΔV = 8.0 m/sec	P_T, T_T, $T_{B2,4,6,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses	Fire: 2,4,6,8	-	dV = .48 m/sec % Error = .44% $d\alpha$ = .20° dN = .025 RPM
10) a)	Reorient for Cruise $\Delta\alpha$ = 27.5°	P_T, T_T, $T_{B2,5}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	This maneuver will be done in three steps.	$d\alpha$ = 1.25° dN = .30 RPM
b)	Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{b1,2,3,4}$, N	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	Trimming accumulated spin errors.	dN = .019 RPM
c)	$\Delta\alpha$ = 20°	P_T, T_T, $T_{B2,5}$, N, $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	$d\alpha$ = .90° dN = .22 RPM
d)	$\Delta\alpha$ Course Trim $\Delta\alpha$ = 2.5	P_T, T_T, $T_{B2,5}$, N, $\vec{\alpha}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 2 and 5 Close Latching Valves After Firing	-	$d\alpha$ = .11° dN = .027 RPM

FOLDOUT FRAME 2

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TABLE 5.4-II (CONTINUED)
FLIGHT OPERATIONS CHART - PROBE MISSION
SYSTEM: I (P-5/FS-2)

EVENT	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTION	ERRORS
11) Orient for 2nd M/C a) $\Delta\alpha = 18°$	P_T, T_T, $T_{B2,5}$, N $\vec{\alpha}$, $\vec{x}$, Latching Valve Status	$\Delta\alpha$, EPW, Timing, No. of Pulses	Open Latching Valves Fire: 2 and 5	This maneuver will be done in two steps.	$d\alpha$ .82° dN = .20 RPM
b) $\Delta\alpha$ Course Trim $\Delta\alpha = 2°$	P_T, T_T, $T_{B2,5}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	$d\alpha$ = .098° dN = .022 RPM
12) 2nd Midcourse Correction a) ΔV = 9 m/sec	P_T, T_T, $T_{B2,4,6,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses	Fire: 2,4,6,8	-	dV = .54 m/sec $d\alpha$ = .23° dN = .028 RPM
b) ΔV = 1 m/sec	P_T, T_T, $T_{B2,4,6,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses	Fire: 2,4,6,8	-	dV = .092 m/sec % Error = .92% $d\alpha$ = .026° dN = .0031 RPM
13) Reorient for Cruise a) $\Delta\alpha = 27.5°$	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dV = .031 m/sec $d\alpha$ = 1.25° dN = .30 RPM
b) Spin Control $\Delta N = \pm 0 - .3$ RPM	P_T, T_T, $T_{B1,2,3,4}$ N, $\vec{\alpha}$	ΔN, T_{on} if Necessary	Fire: 1 and 4 or 2 and 3	-	dN = .019 RPM
c) $\Delta\alpha = 10°$	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	$d\alpha$ = .45° dN = .11 RPM
d) $\Delta\alpha$ Course Trim $\Delta\alpha = 2.5°$	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8 Close Latching Valves After Firing	Since the next maneuver is E-10 days off the latching valves will be closed after this maneuver.	$d\alpha$ = .11° dN = .027 RPM
14) Orient for 3rd M/C a) $\Delta\alpha = 18°$	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$, Latching Valve Status	$\Delta\alpha$, EPW, Timing, No. of Pulses	Open Both Latching Valves Fire: 3 and 8	-	$d\alpha$ = .82° dN = .027 RPM
b) $\Delta\alpha$ Course Trim $\Delta\alpha = 2°$	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	$d\alpha$ = .098° dN = .022 RPM
15) 3rd Midcourse Correction a) ΔV = 1 m/sec	P_T, T_T, $T_{B2,4,6,8}$ N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 2,4,6,8	-	dV = .060 m/sec $d\alpha$ = .028 dN = .0034 RPM
b) ΔV = 1 m/sec	P_T, T_T, $T_{B2,4,6,8}$ N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 2,4,6,8	-	dV = .0598 m/sec $d\alpha$ = .028 % Error = 3% dN = .0034 RPM

15) 3rd Midcourse Correction a) ΔV = 1 m/sec	P_T, T_T, $T_{B2,4,6,8}$ N, $\vec{\alpha}$, $\vec{x}$	ΔV, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 2,4,6,8	-	dV = .060 m/sec dα = .028 dN = .0034 RPM
b) ΔV = 1 m/sec	P_T, T_T, $T_{B2,4,6,8}$ N, $\vec{\alpha}$, $\vec{x}$	$\vec{\Delta V}$, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 2,4,6,8	-	dV = .0598 m/sec dα = .028 % Error = 3% dN = .0034 RPM
16) Attitude Control Maintenance a) $\Delta\alpha$ = 18°	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dα = .82° dN = .2 RPM
b) $\Delta\alpha$ Course Trim $\Delta\alpha$ = 2°	P_T, T_T, $T_{B3m\,7}$, N $\vec{\alpha}$	$\Delta\alpha$, Trim, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dα = .098° dN = .022 RPM
17) Orient Spin Axis Parallel to Ecliptic a) $\Delta\alpha$ = 27.5°	P_T, T_T, $T_{B4,7}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 4 and 7	This maneuver will be done in four $\Delta\alpha$ steps.	dV = .015 m/sec dα = 1.25° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{B2,3,5,8}$ N	Δ N, Ton if Necessary	Fire: 2 and 5 or 3 and 8		dN = .019 RPM
Repeat two times for Events 17c), 17d), 17e), and 17f).					
g) $\Delta\alpha$ = 5°	P_T, T_T, $T_{B4,7}$, N $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 4 and 7	-	dα .23° dN = .055 RPM
h) $\Delta\alpha$ Course Trim $\Delta\alpha$ = 2.5°	P_T, T_T, $T_{B4,7}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$ Trim, EPS, Timing, No. of Pulses	Fire: 4 and 7	-	dα = .11° dN = .027 RPM
18) Retarget Maxi Probe a) Δ = 27.5°	P_T, T_T, $T_{B1,6}$, N $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	dα = 1.25° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{B5,6,7,8}$ N, $\vec{\alpha}$	Δ N, Ton if Necessary	Fire: 6 and 7 or 5 and 8	-	dN = .019 RPM
c) $\Delta\alpha$, = 10°	P_T, T_T, $T_{B1,6}$, N $\vec{\alpha}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	dα = .45° dN = .109 RPM
d) $\Delta\alpha$ Course Trim $\Delta\alpha$ = 2.5°	P_T, T_T, $T_{B1,6}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 1 and 6	-	dα = .11° dN = .027 RPM
19) Separate Maxi Probe	-	-	-	-	-
20) Attitude Control Maintenance a) $\Delta\alpha$ = 9°	P_T, T_T, $T_{B2,5}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα = .41° dN = .10 RPM
b) $\Delta\alpha$ = 1° Trim Maneuver	P_T, T_T, $T_{B2,5}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα = .10° dN = .0109 RPM
21) Retarget Mini-Probe a) ΔV = 4 m/sec	P_T, T_T, $T_{B1,3,5,7}$ N $\vec{\alpha}$, $\vec{x}$	$\vec{\Delta V}$, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 1,3,5,7	-	dV = .24 m/sec dα = .059° dN = .0074 RPM
b) ΔV = 1 M/S	P_T, T_T, $T_{B1,3,5,7}$ N, $\vec{\alpha}$, $\vec{x}$	$\vec{\Delta V}$, EPW, Timing, No. of Pulses taking into Account C.G. Shift	Fire: 1,3,5,7	-	dV = .074 m/sec % Error = 1.48% dα = .0148° dN = .00184 RPM

FOLDOUT FRAME 2

TABLE 5.4-II (CONTINUED)
FLIGHT OPERATIONS CHART - PROBE MISSION
SYSTEM: I (P-5/FS-2)

EVENTS	DATA REQUIRED FROM VEHICLE	GROUND COMPUTATION	DATA (COMMAND) TRANSMITTED TO VEHICLE	ASSUMPTIONS	ERRORS
22) Attitude Control Maintenance a) $\Delta\alpha = 18°$	P_T, T_T, T- 2,5 N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPS, Timing, No. of Pulses	Fire: 2 and 5	-	dV = .017 m/sec dα = .82° dN = .196 RPM
b) $\Delta\alpha = 2°$ Trim Control	P_T, T_T, $T_{B2,5}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$ Trim, EPW, Timing, No. of Pulses	Fire: 2 and 5	-	dα = .11° dN = .022 RPM
23) Spin Up Mini Probes a) ΔN = 65 RPM	P_T, T_T, $T_{B2,3}$, N, $\vec{\alpha}$, $\vec{x}$	ΔN, T_{on}	Fire: 2 and 3	-	dV = .0023 m/sec dα = .085° dN = 4.2 RPM
b) Spin Up Trim ΔN = 4 - 12 RPM	P_T, T_T, $T_{B2,3}$, N, $\vec{\alpha}$, $\vec{x}$	ΔN, T_{on}	Fire: 2 and 3	-	dα = .085° dN = .77 RPM
c) Spin Up Trim ΔN = ±0 - .77 RPM	P_T, T_T, $T_{B5,6,7,8}$ N, $\vec{\alpha}$, $\vec{x}$	ΔN, Ton	Fire: 6 and 7 or 5 and 8	-	dN = .049 RPM
24) Separate Mini Probes	-	-	-	-	-
25) Attitude Control Maintenance a) $\Delta\alpha = 3.9°$	P_T, T_T, $T_{B3,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	This maneuver of $\Delta\alpha = 20°$ total has to be performed in $\Delta\alpha = 3.9°$ increments because of the high spin perturbation which occurs at N = 85 RPM.	dV = .014 m/sec dα = .18° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{B5,6,7,8}$, N	ΔN, T_{on} if Necessary	Fire: 5 and 8 or 6 and 7	-	dV= .0044 m/sec dα= .031° dN= .019 RPM
Repeat three more times for Events 25 c), d), e), f), g), h), i), j).					
k) $\Delta\alpha$ Trim $\Delta\alpha = .5°$	P_T, T_T, $T_{B3,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses taking	Fire: 3 and 8	-	dα= .051°
26) Retarget S/C Bus	P_T, T_T, $T_{B2,4,6,8}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking	Fire: 2,4,6,8	-	dV = .9 m/sec

h), i), j).					
k) $\Delta\alpha$ Trim $\Delta\alpha = .5°$	P_T, T_T, $T_{B3,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses taking	Fire: 3 and 8	-	$d\alpha$ = .051°
26) Retarget S/C Bus a) ΔV = 15 m/sec	P_T, T_T, $T_{B2,4,6,8}$ N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking into Account C. G. Shift Due to Maneuvers	Fire: 2,4,6,8	-	dV = .9 m/sec $d\alpha$ = .058° dN = .0102 RPM
b) ΔV = 3 M/Sec	P_T, T_T, $T_{B2,4,6,8}$N, $\vec{\alpha}$, $\vec{x}$	$\Delta\vec{V}$, EPW, Timing, No. of Pulses taking into Account C. G. Shift Due to Previous Maneuvers	Fire: 2,4,6.8	-	dV = .23 m/sec % Error = 1.3%
27) Correct Sun Angle Drift a) $\Delta\alpha$ = 3.9°	P_T, T_T, $T_{B3,8}$, N, $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dV = .014 m/sec $d\alpha$ = .18° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{B5,6,7,8}$ N	ΔN, T_{on} if Necessary	Fire: 5 and 8 or 6 and 7	-	dV = .0044 m/sec $d\alpha$ = .031° dN = .019 RPM
Repeat three more time for Events 27c), d), e), f), g), h), i), j)					
k) $\Delta\alpha$ = .5°	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	$d\alpha$ = .051°
28) Orient Spin to Velocity Vector a) $\Delta\alpha$ = 3.9°	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing, No. of Pulses	Fire: 3 and 8	-	dV = .014 m/sec $d\alpha$ = .18° dN = .30 RPM
b) Spin Control ΔN = ±0 - .3 RPM	P_T, T_T, $T_{B5,6,7,8}$ N	ΔN, T_{on} if Necessary	Fire: 5 and 8 or 6 and 7	-	$d\alpha$ = .031°
Repeat for Events 28c), d), e), f)					
g) $\Delta\alpha$ Trim $\Delta\alpha$ = .3°	P_T, T_T, $T_{B3,8}$, N $\vec{\alpha}$, $\vec{x}$	$\Delta\alpha$, EPW, Timing No. of Pulses	Fire: 3 and 8	-	$d\alpha$ = .051°

FOLDOUT FRAME 2

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5.5 Components

After determining which propulsion subsystems were to be evaluated, a survey was performed of operational hydrazine propulsion subsystems and those presently being developed, to identify those components which have been qualified, and can be considered for the Planetary Explorer applications. Data was gathered on components which would be considered as candidates for the propulsion subsystem concepts and this data was evaluated. In some cases components were considered which have flight history on applications other than hydrazine propulsion systems. The basic criteria used for the component evaluation were the flight and qualification histories of each component, along with judgment from previous component evaluation studies. Components recommended for the Planetary Explorer applications are listed in Table 5.5.0-I and the technical evaluations are summarized in Tables 5.5.0-II through 5.5.0-VII. Each component was rated from 0 to 3 points in the categories shown. A weighting factor was then applied to each rating to reflect the relative importance of each category. Component evaluation and trade-off information generated during the study and from previous component selection studies was referred to and modified, as necessary, to reflect Planetary Explorer requirements. The following is a definition of the rating scale and weighting factors used.

<u>Rating Scale</u>

3 - Acceptable: Actual demonstrated - no risk
2 - Acceptable: Normal risk
1 - Marginal: High Risk
0 - Unacceptable

<u>Weighting Factors</u>

Factor		Definition
10	-	System will not function if criteria are not satisfied
8	-	System will function but specification is in jeopardy
4	-	System functions or specifications not in jeopardy but component characteristic may enhance capability of system

5.5.1 PROPELLANT TANK:

The propellant tank recommended at this time for the Planetary Explorer Orbiter and Probe spacecraft is the tank manufactured by Fansteel Metallurgical Corp. which is used on the IDCSP/A and NATO/SAT satellites. This tank, Fansteel part number 4425034, is designed and ported for a spin stabilized spacecraft application. The tank is designed as a bladderless integral pressurization expulsion system when either in the ground or flight acceleration fields. Location of

5.5.1 (continued)

the propellant outlet ports 90 degrees to each other provides for propellant draining during either ground test or when the satellite is spinning in space. This porting configuration also provides for flushing the tanks and lines after fabrication by allowing a continuous flow of fluid into the pressurant port and out the propellant port. The Fansteel tank is of the proper volumetric size for use in either Planetary Explorer application by utilization of a sufficient number of tanks, and provides a pressurant "blow down" ratio compatible with the IDCSP/A engine which has been considered in this study. Figure 5.5.1-1 shows system tankage weight as a function of propellant weight in the system for both the existing Fansteel design and a theoretically optimum tank design.

The basis for sizing an optimum tank for comparison to the existing design are the following:

- Tank Material: 6AL-4V-Ti
- Tank Outer Diameter: 9.86"
- Tank Wall Thickness: .012 - .015

  (.012 min wall based on F.S. = 2 for stability of tank at 1 atmosphere).

The figure indicates a slight advantage in weight for the optimum tank design; however, this savings probably does not justify the costs associated with developing and requalifying a new design.

Another tank which meets the volumetric requirements but does not have the flexibility of the Fansteel tank is one manufactured by Pressure Systems Incorporated for the Intelsat III satellite. This tank is designed for a bladderless integral pressurization system for a spin stabilized application only where propellant cannot be completely expelled under earth gravity unless the vehicle is turned on its side. Also, the Pressure Systems Incorporated (PSI) tank is trunnion mounted, whereas the Fansteel tank is flange mounted. Flange mounting is more suitable for the Planetary Explorer application and provides for a much lighter weight tank mounting structure.

The Fansteel and PSI tanks appear to be the only two flight qualified tanks available which meet the Planetary Explorer volume and size requirements. Of the two tanks, the Fansteel tank is the most suitable from a system integration and mounting standpoint. Table 5.5.1-I summarizes the basic characteristics of the two tanks.

5.5.2 FILTER:

The propellant filter recommended at this time is the filter manufactured by Vacco Industries which is used on the Intelsat IV satellite. The Vacco filter contains a multi-segmented element consisting of a stack of etched filter discs which are chemically milled from the basic stock. This eliminates the possibility of burrs being generated, and permits cleaning and inspection of each individual disc prior to assembly of the element, minimizing self-generated contamination. Another filter which has been incorporated in some spacecraft applications is a wire cloth element manufactured by Wintec Corporation. Filters of this type are difficult to manufacture clean and tend to become contaminant generators since there are a multitude of traps where contamination can be retained during manufacture of the wire cloth and after flushing the assembled filter. A summary of data for the two filters is shown in Table 5.5.2-I. The Vacco filter is judged superior in being able to be manufactured clean, retain contaminant to a specific absolute value, and not be a contaminant generator.

5.5.3 FILL AND DRAIN VALVE:

The fill and drain valve recommended for the propellant and pressurant fill and drain functions is the valve manufactured by Vacco Industries. This valve is a manually operated shut-off valve. The sealing of this valve is effected by seating a tungsten carbide ball into a seat formed in the valve body. The ball is held captive in the poppet which is activated open and closed by rotation of a retainer nut. The poppet is sealed when the valve is open during fill and drain operations by an "O" ring forming the poppet/body seal. Seating and sealing of the poppet into the seat is achieved by torqueing the retainer nut to a prescribed torque value. Redundant sealing is achieved by capping the valve port, and by the poppet/body seal. The Pyronetics fill and drain valve is similar to the Vacco in that primary sealing is achieved by metal-to-metal contact, but the Pyronetics valve utilizes a cylindrical sharp edge poppet seating into a tapered seat. A valve of this type is sensitive to overtorqueing with subsequent leakage problems. The Pyronetics valve has flown on spacecraft and is qualified for space applications, but is considered to have more potential problems than the Vacco fill and drain valve.

Other fill and drain valves considered were those manufactured by Futurecraft and Snap-Tite. Table 5.5.3-I is a listing of the characteristics for the fill and drain valves considered.

5.5.4 ORDNANCE VALVE:

The ordnance valve recommended is the valve manufactured by Pyronetics Incorporated (Part Number 1365), which is functionally identical to valves supplied for the Gemini, Minuteman III and Intelsat III programs. The valve is internally the same as Part Number 1078 used on the Gemini spacecraft, the difference being the use of the Apollo Standard Initiator for actuation. The same valve, but

5.5.4 (continued)

designated Part Number 1259, has flown on the Intelsat III satellite, with the difference being the squib and mounting provisions. This unit provides hermetic isolation of the propellant from the downstream components until actuated open by firing the squib. The flow path is opened when pressure generated from firing the squib drives a ram which shears a solid hermetic closure on the valve inlet port. The squib considered is an "off-the-shelf" dual bridge wire cartridge designed for deep space applications. Conax Corporation and Futurecraft also supply ordnance valves which provide a hermetic closure at the valve inlet port which is sheared off by a ram actuated by the squib gases. These companies have supplied units for many military missile and space applications. Pyrotechnics Incorporated has supplied units for hydrazine propulsion subsystems which are now operational and are readily available and qualified for the Planetary Explorer application. Table 5.5.4-I summarizes the characteristics of the ordnance valves considered.

5.5.5 LATCHING SOLENOID VALVE

A review of latching solenoid valves that are manufactured for space applications shows that only two designs have been qualified for hydrazine applications. Table 5.5.5-I summarizes the characteristics of these and other designs. The Carleton Controls design, Part Number 2217002, is qualified for the Intelsat IV program and represents the only qualified, lightweight valve for hydrazine use. Although it has these favorable characteristics, the valve is very complex in construction and operation. The design utilizes a stainless steel poppet head with a teflon seating surface, and metal bellows are used to internally seal the valve from fluid flow, and also effectively pressure balance the valve poppet against both inlet and outlet differentials. The valve has sliding fits, many moving parts (including the bellows, poppet, belleville spring, and plunger), and dynamic seals are used as backups to the bellows. Valve latch holding forces for both open and closed positions are generated by a belleville spring and the latching force is overcome by dual electromagnets which move the armature in opposite directions when energized. Unlike the other designs, the Carleton valve does not use the force field of a permanent magnet to hold the valve in the open position, therefore eliminating the weight associated with the permanent magnet. In addition to the use of the permanent magnets for latching, the Parker and the Consolidated Controls valves operate by actuating a sliding fit plunger which moves the poppet to the open position. To close these valves, current of reversed polarity is supplied to a solenoid coil, thus producing a magnetic field which cancels a portion of the latching force field and a spring device returns the poppet to the closed position.

The Carleton valve appears to be the best choice of what is available based primarily on the qualified/lightweight characteristics of the valve. However, due to the inherent complexity of the sliding fits, bellows and other moving parts, there may be an advantage in considering the use of a torque motor operated type latching valve. It has been Hamilton Standard's experience to receive proposals (for various

5.5.5 (continued)

programs) from manufacturers of torque motor valves which utilize the single flapper concept with a permanent magnet to maintain the valve in the open position. This approach may prove to have advantages over the solenoid type due to simplicity and minimization of moving parts. However, the advantages still have to be compared to the cost of development and qualification since this status does not exist for a torque motor design.

5.5.6 PROPELLANT CONTROL VALVE:

The propellant control valve recommended for the Planetary Explorer application is the valve manufactured by Hydraulic Research and Manufacturing Corporation (Part Number 48000680) used on the IDCSP/A and NATOSAT satellite Rocket Engine Assemblies. The valve is a normally closed torque motor operated dual seat valve. The design incorporates two metal-to-metal flat-lapped poppets and seats in series to provide valve sealing redundancy. This valve is considered for Planetary Explorer for the following reasons:

- The engine recommended by Hamilton Standard for the Planetary Explorer applications was qualified with this valve for the IDCSP/A and NATOSAT programs.

- The redundant seat configuration provides a higher degree of confidence in mission success than a single seat valve.

A summary of all engine valves considered, including torque motor and solenoid types, is shown in Table 5.5.6-I. All units, with the exception of the Moog valve, have qualification status and space program history. However, only three designs (Hydraulic Research and Manufacturing, Stratos and Kidde), have actual flight history, and of these three, only the Stratos (a solenoid operated design with sliding parts) and the HR&M valves utilize the dual series seat configuration. Final consideration resulted between these two valves, and since the HR&M valve is qualified for use with the engine selected for the Planetary Explorer application, it was chosen and is thus recommended.

5.5.7 PRESSURE TRANSDUCER:

Table 5.5.7-I lists the possible candidates for the selection of a pressure transducer. Each represents a design utilized on various space applications requiring various degrees of stability and accuracy, low weight and volume, and low power consumption. The Fairchild Controls and Dynasciences Designs are similar, in that they utilize a pressure sensitive diaphragm to which is bonded silicon semiconductor strain gages. The Statham transudcer is a vacuum deposited thin film strain-gage type which eliminates the use of bonding agents for attachment of the strain gage

5.5.7 (continued)

to the sensing element, and the Bourns units represents still a different type of transducer, in that it is a variable reluctance transducer, utilizing a twisted Bourdon tube as a strain sensing element.

Since all of the candidate designs presented have comparable qualification background and experience, and all appear to be satisfactory for the Planetary Explorer applications, no particular unit is recommended at this time.

5.5.8 GIMBAL ACTUATOR:

Based on limited data available on small, lightweight actuators used for space applications, the only actuator satisfying the above requirements and suited for the Planetary Explorer applications is manufactured by Nash Controls, Inc. and is Part Number DL2323M1. The design is a linear actuator -- used on the Lunar Excursion Module to position the landing radar antenna from the stowed to operating position. The unit is compact (approximately 2.25 inches by 2.25 inches by 4.00 inches), lightweight (1.05 lbs) and is compatible with a hard vacuum environment. The unit is powered by a permanent magnet direct current motor which drives the extending ram to the desired position which is monitored by an indicator switch which signals the ram position to external circuits. Designed to operate normally with a 28 volts dc supply, the actuator has a normal stroke range of .1 inch to 1.25 inches with a high positioning accuracy, and with the capability of intermittent or continuous operation.

TABLE 5.5.0.-I. DEVELOPMENT STATUS AND

WEIGHT SUMMARY-SELECTED COMPONENTS

Component	Manufacturer & Part Number	Weight lbs	Development Status	Program History
Propellant Tank	• Fansteel PN 4425034	1.60	Flight	IDSCP/A
Propellant Valve	• Hydraulic Research and Manufacturing PN 8000680	.47	Flight	IDSCP/A
Latching Solenoid Valve	• Carlton Controls PN 2217001-2	.54	Qualified	Intelsat IV
Ordnance Valve	• Pyronetics PN 1365	.30	Similar to Qualified Design except for initiator (1)	Similar design used on Gemini, Intelsat III and Minuteman III
Fill and Drain Valve	• Vacco PN 3181407	.25	Qualified	Intelsat IV
Filter	• Vacco PN-F1D10064	.30	Qualified	Intelsat IV
Pressure Trans-(2) ducer	See Table 5.5.7-I			
Linear Actuator	• NASH Controls PN DL2323M1	1.05	Flight	LEM

(1) Design utilizes Apollo standard initiator (ATI)

(2) Final selection not made

TABLE 5.5.0-III. ENGINEERING EVALUATION OF CANDIDATE LATCHING SOLENOID VALVES

Criteria	Weighing Factor A	Carleton PN 2217001-2 Rating B	Carleton PN 2217001-2 Tot Score A x B	Parker PN 5640014 Rating B	Parker PN 5640014 Tot Score A x B	Parker PN 5680011 Rating B	Parker PN 5680011 Tot Score A x B	Consolidated Cont PN 3795000-3 Rating B	Consolidated Cont PN 3795000-3 Tot Score A x B
Design									
1. Weight	4	3	12	1	4	1	4	1	4
2. Simplicity	4	2	8	2	8	2	8	2	8
3. Envelope Adaptability	4	3	12	2	8	2	8	2	8
4. Operating Concepts	10	3	12	3	12	3	12	3	12
5. Number of potential problem areas	10	2	20	2	20	2	20	1	10
6. Sealing capability	10	3	30	2	20	2	20	1	10
Performance & Life									
1. Expected ability to meet acceptance test	8	3	24	3	24	2	16	1	8
2. Expected ability to meet qual. test	10	3	30	3	30	3	30	2	20
3. Expected ability to meet design life	10	3	30	3	30	3	30	3	30
Vendor Background									
1. Experience with similar units	4	3	12	3	12	3	12	3	12
2. Awareness of potential trouble areas	4	3	12	3	12	3	12	3	12
TOTALS			202		180		172		134

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TABLE 5.5.0-IV. ENGINEERING EVALUATION OF CANDIDATE ORDNANCE VALVES

Criteria	Weighing Factor A	Pyronetics PN 1365 Rating B	Pyronetics PN 1365 Tot Score A x B	Conax PN 1832-131 Rating B	Conax PN 1832-131 Tot Score A x B	Futurecraft PN 3467 Rating B	Futurecraft PN 3467 Tot Score A x B
Design							
1. Weight	4	3	12	3	12	3	12
2. Simplicity	4	3	12	3	12	3	12
3. Envelope Adaptability	4	3	12	3	12	3	12
4. Operating Concepts	10	3	30	2	20	2	20
5. Number of Potential Problem Areas	10	2	20	2	20	2	20
6. Sealing capability	10	2	20	2	20	2	20
Performance & Life							
1. Expected ability to meet acceptance test	8	3	24	2	16	1	8
2. Expected ability to meet qual test	10	3	30	1	10	1	10
3. Expected ability to meet design life	10	3	30	2	20	2	20
Vendor Background							
1. Experience with similar units	4	3	12	2	8	2	8
2. Awareness of potential trouble areas	4	2	8	2	8	2	8
TOTALS			210		148		140

TABLE 5.5.0-V. ENGINEERING EVALUATION OF CANDIDATE FILL & DRAIN VALVES

Criteria	Weighing Factor A	Pyronetics P/N 1805		Vacco P/N 3181407		Futurecraft P/N 50448		Snaptite P/N 4274	
		Rating B	Tot Score A x B	Rating B	Tot Score A x B	Rating B	Tot Score A x B	Rating B	Tot Score A x B
Design									
1. Weight	4	3	12	3	12	3	12	3	12
2. Simplicity	4	3	12	3	12	1	4	1	4
3. Envelope Adaptability	4	3	12	3	12	2	8	2	8
4. Operating Concepts	4	3	12	3	12	2	8	2	8
5. Number of Potential Problem Areas	8	2	16	2	16	1	8	1	8
6. Sealing Capability	10	2	20	3	30	1	10	1	10
Performance & Life									
1. Expected ability to meet acceptance test	8	2	16	3	24	2	16	2	16
2. Expected ability to meet qual. test	10	2	20	3	30	2	20	2	20
3. Expected ability to meet design life	10	2	20	3	30	2	20	2	20
Vendor Background									
1. Experience with similar units	4	2	8	2	8	3	12	2	8
2. Awareness of potential trouble areas	4	2	8	2	8	2	8	2	8
TOTALS			156		194		126		122

TABLE 5.5.0-VI. ENGINEERING EVALUATION OF CANDIDATE PROPELLANT TANKS

Criteria	Weighing Factor A	Psi PN 80076 Rating B	Psi PN 80076 Tot Score A x B	Fansteel PN 4425034 Rating B	Fansteel PN 4425034 Tot Score A x B
Design					
1. Weight	4	3	12	3	12
2. Simplicity	4	2	8	2	8
3. Envelope Adaptability	8	2	16	3	24
4. Operating Concepts*	8	2	16	3	24
5. Number of Potential Problem Areas	4	1	4	2	8
6. Sealing Capability	4	3	12	3	12
Performance & Life					
1. Expected ability to meet acceptance test	8	3	24	3	24
2. Expected ability to meet qual test	10	3	30	3	30
3. Expected ability to meet design life	10	3	30	3	30
Vendor Background					
1. Experience with similar units	4	2	8	3	12
2. Awareness of potential trouble areas	4	3	12	3	12
TOTALS			172		196

*Port arrangement for expulsion

TABLE 5.5.0-VII. ENGINEERING EVALUATION OF CANDIDATE FILTERS

Criteria	Weighing Factor A	Vacco PN F1D10064-01 Rating B	Vacco PN F1D10064-01 Tot Score A x B	Wintec PN 14251-569 Rating B	Wintec PN 14251-569 Tot Score A x B
Design					
1. Weight	4	2	12	3	12
2. Simplicity	4	2	8	3	12
3. Envelope Adaptability	4	3	12	3	12
4. Operating Concepts	8	3	24	2	16
5. Number of potential problem areas	8	3	24	2	16
6. Sealing capability	4	3	12	3	12
Performance & Life					
1, Expected ability to meet acceptance test	8	3	24	3	24
2. Expected ability to meet qual. test	10	3	30	3	30
3. Expected ability to meet design life	10	3	30	3	30
Vendor Background					
1. Experience with similar units	4	3	12	3	12
2. Awareness of potential trouble areas	4	3	12	3	12
TOTALS			200		178

TABLE 5.5.1-I. PROPELLANT TANK CHARACTERISTICS

Characteristics	Manufacturer	
	Fansteel	PSI
Manufacturer's Part Number	4425034	80076
Envelope (in.)	9.86 o.d.	9.56 o.d.
Volume (cu in.)	497	457
Porting	3 ports	2 ports
Expulsion Device	None	None
Weight (lb-max)	1.6	1.6
Operating Pressure (psia)	400	600
Burst Pressure (psid)	100	1200
Program History	IDCSP/A	Intelsat III

TABLE 5.5.2-I. FILTER CHARACTERISTICS

Characteristics	Manufacturer	
	Wintec	Vacco
Manufacturer's Part Number	15241-569	FID 10064-01
Envelope (in)	1.25 dia x 2.2 lg	1.0 dia x 2.0 lg
Capacity (mg of AC fine dust)	100	100
Pressure Drop @ Rated Flow (Δ psi @ lb/sec)	< 3.0 @ .015	< 5.0 @ .025
Type	Pleated wire mesh (reinforced)	Etched disc stack
Weight (lb)	0.33	0.30
Filtration Level (microns)	10 ABS	10 ABS
Burst Pressure (psid)	2400	1200
Program History	Intelsat III	Intelsat IV

TABLE 5.5.3-I. FILL AND DRAIN VALVE CHARACTERISTICS

Characteristics	Manufacturer			
	Snap-Tite	Pyronetics	Vacco	Futurecraft
Manufacturer's Part Number	4274	1805	3181407	50448
Pressure Drop @ Rated Flow (Δ psi @ lb/sec)	20 @ 0.1	(.022 in^2 area)	20 @ 0.5	20 @ 0.5
Physical Size (Envelope) - (in.)	1.0 dia x 1.47 lg (not incl fitt)	1.3 dia x 1.99 lg (incl cap)	1.0 x 1.5 x 3.06 lg	1.3 x 2.0 x 1.57 lg (not incl fitt)
Leakage, Uncoupled (scc N_2/hr)	0.5 350 psia)	1×10^{-5} (scc He/ sec, 350 psia)	1×10^{-6} (sec He/ sec, 200 psia)	0.5 (300 psig)
Seat Arrangement	poppet/ o-ring	poppet/ hard seat	ball/ hard seat	poppet/ soft seat
Weight	0.18	0.25	0.15 Ti housing	0.2
Coupling Arrangement	quick disconnect ball-lock	MS flare-tube fitting	MS flare-tube fitting	threaded coupling
Operating Pressure (psid)	0 - 350	600	0 - 200	0 - 300
Burst Pressure (psid)	700	2400	600	1178
Program History	P-95	Intelsat III	Intelsat IV	Titan Sandia

TABLE 5.5.4-I. ORDNANCE VALVE CHARACTERISTICS

Characteristics	Manufacturer		
	Conax	Futurecraft	Pyronetics
Manufacturer's Part Number	1832-131	31467	1365 (1)
Operating Pressure (psig)	1000	3000	5000
Physical Size (Envelope) - (in.)	.88 x 1.72 x 3.47 lg	.75 x 1.25 x 3.45 lg	.87 x 1.25 x 2.74 lg
Leakage (scc/sec of He)	$.5 \times 10^{-6}$	1×10^{-6} @ 6000 psi	1×10^{-6} @ 5000 psi
Opening Response (ms)	5 @ 5.0 a nom	5 @ 2.0 a	6 @ 4.5 a max
Weight (lb)	.45	-	.30
"Seat" Arrangement	Shear plug	Shear plug	Shear plug
Min Recommended Firing Current (amps)	5.0/circ.	1.0	4.5
Bridge Wire Resistance (ohms)	$1.0 \begin{smallmatrix}+.2\\-.1\end{smallmatrix}$	0.9 ±0.3 (@ .01 a max)	1.0 ± 0.1
Flow Passage (Min) After Actuation (in.)	.168 dia	.188 dia	.170 dia
No Fire Current (Max) (amps)	1.0/5 min (1 watt)	0.2/5 min	1.0/5 min (1 watt)
All Fire Current (amps)	-	2.0	2.77
Program History	-	-	Gemini Intelsat III Minuteman IV
Burst Pressure (psid)	14,520	12,000	10,000

TABLE 5.5.5-I. LATCHING VALVE CHARACTERISTICS

CHARACTERISTICS	MANUFACTURER			
	Consolidated Controls	Parker	Parker	Carleton Controls
Mfr. Part Number	3795000-3	5640014	5680011	2217001-2
Flow Rate lb/sec	.166	.166	.166	.022
Operating Pressure psia	0 - 400	0 - 250	0 - 400	0 - 300
Pressure Drop @ Rated Flow psi @ lb/sec	18 @ .166	~	18 @ .166	1. @ .022
Physical Size (Envelope) in.	1.48 dia. x 6.0 LG + Bosses	2.50 dia. x 3. LG + Bosses	2.46 dia. x 5.8 LG + Bosses	1.4 dia. x 4.6 LG + Tubes
Voltage Range vdc	20 - 33	18 - 32	20 - 33	18 - 50
Power (Max) watts	100 (33 vdc)	70 (28 vdc)	99 (33 vdc)	67.5 (27 vdc)
Leakage scc N_2/Hr	50 (425 psig)	10 (He)	10	1.4
Opening Response ms	50	~	50	30
Closing Response ms	50	~	50	30
Seat Arrangement	Poppet	Poppet	Poppet	Poppet
Weight	1.98	1.45	2.6	0.54 (No cable)
Program History	Apollo	Apollo (LEM)	P-95	Intelsat IV
Position Indicator	Yes	Yes	Yes	Yes
Life cycles	5000	~	5000	1000
Latching Mechanism (Open Position)	Permanent Magnet	Permanent Magnet	Permanent Magnet	Belleville Spring
Burst Pressure (psid)	800	1050	800	1250

FOLDOUT FRAME

TABLE 5.5.6-I. THRUST CHAMBER VALVE CHARACTERISTICS

CHARACTERISTICS		MANUFACTURER							
		HR&M	HR&M	Stratos	Stratos	Parker	Parker	Moog	Kidde
Manufacturer Part Number		48000680	48001000	449000	403000	5680036	5690023	Develop Unit	141806
Type		Torque Motor	Torque Motor	Single Solenoid	Dual Coil Sol.	Solenoid	Solenoid	Solenoid	Solenoid
Seat Configuration		Dual Series	Dual Series	Dual Series	Dual Series	Hard	Soft	Soft	Single Soft
Sealing Material		Tung. Carb.	Tung. Carb.	Ethylene Propylene	Ethylene Propylene	304L Seat 17-4 Poppet	Teflon Seat 17-4 Poppet	Teflon Poppet Hard Seat	Saphire - Teflon
Number of Coils		2	2	1	2	1	1	1	1
Coil Resistance	ohms	73-78	43-45	73-78	-	42 at 70°F	42 ±1 at 70°F	-	58 ±2
Size		1.33x2.17x4.62	1.4x2.13x5.22	1x2x4.65	-	1.6x1.83x3.03	1.6x1.83x3.03	-	1.07 Diax2.75
Weight	lb	<.047	< 0.67	< 0.34	.38	0.4	0.4	0.4	0.3
Power Profile (See Curves)		See Note 1	See Note 1	-	-	See Note 1	-	-	-
Maximum Power	watts	14 (33r)	21 (30r)	14 (24-304)	10	35 at 38 vdc	35 at 38 vdc	29 at 33 vdc	-
Max. Opening Response (Over Design Voltage Range)	ms	15 (spec)	15 (spec)	15 (spec)	15	6.5 at 24 vdc 250°F	6.9 at 24 vdc 250°F	-	11 at 30 vdc (spec)
Max. Closing Response (Over Design Voltage Range)	ms	10 (spec)	10 (spec)	10 (spec)	5	6.2 at 33 vdc 35°F	6.2 at 33 vdc 35°F	-	5 at 30 vdc (spec)
Max. Opening Response (28 ±2% vdc)	ms	11.0	11.0	12.5	-	5.1	5.3 at 70°F	4.5 at 70°F	-
Max. Closing Response (28 ±2% vdc)	ms	5.0	4.5	3.8	-	5.0	5.0 at 70°F	4.0 at 70°F	-
Operating Voltage Range	volts	24-36	18.6-35	24-36	25-31	24-33	24-33	24-33	23-35
Internal Leakage (Over Pressure Range)	N_2 scc/hr	0.50	1.00	0.50	0.50 (at 600)	10	5 (Life to 10)	1.0	0.24
Operating Pressure	psia	50-300	75-300	50-275	600	50-300	50-300	0-300	65-235
Max. Internal Operating Temperature	°F	250	160	250	250	250	250		

[illegible]	ms	5.0	4.9	3.8	-	5.0	5.0 at 70°F	4.0 at 70°F	-
Operating Voltage Range	volts	24-36	18.6-35	24-36	25-31	24-33	24-33	24-33	23-35
Internal Leakage (Over Pressure Range)	N_2 scc/hr	0.50	1.00	0.50	0.50 (at 600)	10	5 (Life to 10)	1.0	0.24
Operating Pressure	psia	50-300	75-300	50-275	600	50-300	50-300	0-300	65-235
Max. Internal Operating Temperature	°F	250	160	250	250	250	250	-	-
Demonstrated Cycle Life		50,000	350,000	50,000	50,000	50,000	50,000	1×10^6	-
Design Life (Space)	yrs	5	7	5	5		-	-	-
Repeatability, Opening	± ms	1	1	1	-	0.7	0.7	-	-
Repeatability Closing	± ms	1	1	1	-	0.7	0.7	-	-
Pressure Drop at Rated Flow	Δ psi at lb/sec	20 at .0224	28 at .0250	15 at .0224	70 at .0150	4.6 at .0117	14.6 at .0117	25 at .022	10 at .0220
Program Application History		IDCSP/A	I-IV/CP-3	IDCSP/A	I-III	P-95	P-95	Develop/P-95	ATS-III
Sinusoidal Vibration Qual Level		-	20 @ 80-120 cps	-	-	4g's (50-2000)	4g's (50-2000)	-	40(55-65 cps)
Random Vibration Qual Level	RMS g's	19.5	23.6	19.5	-	53.6	53.6	40	-
Flight History		Yes	No	No	Yes	No	No	No	Yes
Min. Dropout Voltage		2.0	1.0	2.0 vdc	-	2.5 vdc	2.5 vdc	2.5 vdc	-
Max. Pull-in Voltage		8.8	-	-	-	20 vdc 200 psig	20 vdc 200 psig	10 vdc	-
Status		Qualified	Qualified	Qualified	Qualified	Qualified	Qualified	Adv. Develop.	Qualified
Built-in Filter		25 μ Abs. 15 μ Nom.	No -	Yes -	20 μ Abs. -	25 μ Abs. 10 μ Nom.	25 μ Abs. 10 μ Nom.	25 μ Abs. 10 μ Nom.	No
Filter Source		HR&M	-	-	-	Wintec	Wintec	-	-
Burst Pressure (PSID)		900	>600	700	-	1200	1200	>1200	

Note: 1. Power Profile per Table 5.5.6-II

FOLDOUT FRAME 2

TABLE 5.5.7-I. PRESSURE TRANSDUCER CHARACTERISTICS

CHARACTERISTICS	MANUFACTURER			
	FAIRCHILD	DYNASCIENCES	STATHAM	BOURNS
Manufacturer's Part Number	TF125-4-0075	1025	PA 493 - 500	2309
Voltage (vdc)	24-32	28-30	24-32	24-32
Operating Pressure	0-500	0-300	0-500	0-300
Envelope	1.18 x 2.01 x 2.7 LG (Less Fitt. &Conn.)	1.0 Dia x 2.5 LG (Less Fitt. &Conn.)	1.0 Dia. x 2.5 LG (Less Fitt. &Conn.)	1.0 Dia x 2.0 LG (Less Fitt. & Conn.)
Weight	.32	.32	.32	.44
Output Voltage (vdc)	0-5.0 ±0.5	0-5	0-5	-.5 to +7.5
Program History	Lunar Orbiter IDCSP/A Natosat	Apollo LEM	Minuteman Pioneer Nimbus	P-95 Saturn V
Sensor Type	Bonded Silicon Strain Gage	Bonded Silicon Strain Gage	Deposited Film Strain Gage	Twisted Bourdon
Burst Pressure (PSID	10,000	7500	> 1000	1000

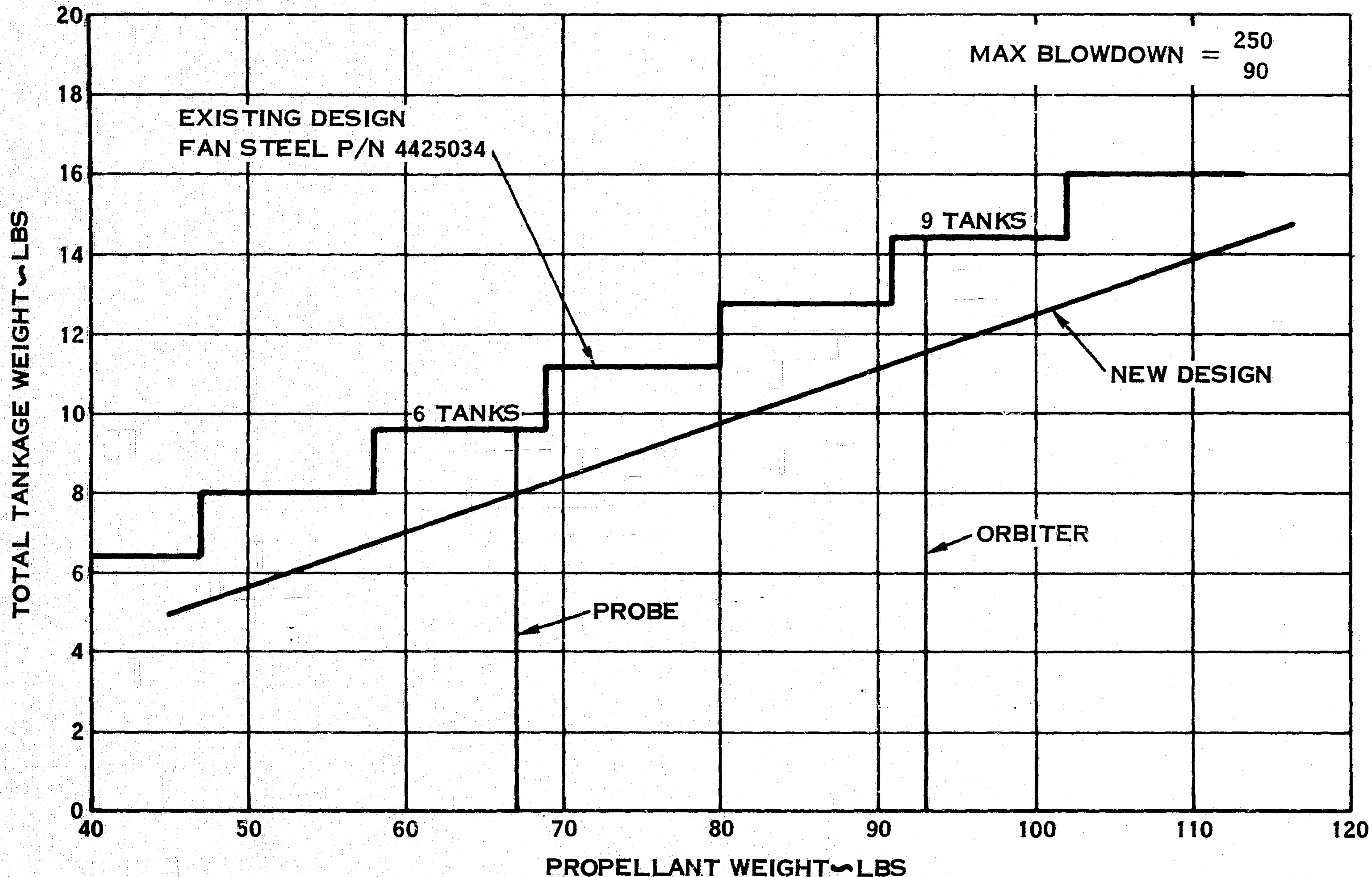

FIGURE 5.5.1–1. TOTAL TANKAGE WEIGHT VS PROPELLANT WEIGHT

5.6 Test Plans

In establishing an overall Test Plan for Planetary Explorer, as for any other program, the following factors constitute major influences:

- The degree of confidence desired by the customer before qualification testing of the system is initiated.

- The minimum requirements established by the customer for scope and depth of the system qualification test in terms of the number of test samples, and types of tests conducted.

- The extent to which selected components have previously demonstrated the capability of satisfying requirements that they have to meet for the Planetary Explorer program.

- The type of contract under which the program will be funded -- Fixed Fee or Cost type. It is natural to expect potential propulsion subsystem subcontractors to attempt to reduce program target costs by reducing the pre-qualification test effort where the increased risk in this approach has a limited liability to the subcontractor under the provisions of a cost type contract.

The program test plan presented herein is based on the following programmatic assumptions which were made on a best judgment basis to establish a frame of reference.

- The program will be funded under a Fixed Fee type of contract.

- Where equipment has been qualified at the component level or system level to environmental requirements which differ in some cases from those of Planetary Explorer, the subcontractor's engineering judgment, with supporting rationale, will suffice to establish the level to which additional component level testing is required. The judgment here will decide whether or not sufficient confidence can be established to enter system level qualification testing without additional component level testing, or if supplemental testing or complete requalification of the component is required.

- One flight configuration system will constitute the test sample for qualification of the design.

- The system design will be based on using individually mounted rocket engine assemblies (REA's), rather than the clustering of REA's in a module (reference Section 4.4.5 for rationale).

5.6 (continued)

- System level testing prior to qualification testing of the system will not be a program requirement.

- Spin balance testing will not be required for qualification or acceptance testing because balancing will be accomplished by GSFC at the spacecraft level (reference Section 5.12 for rationale).

- Reduced program costs will be a major program objective.

With the above information as a basis, the Test Plan recommended by Hamilton Standard is discussed below:

A test matrix for the overall test program is presented in Table 5.6.0-I. In summary, this matrix defines a test program wherein component level testing, other than acceptance testing, is limited to design verification tests for a few items, with qualification testing at the component level not required. This matrix also defines a program in which there is no system level testing prior to the system level qualification test. The rationale for the approach defined in the test matrix is discussed below:

A basic approach in reducing costs of any program such as Planetary Explorer is to reduce the technical risk associated with meeting program requirements, and the utilization of equipment with previously established capability so as to minimize the cost of proving that a capability does, in fact, exist. One of the major considerations in the selection of components for the candidate systems was the extent to which each component had previously demonstrated a capability to meet Planetary Explorer mission requirements. In almost every case, the selected components have been qualified at the component level, and at the system level, in monopropellant hydrazine propulsion systems for flight programs. Major items of equipment such as the rocket engine assemblies and the propellant tanks are presently flight operational in the configuration for which they have been selected for the candidate systems described herein. The various phases of the recommended test program are discussed below.

5.6.1 COMPONENT VERIFICATION TESTS:

The primary purpose of the verification testing is to confirm an engineering judgment that the components which have been qualified to requirements which differ somewhat from those of Planetary Explorer will be able to perform as required when subjected to qualification testing on the propulsion system level. Figure 5.6.1-1 (Verification Test Sequences) illustrates the test sequences proposed for the propellant control valve and the propellant tanks. These tests are to assure that the difference between the vibration environment which these components were subject to, and those of the

TABLE 5.6.0-I. TEST MATRIX

Equipment Level	Acceptance Test	Verification Test	Performance Mapping	Qualification Test
Component				
• Latching Solenoid Valve	X	X		
• Fill & Drain Valve	X			
• Filter	X			
• Pressure Transducer	X			
• Propellant Tank	X			
• Engine Propellant Valves	X	X		
• Rocket Engine Assembly	X	X	X	
System	X			X

NOTE: Performance mapping of the rocket engine assembly is part of the system qualification test which is run at the component level in order to obtain propellant flow rate data during simulated Planetary Explorer duty cycle pulsing operation.

5.6.1 (continued)

Planetary Explorer mission, will not affect the component qualification status. A comparison of the vibration requirements is presented in Figure 5.6.1-2. Although the Planetary Explorer requirement is lower in magnitude than the IDCSP/A level, the duration of 4 minutes per axis instead of 1 1/2 minutes per axis requires verification.

5.6.2 SYSTEM QUALIFICATION:

The planned qualification test sequence illustrated in Figure 5.6.2-1 provides assurance that the Planetary Explorer propulsion system meets all technical requirements. A description of each phase of the qualification test is presented in Table 5.6.2-I.

5.6.3 ACCEPTANCE TESTING:

The objectives of acceptance testing are to assure that the materials, workmanship and performance of assemblies to be subjected to qualification tests, or programmed for delivery, perform as required, and that these assemblies have been manufactured to approved drawings and specifications.

Since acceptance testing is a recurring program task, particular attention is directed toward minimizing the individual test cost involved while maintaining test objectives.

The propellant control valve, propellant tank, latching solenoid valve, fill and drain valve, pressure transducer, and filter are all acceptance tested at the component supplier's facilities. This testing is witnessed by Hamilton Standard source inspection. In addition, the reaction engine assemblies are tested at Hamilton Standard's facilities. Component acceptance tests are described in test flow chart form in Figure 5.6.3-1.

The propulsion system is assembled using components that have successfully completed component level acceptance tests and is then subjected to the system level acceptance tests defined in Figure 5.6.3-2.

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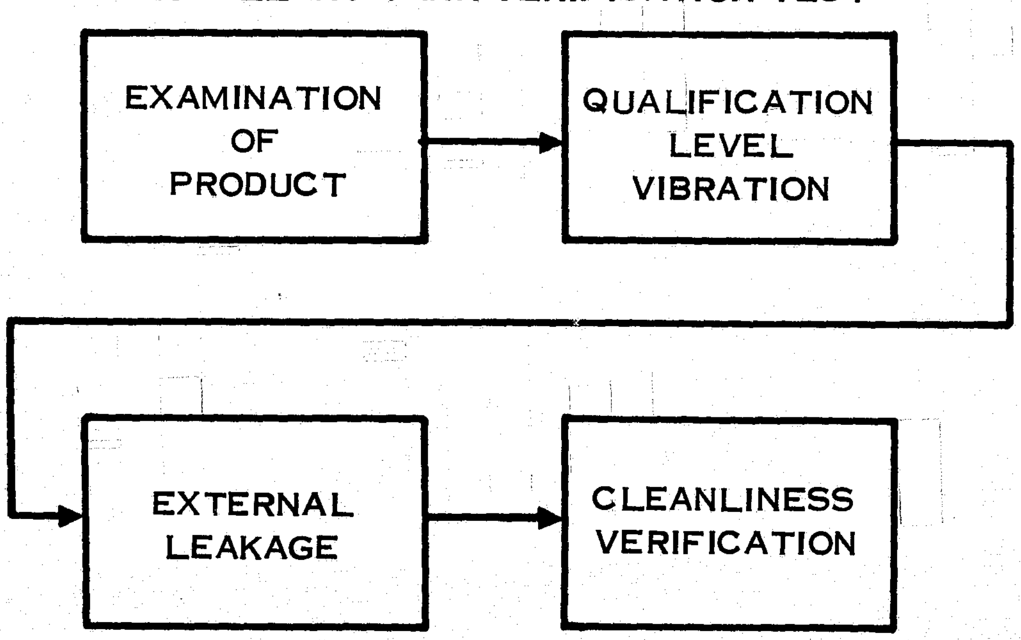
FOLDOUT FRAME

1

PROPELLANT TANK VERIFICATION TEST



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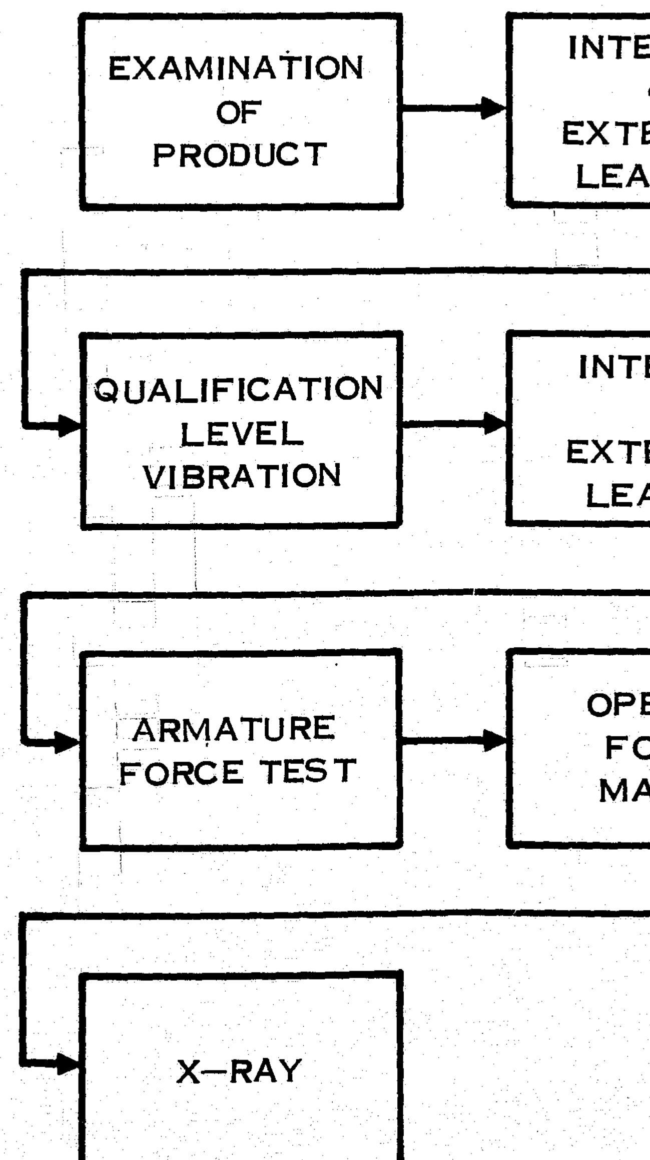


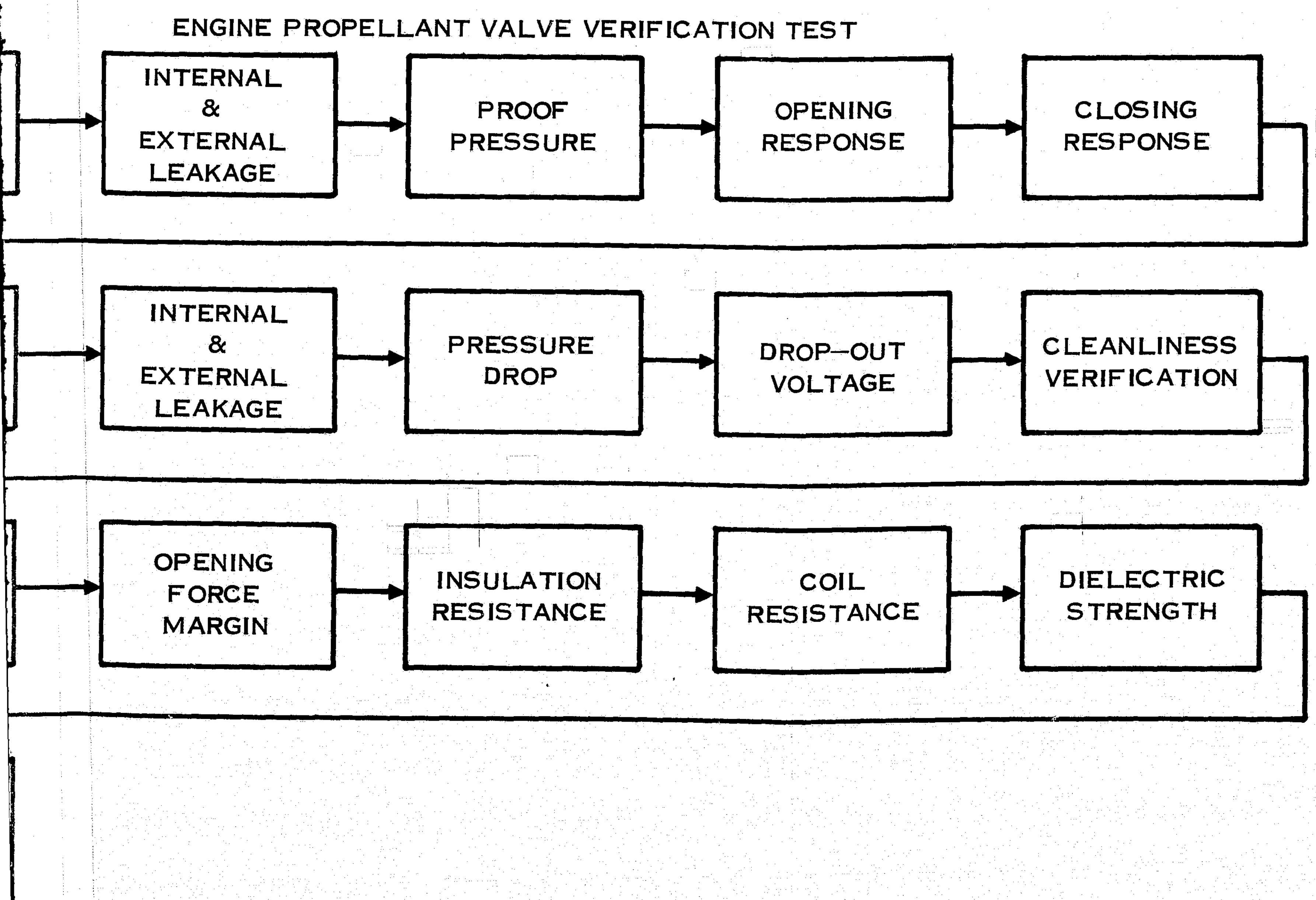


FIGURE 5.6.1—1. VERIFICATION TEST SEQUENCES

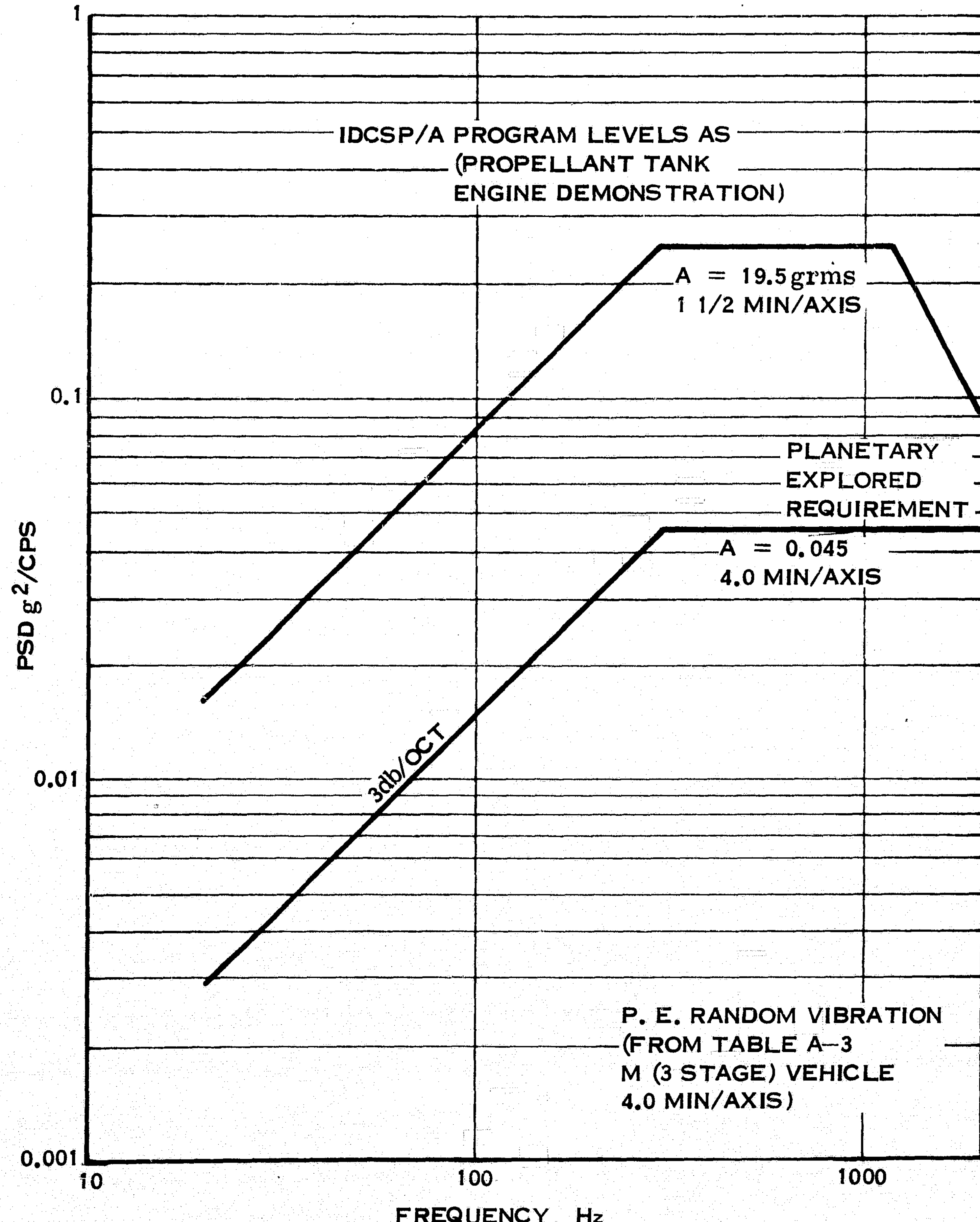
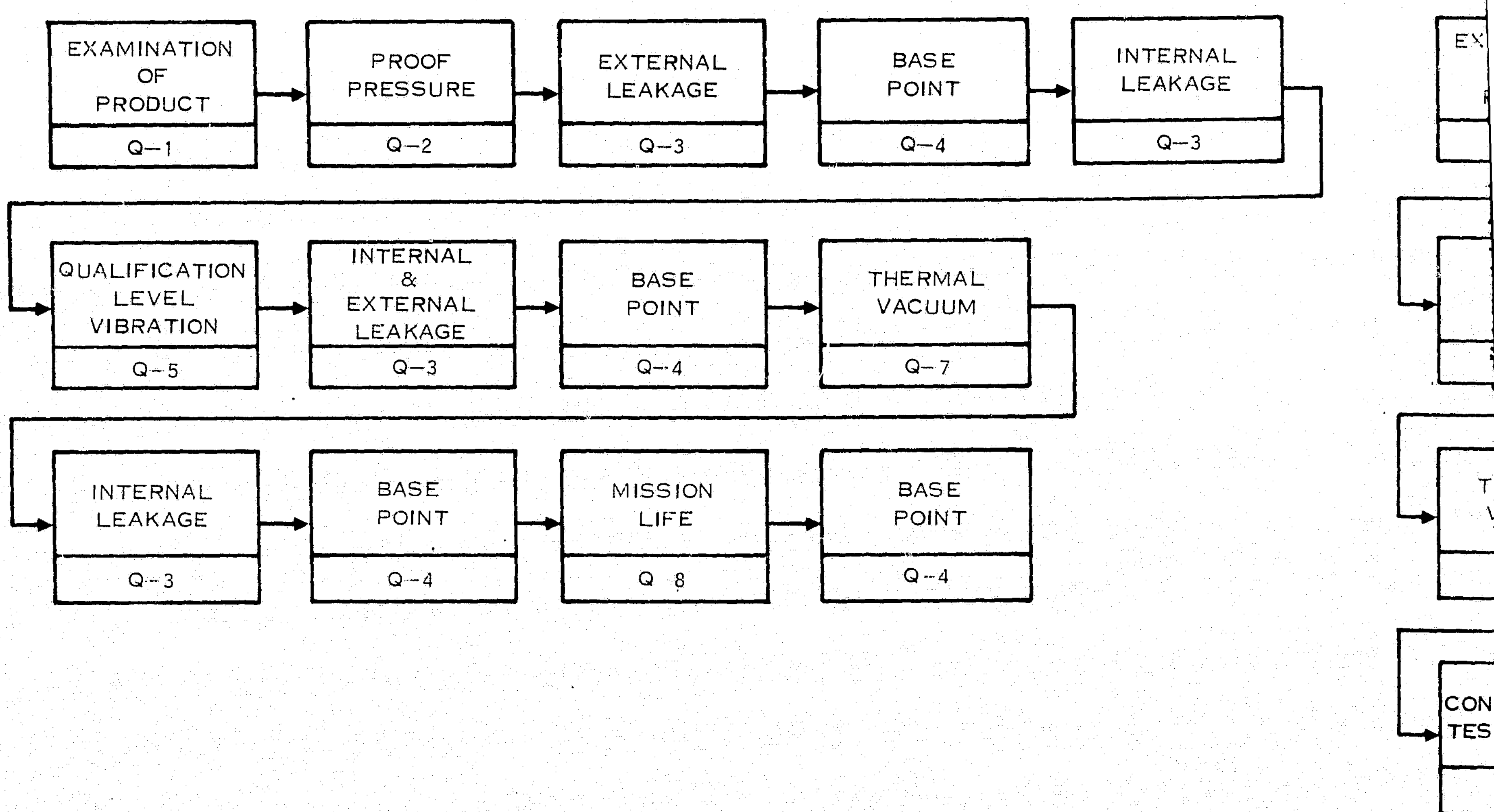


FIGURE 5.6.1-2. VIBRATION LEVELS - DEMONSTRATED VS REQUIRED

FOLDOUT FRAME

ROCKET ENGINE ASSEMBLY – PERFORMANCE MAPPING



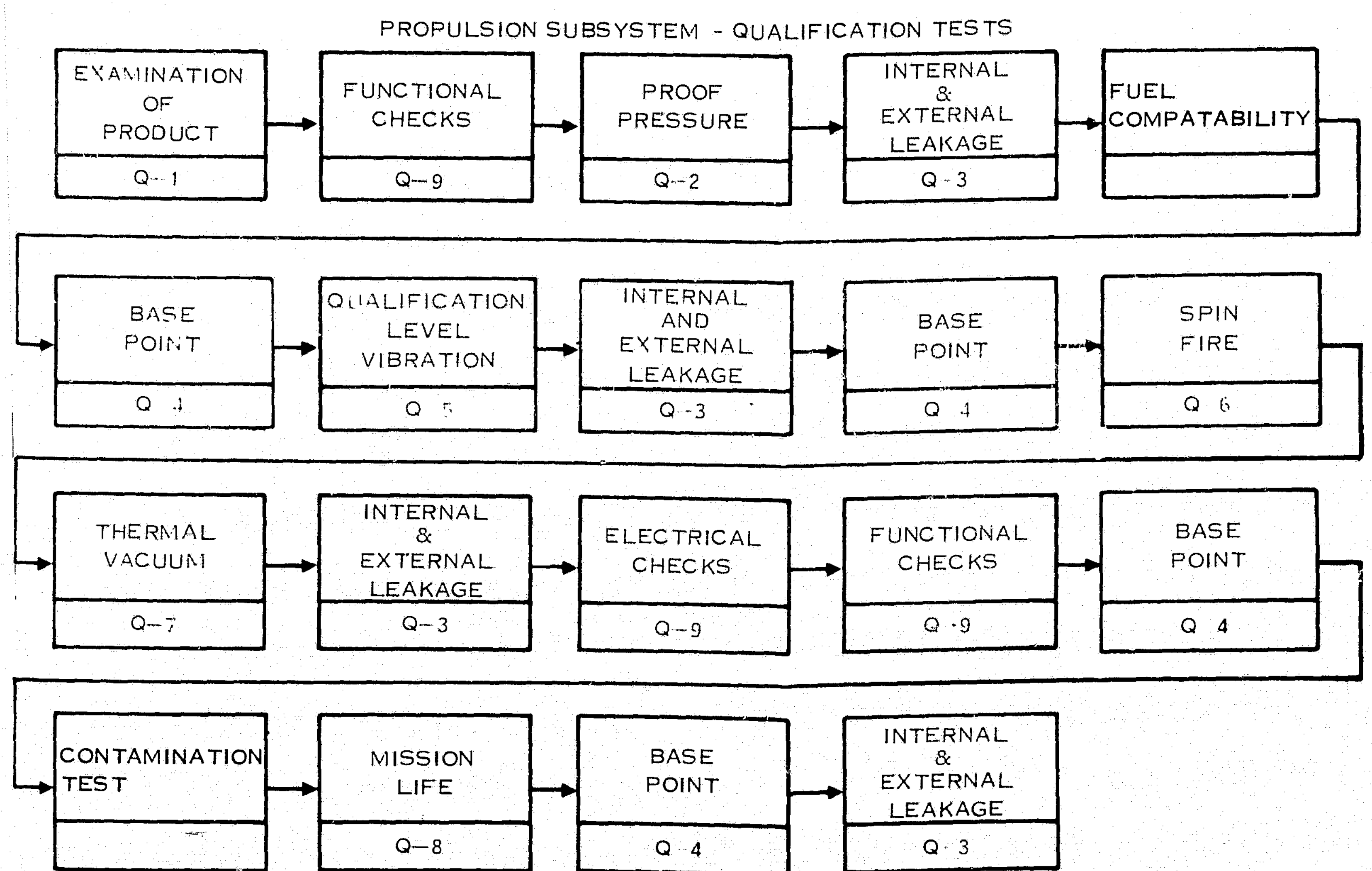
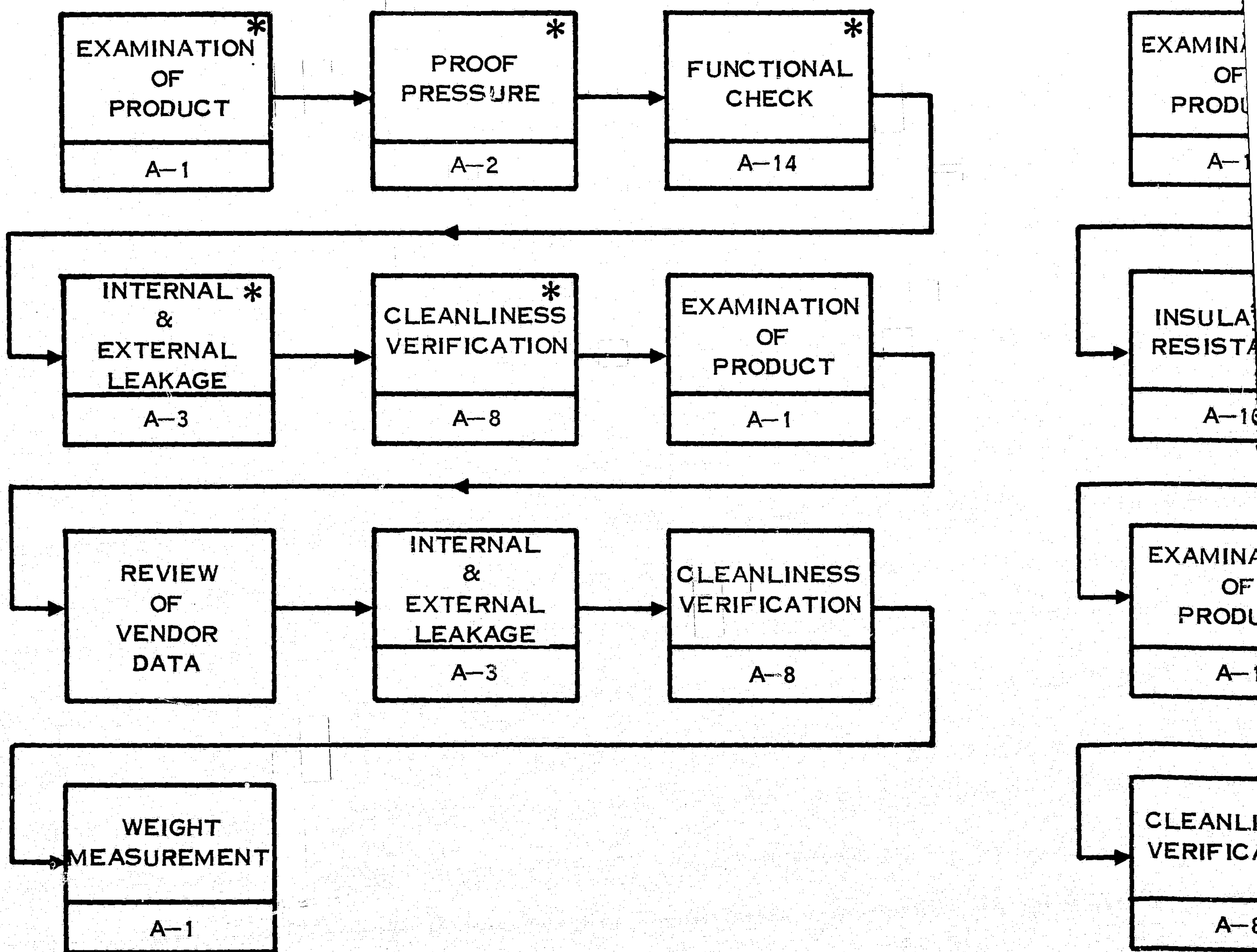


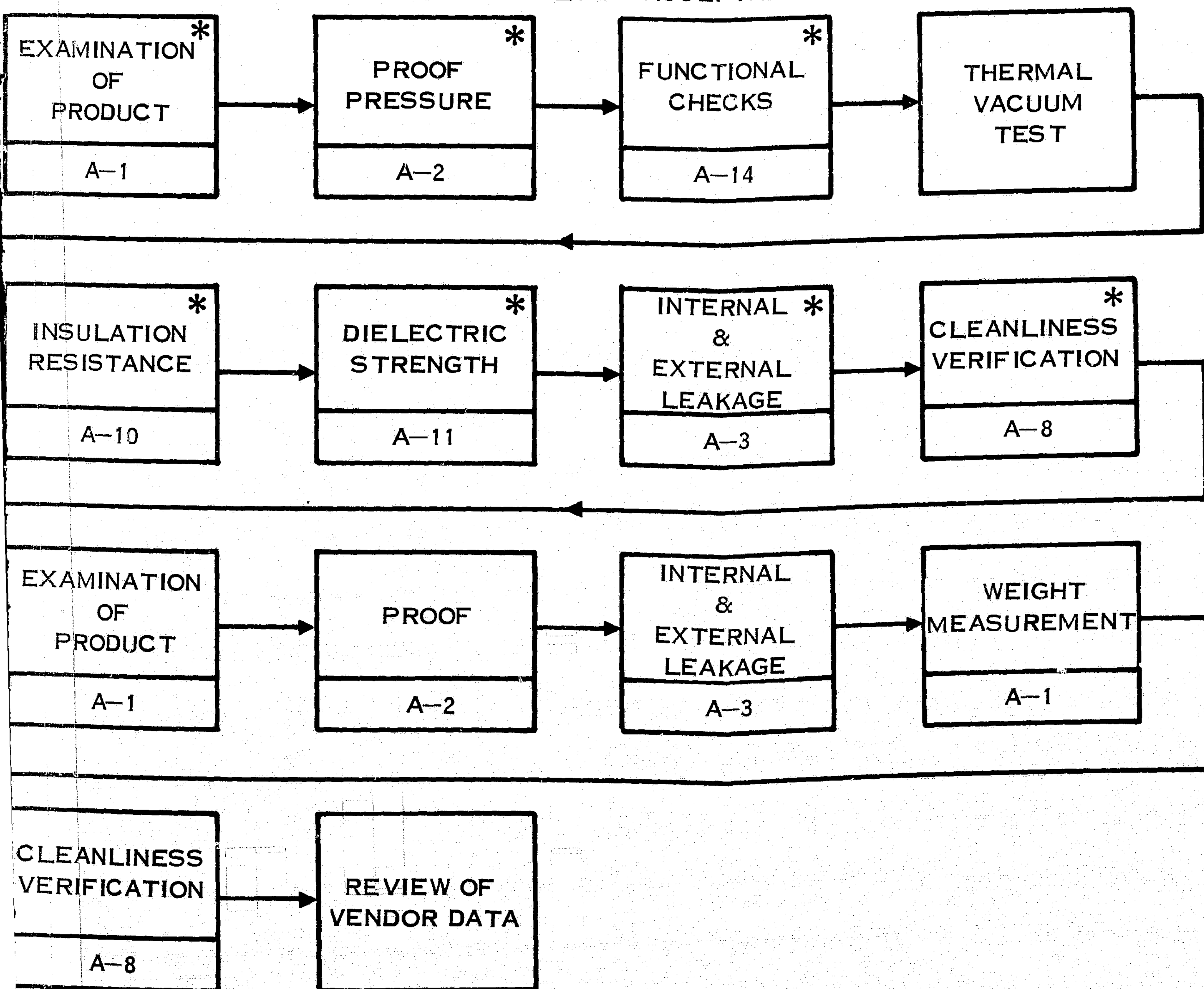
FIGURE 5.6.2–1. QUALIFICATION TEST SEQUENCE

FULL & DRAIN VALVE — ACCEPTANCE TEST



*DENOTES TESTS PERFORMED AT

LATCHING SOLENOID VALVE — ACCEPTANCE TEST



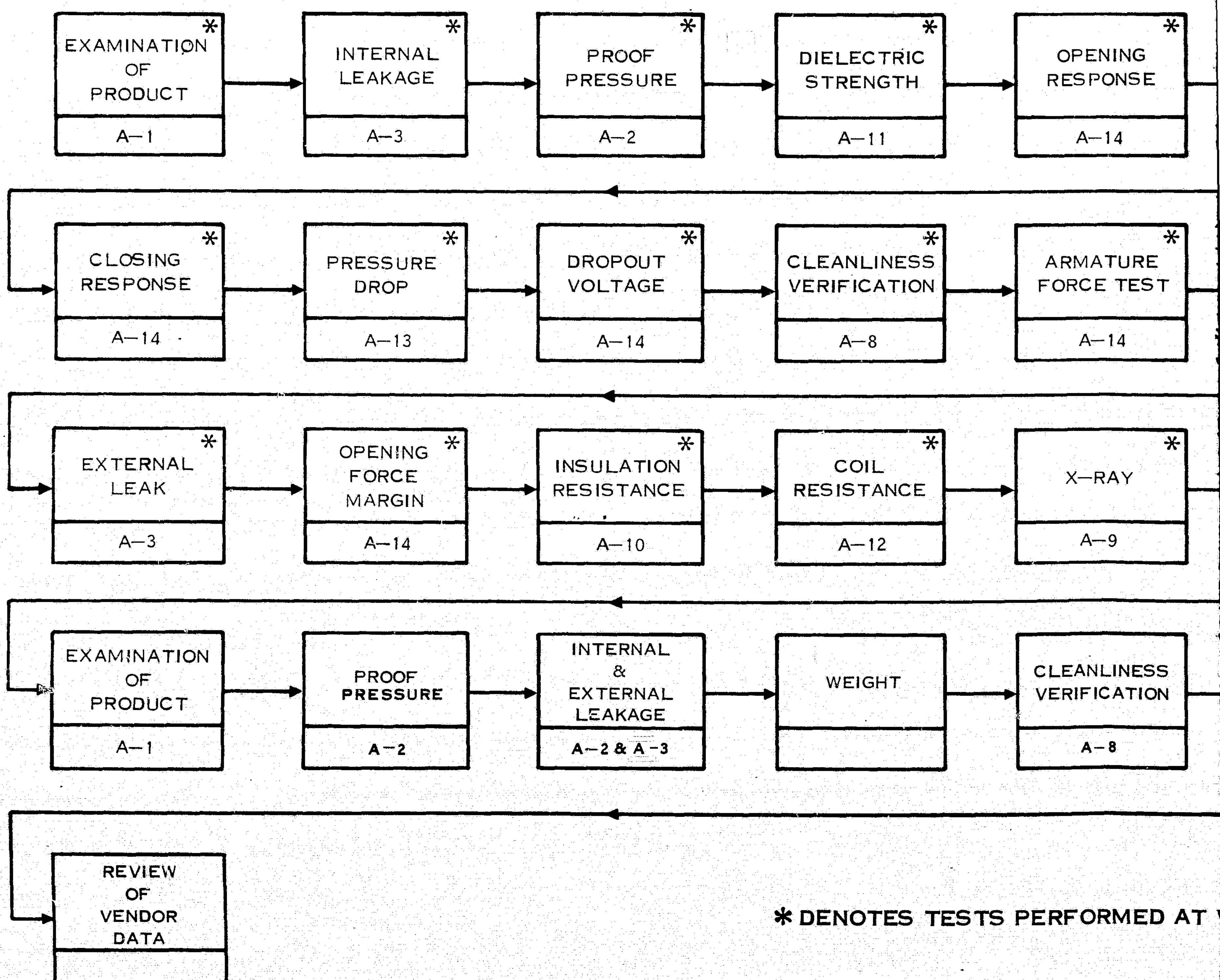
MED AT VENDOR FACILITY

FIGURE 5.6.3–1. COMPONENT ACCEPTANCE TESTS

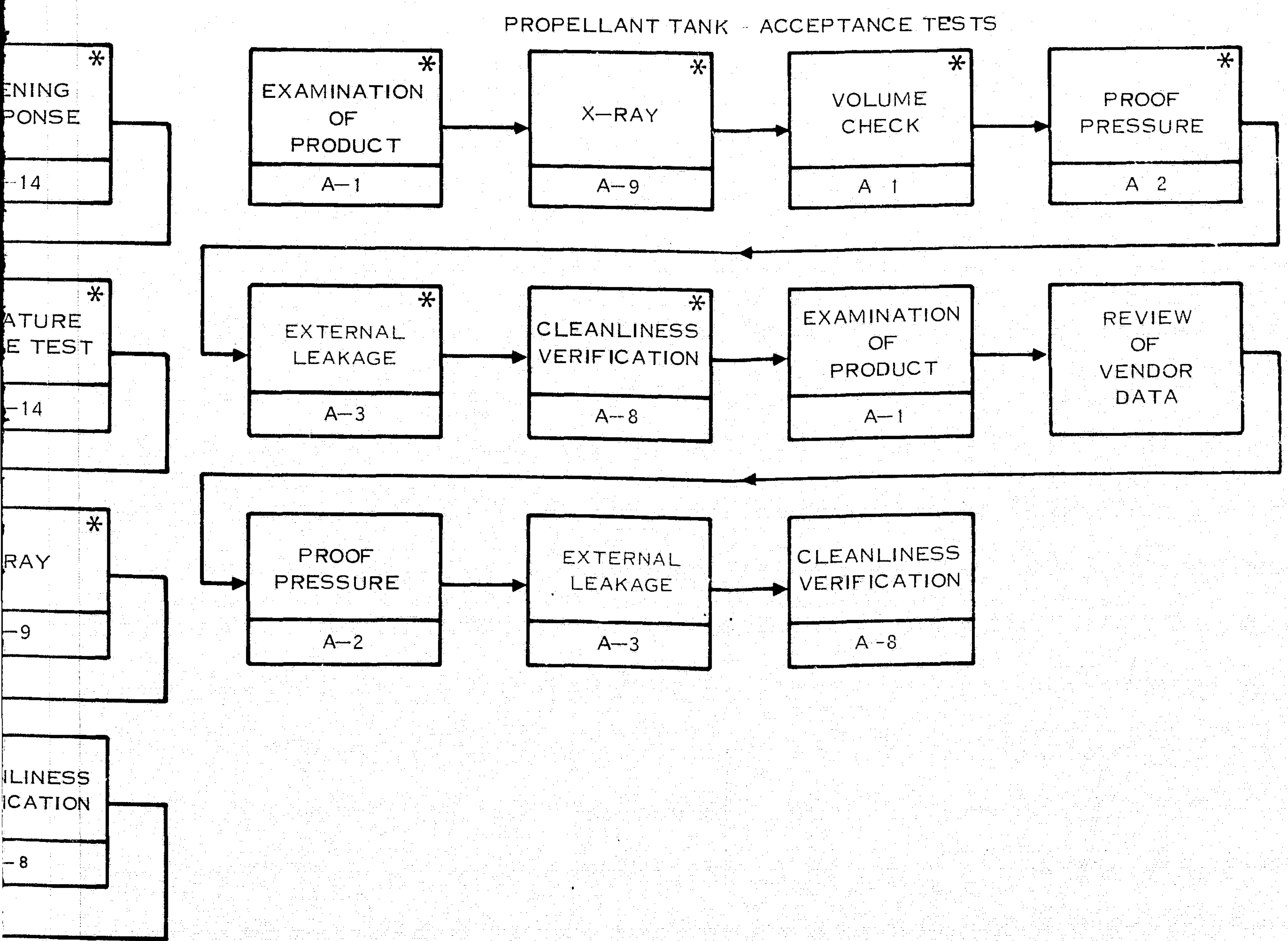
FOLDOUT FRAME

Hamilton Standard DIVISION OF UNITED AIRCRAFT CORPORATION

PROPELLANT VALVE ACCEPTANCE TESTS



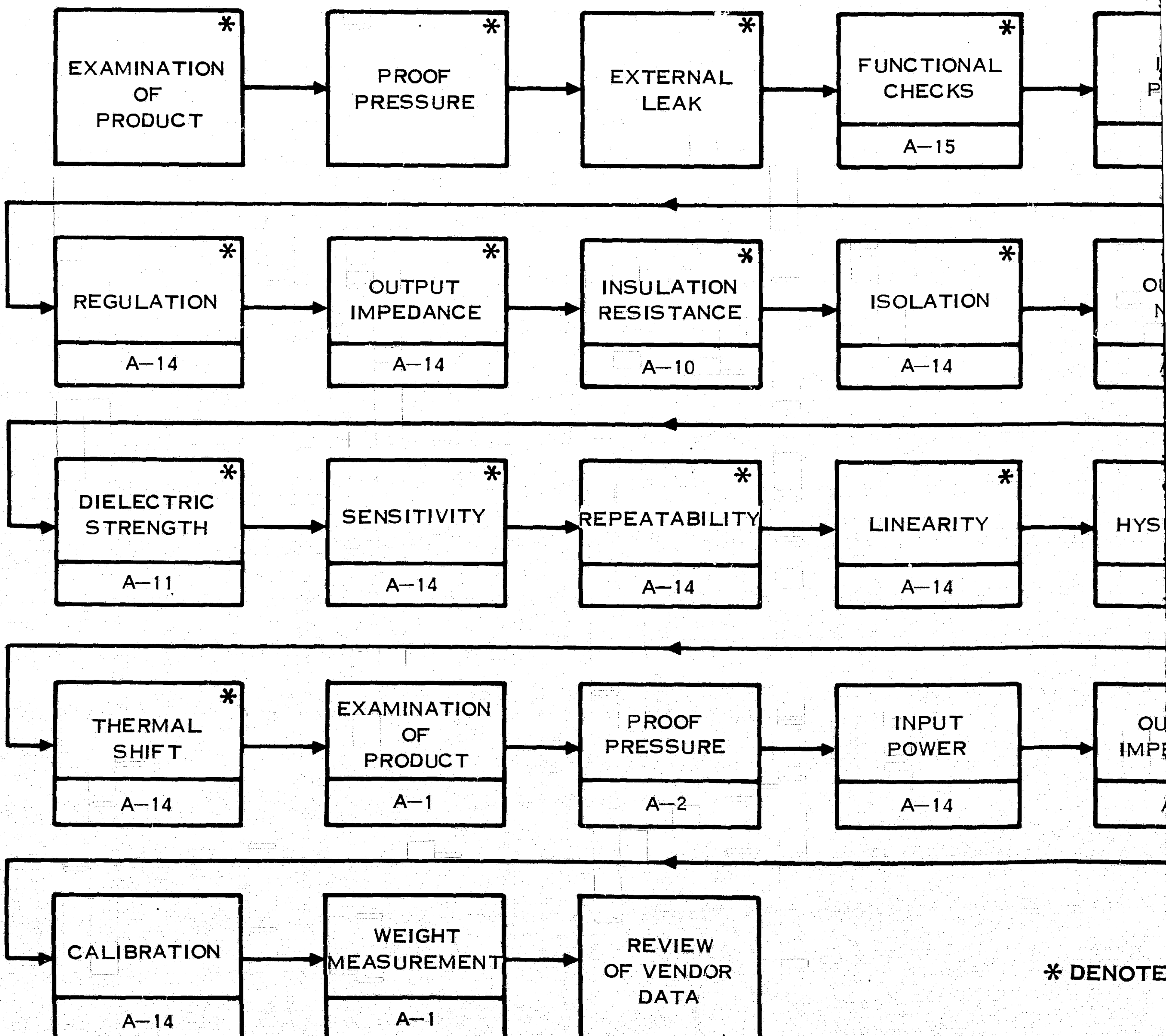
* DENOTES TESTS PERFORMED AT V

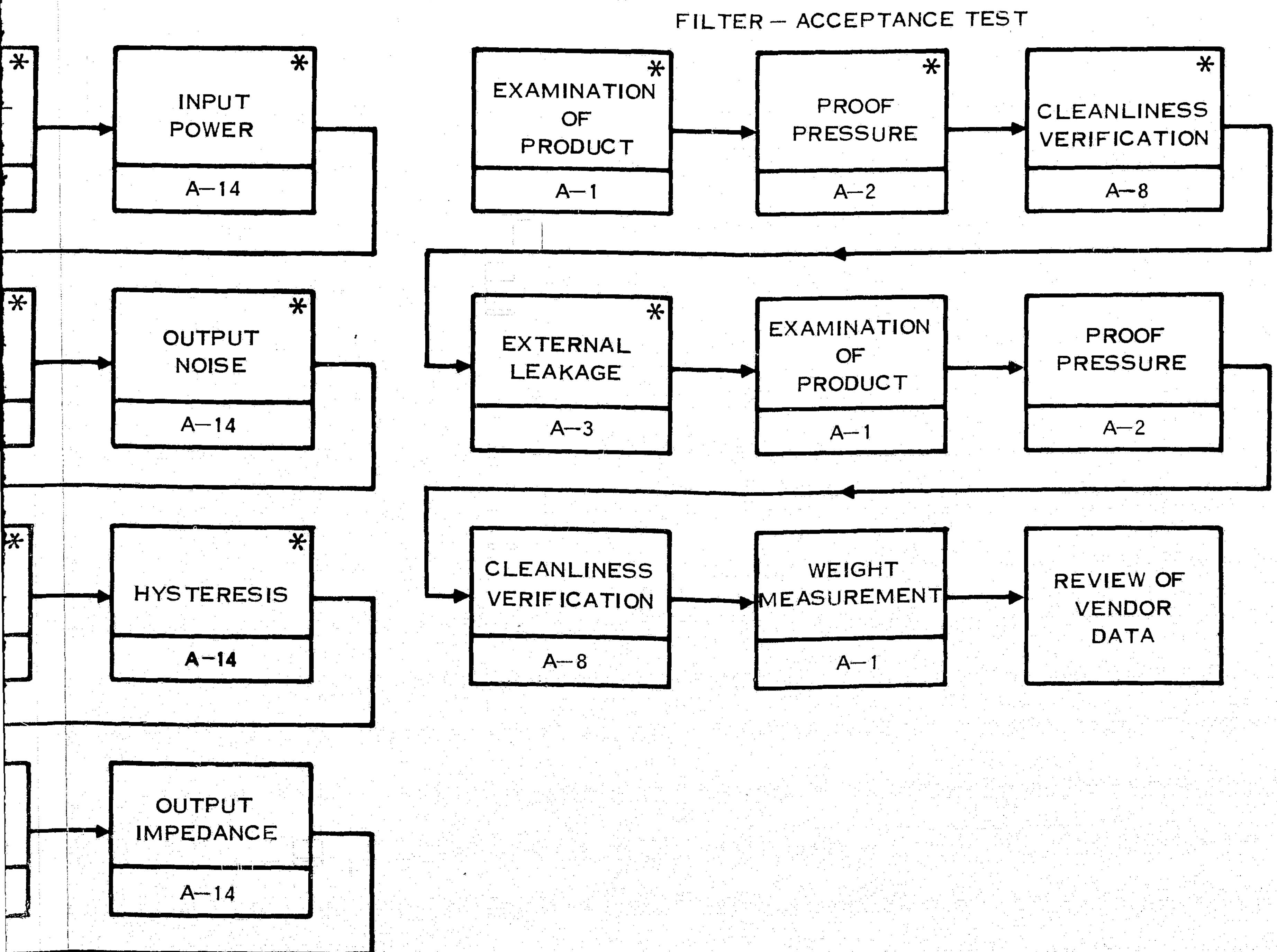


RMED AT VENDOR FACILITY

FIGURE 5.6.3-1. COMPONENT ACCEPTANCE TESTS

PRESSURE TRANSDUCER — ACCEPTANCE TEST





* DENOTES TESTS PERFORMED AT VENDOR FACILITY

FIGURE 5.6.3—1. ACCEPTANCE TESTS

TABLE 5.6.2-I. QUALIFICATION TEST SUMMARY DESCRIPTION

Test No.	Test Type	Objective	Success Criteria
Q-1	Examination of Product	Compliance with Drawing and Specification Requirements	Meet Drawing and Specification Requirements
Q-2	Proof Pressure	To Demonstrate Structural Integrity of Unit	No Permanent Deformation
Q-3	Leakage - External and Internal	To Verify that Leakage Rates are Within Specification Requirements	Leakage Does not Exceed Specification Limits
Q-4	Base Point	Obtain Operating Value of Unit Performance	No Out of Specification Change in Performance as Result of Any Intermediary Testing
Q-5	Qualification Vibration	Determine Effect of Vibration on Structural Integrity	No Failure, Malfunction or Out of Tolerance Performance Shall Occur Structural Integrity Shall be Maintained
Q-6	Spin Fire	To Verify System Functional Capability at Mission Spin Rates	System Successfully Meets Specification Requirements With no Out of Tolerance Performance
Q-7	Thermal Vacuum	To Demonstrate the Capability of the System to Operate in a Simulated Space Thermal Environment	No Out of Specification Degeneration Shall Occur as a Result of Operation of the System at the Extremes of Environment
Q-8	Mission Life	To Acquire Sufficient Operating Time to Meet System Mission Requirements	System Successfully Meets Life Requirements with no Out of Tolerance Performance

TABLE 5.6.3-I. ACCEPTANCE TEST SUMMARY DESCRIPTION

Test No.	Test Type	Objective	Success Criteria
A-1	Examination of Product	Compliance with Drawing and Specification	Meets all Drawing and Specification Requirements
A-2	Proof Pressure	To Demonstrate Structural Integrity of Unit	No Permanent Deformation
A-3	Leakage - External and Internal	To Verify Leakage Rates Are Within Specification Requirements	Leakage Does Not Exceed Specification Limits
A-4	Base Point	Obtain Operating Value of Unit Performance	No Out of Specification Change in Performance as a Result of Intermediary Testing
A-5	Acceptance Vibration	Determine Effect of Vibration on Structural Integrity	No Failure, Malfunction or Out of Tolerance Performance Shall Occur Structural Integrity Shall be Maintained
A-6	Performance	Demonstrate Acceptable Performance	No Out of Specification Perf.
A-7	Thrust Calibration	Match Engine Thrust Levels	Thrust Levels Within Tolerance for Given Inlet Pressure
A-8	Cleanliness Verification	Verification of Contamination Level	Measured Particulate Count in Tolerance
A-9	Radiographic Inspection	Verify Integrity of Welds	No Cracks, Inclusions or Imperfections
A-10	Insulation Resistance	Prove Test Unit Insulation Adequate	Measured Resistance Over Specification Requirement
A-11	Dielectric Strength	Prove Test Unit Housing Leakage Resistance	Measured Current Leakage Within Tolerance
A-12	Coil Resistance	Determine Test Limit Coil Resistance	Resistance Within Specified Tolerance
A-13	Flow	Check Unit Flow and Pressure Drops	Flow Within Specification Tolerance
A-14	Function Checks	Verification of Unit Operation	Successful Operation With no Out of Tolerance Parameters

5.7 Thermal Analysis

On the basis of the preliminary thermal information presented in the GSFC subsystem specification, all internally mounted equipment of the propulsion subsystem can be thermally controlled using passive design techniques. Based on this technical assessment, the thermal analysis performed during the study was directed toward aspects concerned with the Rocket Engine Assemblies (REA's) and their thermal interface with the spacecraft.

A study was conducted to evaluate the operational characteristics of the IDCSP/A engine assembly with the thermal interface defined for the Planetary Explorer mission in the GSFC subsystem specification. The engine installations must satisfy the following criteria:

1. The engine package external temperature must remain below 215°C (420°F) and have a low emittance ($\epsilon \leq .2$).

2. The engine mount temperature must not exceed 215°C with a mount resistance of 50°C/watt (26.4 °F hr/Btu).

3. The engine valve must not overheat during soakback.

4. Propellant at the engine valve inlet must not freeze at low compartment temperatures.

The results of this study are summarized in Table 5.7-I, along with a listing of assumptions used. These results are discussed below.

MAXIMUM SURFACE TEMPERATURE DURING VENUSIAN ORBIT:

The surface temperature and emissivity constraints mentioned above dictate the use of thermal shielding around the engine to reduce solar heat pick-up and to limit the external temperature of the engine during firing and soakback periods. The worst case, from an external temperature standpoint, are the spin engines which overhang the vehicle and are, therefore, heated for the full rotational period. With an emissivity of .2 and an α = .4, for the outer insulation covering, the solar soak temperature of the shield becomes 298°F for maximum orbital solar and albedo input, and 228°F for average orbit input. This is excessively hot for the non-firing case. Application of a radiator coating (ϵ = .7, α = .2) over the outer quadrants of these shields which look into space will aid heat rejection to space while lowering solar energy absorption. Although this approach technically violates the specified $\epsilon \leq .2$, the intent of the specification is not violated, since the low emissivity surface is retained on the portions of the shield which have significant radiative coupling to the spacecraft surface. The equilibrium solar soak temperature for the coated shield becomes 118°F for average orbital heat load. For a shield of this type, the outer skin would have to be fabricated from a material with high thermal

5.7 (continued)

conductivity in order to distribute engine firing heat evenly over the shield surface. Insulation requirements for the inner portions of the shield were determined under the constraint that the outer surface remain below 420°F during steady state engine firing while being simultaneously exposed to orbital solar plus albedo heat input. The results indicate that a one-half inch thick layer of MIN-K 2000 surrounded by a metal shroud with the previously described characteristics will meet the external temperature criterion.

MOUNTING FLANGE SOAKBACK TEMPERATURE:

Detailed thermal analyses performed for other applications of the IDCSP/A engine (with radiation shields) have shown that flange soakback temperatures will reach 425°F at approximately 10 minutes following shut down from a steady state firing if the engine mount resistance is 15°F/Btu-hr. The Planetary Explorer subsystem specification calls out a mount resistance of 26.4 °F/Btu-hr. A mount resistance this high will result in soakback flange (and propellant manifold) temperatures of roughly 500°F which are excessive from the standpoint of avoiding propellant boiling in the engine manifold, and overheating of the spacecraft structure at the engine interface. It is recommended that this requirement be relaxed to 15°F/Btu-hr.

ENGINE PROPELLANT VALVE SOAKBACK TEMPERATURE:

The IDCSP/A engine configuration effectively isolates the propellant valve from the hotter engine parts through the high thermal impedance of the propellant feed tube. Maximum valve soakback temperatures for the Planetary Explorer application will be approximately 200°F.

INTERNALLY MOUNTED EQUIPMENT

The following discussion concerns equipment which is physically located in compartments within the spacecraft structure.

1. Engine Valves - Radial Engines

   The equipment compartment temperature extremes given in Table 3 of the GSFC subsystem specification are not sufficiently severe (on the cold side) to warrant active heating for the engine valves if the small amount of solar heat picked up from the external portions of the engines is taken into account.

2. Tankage

   If the assumption is made that, between the two sets of upper compartment temperature extremes labeled "completely insulated" (-20°C - 25°C) and "partially insulated" (+13°C - 76°C), there is an intermediate situation (+5°C - 65°C) which can be achieved, then nothing in the way of thermal protection for the tankage is required.

TABLE 5.7-I. SUMMARY RESULTS OF THERMAL ANALYSIS

Equipment Condition	Temperature	Mission Event	Assumptions
Maximum temperature of engine external surfaces	400°F	Venus orbit - full solar plus orbital avg. albedo incident heat flux with engine firing steady state	1/2 inch of MIN-K conformal insulation enclosed by high conductivity can with $E \leq .2$ on spacecraft side, and $E = .7/\alpha$ solar = .2 on sun side (achieved with radiator coating on sun side and polished aluminum on vehicle side)
Maximum temperature at engine mounting flange	420°F	Soakback after engine shutdown following steady state firing under conditions described above	Engine mount thermal resistance = 15 °F hr/Btu instead of spec value of 26.4 °F hr/Btu (Note: 26.4 °F hr/Btu will yield $\approx$500°F soakback temperature)
Maximum valve soakback temperature	$\approx$200°F	Same as above	Same as above
Engine valves for radial engines which are buried in equipment compartment	40°-130°F	All	Small amount of heat picked up by valves from external parts of engine
Propellant tanks	40°-140°F	All	There is a "middle ground" between the two sets of temperature ranges in Table 3 of spec. headed "partially insulated" and "fully insulated"

5.8 Engine Exhaust Plume Impingement Effects

In spacecraft applications utilizing monopropellant hydrazine rocket engines, three aspects pertaining to the engine exhaust have to be taken into consideration. These are: (1) contamination of sensitive surfaces, (2) thermal effects, and (3) induced disturbing torques.

This section presents the results of a study to determine the effect of plume impingement on each of the 15 candidate systems presently being evaluated for the Planetary Explorer application. The 15 candidate system configurations utilize the three basic types of mounting which are described below.

- Type I Engine Mounting - Engines of this type are mounted above and/or below the main solar cell arrays on the spacecraft center structure and outboard of it, firing tangentially.

- Type II Engine Mounting - Engines of this type are mounted on the circumference of the main body of the spacecraft and fire perpendicularly outward.

- Type III Engine Mounting - Engines of this type are mounted on the circumference of the main body of the spacecraft and fire tangentially.

The Type II mounting configuration yields the lowest possibility of plume impingement and so will not be considered.

At specimen temperatures below -17°C, ammonia, water and traces of hydrazine in the plume may condense on vehicle surfaces. Of these, hydrazine has the lowest vapor pressure and will thus be the slowest to sublimate after the engines are turned off. However, Hittman Associates Inc., under NASA Contract NAS 5-11826 report that at temperatures above -73°C hydrazine will sublimate in a short period of time. It is thus concluded that for surfaces above -17°C plume condensation is not a problem and for surfaces above -73°C plume condensation may cause a temporary coating during engine firing but the coatings will rapidly sublimate after engine shutdown.

The solar cell array varies in temperature from 15°C to 70°C as it cruises toward Venus. Once in orbit, the greatest temperature variation, and the lowest solar cell temperature, occur for the Maximum Shadow orbit when temperatures vary from +100°C to -100°C over a period of approximately three hours. For other orbits, the lowest solar cell temperature achieved is 68°C. Exhaust product condensation would not appear to be a problem even for the Maximum Shadow orbit since any condensate

5.8 (Continued)

would only build up when the cells are in shadow and are therefore dormant, and when heat-up occurs, this condensate will quickly evaporate.

The contamination effects on space-borne equipment such as thermal control paint, solar cells and optics were investigated by the Air Force Rocket Propulsion Laboratory (Reference 3). Using a Hamilton Standard supplied 25 lb_f hydrazine rocket engine, a series of more than 200 firings were conducted at each test position. The severity of the exhaust impingement effects was evaluated by: (1) measuring any change in the initial ratio of absorption and emittance of thermal control paint, (2) observing change in solar cell output and physical damage incurred, (3) observing image distortion and loss of transmittance through the optics as well as physical damage incurred, and (4) attempting to identify the exhaust plume contaminants. The thermal paint specimen which showed the greatest change in absorption value (specimens varied from .04% to 18.5% depending on axial location) was located only 5 inches from the nozzle of the 25 lb_f engine. The optics and solar cell tests consisted of two different positions varying the distance of the test panel from the nozzle. In general, only very slight degradation of the test specimens occurred. In some cases, the ammonia, which is a product of the hydrazine reaction, appeared to have cleaned some of the optical specimens.

The effects of hydrazine exhaust plumes and propellant spills was also investigated in Reference 4 at LTV Aerospace Corporation. Using a two pound thrust engine, tests were run on the following materials:

- Polished 1060 aluminum
- Oxidized 1060 aluminum
- Zinc oxide thermal coating
- Potassium silicate thermal coating
- Aluminized Mylar (Schjel-Clad)
- Optical Glass
- Finned Copper Raschel Weave Lace
- Chromel Cloth
- Reinforced Polyethylene

No deleterious effects were found due to hydrazine spillage. During plume impingement testing, samples were exposed to pulses of varying length while located at different distances from the nozzle, although all the samples including the 1 mil Mylar withstood pulsed operation of the engine with the specimen located 5 inches from the nozzle. When on-times were increased to 1 second, only the polyethylene was damaged. Data for the finned copper lace was presented and showed a 500°F temperature rise for 1 second duration firing at a nozzle-specimen distance of 5 inches, a 400°F rise at 10 inch distance and a 325°F rise at 15 inch nozzle to specimen distance.

0

5.8 (Continued)

In order to determine the possible heating rates from the plume impingement for the configurations being considered, the analysis presented in Reference 1 was applied to the Type I mounting configuration.

Using the following equations, the density ratio profile of the exhaust plume may be determined:

$$\frac{V_e}{V_{max}} = Me \left[\frac{\gamma - 1}{2} \left(1 + \frac{\gamma - 1}{2} M_e^2\right)^{-1} \right]^{1/2}$$

$$\frac{C_F}{C_{F_{max}}} = 1/2\ (1 + \mathrm{Cos}\ \Theta e)\ \frac{V_e}{V_{max}} \left[1 + (\gamma\ M_e^2)^{-1} \right]$$

$$\delta = \left[\sqrt{\pi}\ \left(1 - \frac{C_F}{C_{F_{max}}}\right) \right]^{-1}$$

$$X = d^* \left(\frac{B\ \rho_o}{\rho} \right)^{1/2}$$

$$X = X\ \mathrm{Cos}\ \Theta\ \exp \left\{ - \frac{\delta 2}{2} \left[1 - \mathrm{Cos}\ \Theta \right]^2 \right\}$$

where:

- V_e = exit velocity
- Me = Mach no. at exit
- γ = ratio of specific heats
- Θ_e = exit angle of the nozzle
- δ = plume spreading parameter

The ρ_o/ρ was then plotted and appears in Figure 5.8.0-1. With the density profile known, the heating values may be determined as follows:

$$\rho_o = (\rho^*) \left(1 + \frac{\gamma - 1}{2} \right)^{\frac{1}{\gamma - 1}} ;\ \rho = \frac{\dot{M}}{U^*\ A^*} ;\ U^* = \sqrt{\frac{2\delta R T_o}{\delta + 1}}$$

With this expression, the local density may be determined, and since the exhaust velocity approaches a constant as the pressure approaches zero, the velocity U may be determined from:

$$U = U^* \left(\frac{2}{\gamma + 1} \right)^{\frac{1}{\gamma - 1}} \frac{\delta}{2B\sqrt{\pi}}$$

where B is a constant which is a function of the nozzle geometry and gas properties and is found tabulated in Reference 1.

5.8 (Continued)

An estimate of the heat transfer coefficient may now be obtained from the analysis presented in References 5 and 6. Using the equations presented and the plume density distribution already derived, the local heating rates which may be experienced along the edge of the lower solar cell array may now be calculated and are presented in Figure 5.8.0-2.

To obtain an estimate of the disturbing torques induced by the plume impingement presented in Figure 5.8.0-1, the plume velocity is once again determined as in the previous discussion, $U = 1.33 \times 10^4$ ft/sec, the normal component of which is used to calculate the dynamic pressure over the impinged surface. For a "worst case" analysis, the highest impingement flow density, $\rho = 4.12 \times 10^{-9}$ lb/ft^2 and the greatest incidence angle $\Theta = 20°$ was used. At these high Mach numbers, the pressure force was calculated by assuming the normal component of momentum provided the total force.

The resulting pressure on the vehicle appears insignificantly low, $P_N = 8.3 \times 10^{-5}$ psia, so that disturbance torques need not be considered a problem for this application.

References

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2. Y.C. Brill, R.C. Stechman and R.J. Reis, Effect of Hydrazine Rocket Fuel on Spacecraft Materials, 14th Annual Meeting, The Institute of Environmental Sciences, St. Louis, Missouri, 1968.

3. P.J. Martinkovic, Monopropellant Exhaust Contamination Investigation, AFRPL-TR-69-72, April 1969.

4. F.T. Esenwein, S.C. Walker, Effects of Hydrazine Exhaust Plumes and Propellant Spills on Selected Spacecraft Materials, Missile and Space Division, LTV Aerospace Corporation, Dallas, Texas, L 708-0980, December 1967.

5. H.K. Cheng, A.L. Chang, Hypersonic Shock Layer at Low Reynolds Number - The Yawed Cylinder, Cornell Aeronautical Labs. ARL #62-453, October, 1962.

6. A. Peracchio, Heating Due to Rocket Exhaust Impingement, Hamilton Standard Report, HSIR 2212, August 1964.

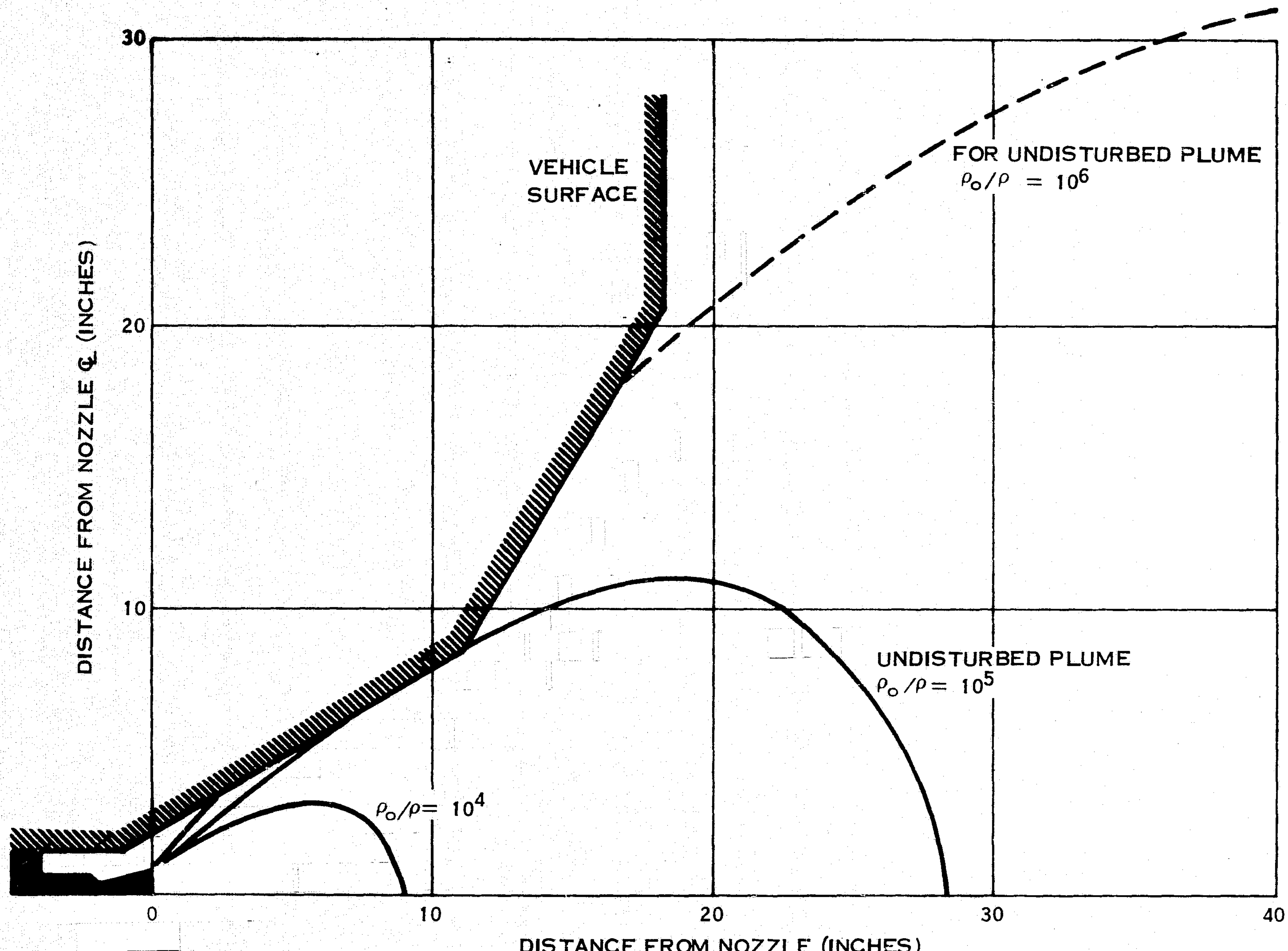

FIGURE 5.8.0-1. EXHAUST PLUME DENSITY

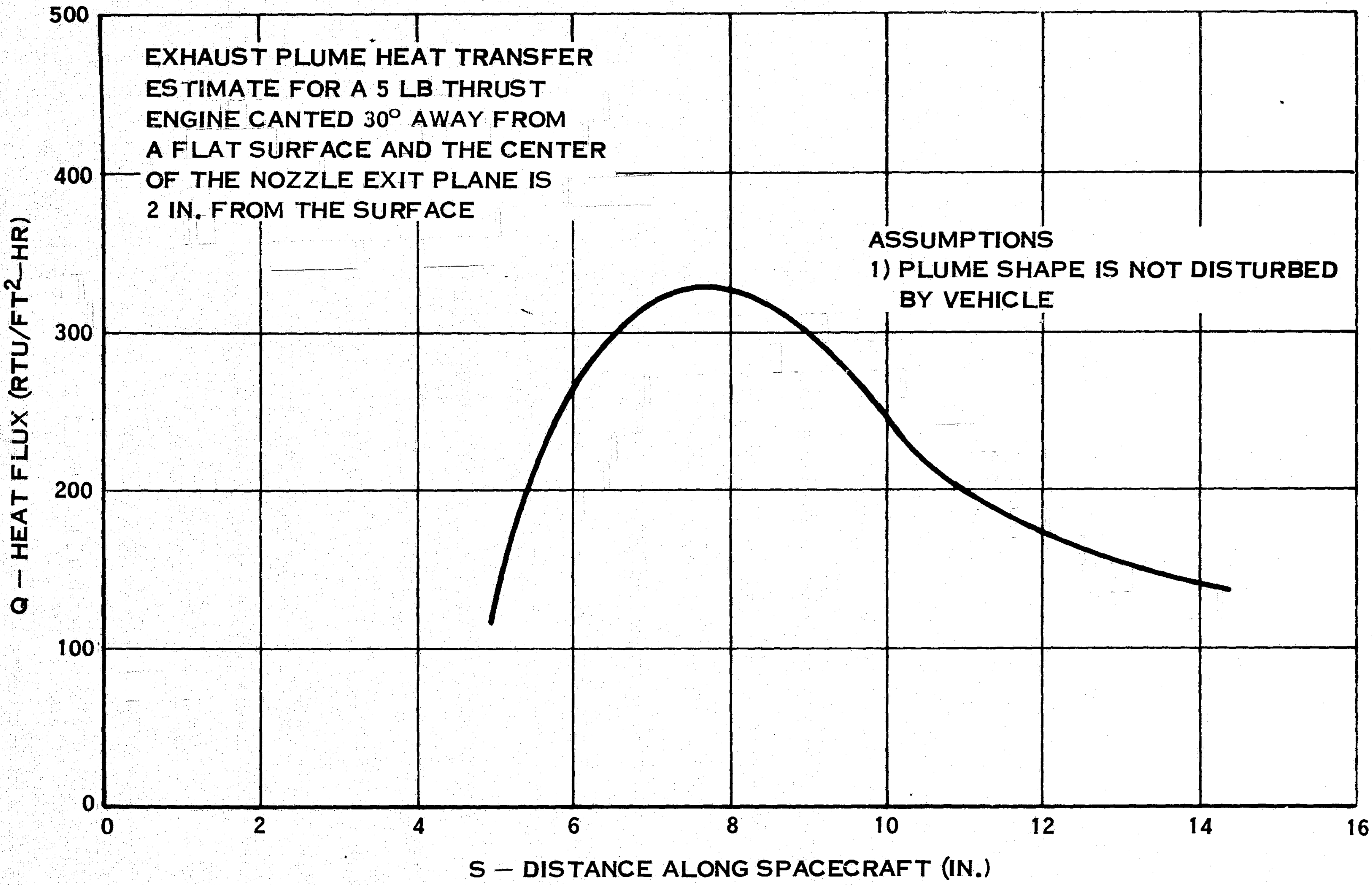

FIGURE 5.8.0—2. EXHAUST PLUME HEAT TRANSFER ESTIMATE

5.9 Propulsion Subsystem Leakage Analysis

A mission leakage analysis was performed on the candidate propulsion subsystems to determine the maximum pressurant and propellant leakage rate during mission launch and armed modes. The leakage rates used for the various components (fill and drain valve, tank, squib valve, filter, latching valves and engine assemblies) were based on the component qualified levels, and the rates for line joints (both mechanical and welded) represent values which are commonly attainable in practice. A summary of the leakage rates are tabulated in Table 5.9.0-I and 5.9.0-II. The propellant leakage rates, expressed in terms of gas leakage, include a breakdown for the various candidate propulsion subsystems whereas the pressurant rates, which are independent of the hardware variables associated with the candidate subsystems, are shown only as a function of the Orbiter and Probe missions. The maximum pressurant leakage rate is 4.0 $(10)^{-5}$ scc/sec GN_2 (.0032 lb GN_2/yr) for the Orbiter mission and 2.8 $(10)^{-5}$ scc/sec GN_2 (.0022 lb GN_2/yr) for the Probe.

These values satisfy the specification limit of .02 lb/yr of GN_2 pressurant leakage. The maximum total estimate for propellant leakage (expressed in terms of gas leakage) is 141.5 $(10)^{-5}$ scc/sec N_2, and represents the leakage of the 8 engine Orbiter configuration with the FS-4 feed system. The estimated equivalent liquid leakage for this system, based on this total gas leakage, is .23 lb/yr* which exceeds the specification value of .08 lb/yr. However, it should be noted that there is a very high degree of conservatism in this estimated value, and there is confidence that this leakage will actually never be experienced. The analysis assumes that the latching solenoid valves are continuously open and that each of the engine valves are leaking the maximum rate of 14 $(10)^{-5}$ scc/sec N_2 (.5 scc/hr N_2). In addition, the effect of dual seats in the propellant valves has not been considered, and the analysis assumes that one of the seats in each of the propellant valves has failed. In addition, practical experience has shown that no liquid leakage will occur where the gas leakage is below 10 scc/hr N_2. Since each of the components have rated values below this limit, experience predicts no liquid leakage from the system, thus, more conservatism is present in the analysis since practical experience is neglected and the theoretical values are used.

The analytical conversion from gas to liquid leakage was obtained using the leakage conversion nomograph (Per Leakage Testing Handbook #5-69-111 pg 6-15 NASA JPL). The conversion method was applied to the sum of all the component gas leakages which is the same as assuming the liquid leakage is linear with gas leakage. A separate analysis, however, shows this assumption to also be conservative. If the conversion were instead applied first to the individual rates and then summed, the total system leakage would be only .070 lb/yr which is below the specification value.

* Ref. Leakage Testing Handbook #S-69-111, NASA JPL per Contract NAS 7-396.

5.9 (Continued)

The total propellant leakage prior to arming is also listed and it can be seen that for those systems not utilizing squib valves for isolation, the leakages are comparable to those for the armed systems. The values shown are maximum and are contingent on maximum rates through the latching valves coupled with the condition of maximum leakage rates through all the propellant valves. Since the cumulative leakage of all of the propellant valves will be greater than the latching valves used for each system, the launch mode leakage will be governed by the leakage rates of the latching valves. The same degree of conservatism also applies to this analysis as was previously stated.

TABLE 5.9.0-II. PRESSURANT LEAKAGE RATIO - ORBITER AND PROBE MISSIONS

COMPONENT	Component Leakage $10^{-5} \frac{scc}{sec}$ N_2	ORBITER		PROBE	
		QTY	Leakage	QTY	Leakage
N_2 Fill & Drain Valve	.1	1	.1	1	.1
Tanks	.1	9	.9	6	.6
Line Joints Welded	.1	30	3.0	21	2.1
TOTAL		$4.0\ (10)^{-5}$ scc/sec N_2		$2.8\ (10)^{-5}$ scc/sec N_2	

FOLDOUT FRAME 1

TABLE 5.9.0-I. PROPELLANT LEAKAGE RATES - ORBITER AND PROBE

Component	Component Leakage $10^{-5}\ \frac{scc}{sec}\ N_2$	CANDIDATE PROPULSION SYSTEM I, II, III, VII, VIII, IX, XIII XIV & XV (P5, P10 & P13) Configurat						
		FS-2			FS-4			
		Qty	Launch Leakage	Armed Leakage	Qty	Launch Leakage	Armed Leakage	Qty
. Propellant Feed System Valve Upstream of Isolation								
N_2H_4 Fill & Drain Valve	.1	1	.1	.1	1	.1	.1	1
Pressure Transducer	.1	1	.1	.1	1	.1	.1	1
Line Joint (Welded)	.1	(a)	6.0 4.2(b)	6.0 4.2(b)	(a)	6.0 4.2(b)	6.0 4.2(b)	(a)
Line Joint (Mech)	2.0	1	2.0	2.0	1	2.0	2.0	1
. Propellant Feed Upstream of Engines								
Squib Valve	.1	-	N/A	N/A	-	N/A	N/A	1
Test Port	.1	2	-	.2	1	-	.1	1
Filter	.1	1	-	.1	1	-	.1	1
Latching Valve (N.O.)		-	N/A	N/A	2	-	-	2
Internal	39.0	-	-	-	-	-	-	-
External	.1	-	-	-	-	-	.2	-
Latching Valve (N.C.)		2	-	-	2	-	-	-
Internal	39.0	-	78.0	-	-	78.0	-	-
External	.1	-	.2	.2	-	.2	.2	-
Line Joints (Welded)	.1	17	1.7	1.7	21	2.1	2.1	15
. Propellant to Engines								
Engines		8	-	-	8	-	-	8
Internal	14.0	-	-	112.0	-	-	112.0	-
External	.1	-	-	.8	-	-	.8	-
Line Joints (Welded)	.1	18	-	1.8	18	-	1.8	18
Line Joints (Mech)	2.0	8	-	16.0	8	-	16.0	8
TOTALS $(10)^{-5}\ \frac{scc}{sec}\ N_2$: ORBITER			88.1	141.0		88.5	141.5	
PROBE			86.3	139.2		86.7	139.7	

NOTES: a) Orbiter - 60 Probe - 42
b) Value is for Probe

FOLDOUT FRAME 2

BITER AND PROBE MISSION

LSION SYSTEM III, IX, XIII P13) Configurations				CANDIDATE PROPULSION SYSTEM IV, V, VI, X, XI & XII (P7 & P12 Configurations)								
-4	FS-8			FS-2			FS-4			FS-8		
Armed Leakage	Qty	Launch Leakage	Armed Leakage	Qty	Launch Leakage	Armed Leakage	Qty	Launch Leakage	Armed Leakage	Qty	Launch Leakage	Armed Leakage
.1	1	.1	.1	1	.1	.1	1	.1	.1	1	.1	.1
.1	1	.1	.1	1	.1	.1	1	.1	.1	1	.1	.1
6.0	(a)	6.0	6.0	(a)	6.0	6.0	(a)	6.0	6.0	(a)	6.0	6.0
4.2(b)		4.2(b)	4.2(b)		4.2(b)	(b)		4.2(b)	4.2(b)		4.2(b)	4.2(b)
2.0	1	2.0	2.0	1	2.0	2.0	1	2.0	2.0	1	2.0	2.0
N/A	1	.1	.1	-	N/A	N/A	-	N/A	N/A	1	.1	.1
.1	1	-	.1	2	-	.2	1	-	.1	1	-	.1
.1	1	-	.1	1	-	.1	1	-	.1	1	-	.1
-	2	-	-	-	N/A	N/A	2	-	-	2	-	-
-	-	-	-	-	-	-	-	-	-	-	-	-
.2	-	-	.2	-	-	-	-	-	.2	-	-	.2
-	-	N/A	N/A	2	-	-	2	-	-	-	N/A	N/A
-	-	-	-	-	78.0	-	-	78.0	-	-	-	-
.2	-	-	-	-	.2	.2	-	.2	.2	-	-	-
2.1	15	-	1.5	17	1.7	1.7	21	2.1	2.7	15	-	1.5
-	8	-	-	6	-	-	6	-	-	6	-	-
112.0	-	-	112.0	-	-	84.0	-	-	84.0	-	-	84.0
.8	-	-	.8	-	-	.6	-	-	.6	-	-	.6
1.8	18	-	1.8	12	.2	1.2	12	-	1.2	12	-	1.2
16.0	8	-	16.0	6	-	12.0	6	-	12.0	6	-	12.0
141.5		8.3	140.8		88.1	108.2		88.5	108.7		8.3	108.0
139.7		6.5	139.0		86.3	106.4		86.7	106.9		6.5	106.2

5.10 Environmental Effects

The environmental loading conditions for the Planetary Explorer spacecraft appear to be well within the requirements for spacecraft which are presently operational. Table 5.10.0-I summarizes the environmental characteristics for the propellant tanks and the engines. Components such as ordnance valves, latching solenoid valves, fill and drain valves, filters, and pressure sensors considered for the Planetary Explorer have been subjected to more severe environmental conditions in previous applications. The tanks and engines have also been exposed to similar environmental conditions for the IDCSP/A satellite application and the characteristics for the IDCSP/A are summarized in Table 5.10.0-I. Reviewing these characteristics relative to the Planetary Explorer requirements provides a basis for confidence in their use. All of the IDCSP/A levels, with the exception of sinusoidal vibration, are equal to or in excess of the Planetary Explorer requirements, however, the sinusoidal levels are satisfied by performing a comparison with the response levels encountered during random vibration. Note that the 19.5 g'rms represents a hard-mounted input therefore the response will be higher. A peak acceleration of 58.5 g's is estimated during random vibration which is somewhat higher than and therefore satisfies the Planetary Explorer sinusoidal requirements. Since the IDCSP/A shock levels are also expressed in terms of the input to the hardware, an estimate of the response was made, and the results indicate the levels to be considerably greater than what is required for the Planetary Explorer application.

TABLE 5.10-I. EQUIPMENT CAPABILITY - DYNAMIC LOADS

ITEM ENVIRONMENT IDCSP/A TANK	PLANETARY EXPLORER QUALIFICATION REQUIREMENTS	HARDWARE QUALIFICATION LEVELS
Sinusoidal Vibration	In 100-150 CPS Range 23.0 g's thrust dir. * 15.0 g's lateral dir.* * Response Using Q = 10	None, however, peak acceleration from random vibration = 58.5 g's
Random Vibration	5.8 g rms 4.0 min/axis (Response Using Q = 10 and assuming for = 120 CPS)	19.5 g rms 1 1/2 min/axis Hardmounted Input
Steady Acceleration	14.7 g's Thrust Dir. 3.0 g's Lateral Dir. + 7.0 g's Due to Spin = 120 RPM	30 g's -x) Thrust Dir. 15 g's +x) 8 g's y, z (Lateral)
Shock	38 g's Response	Input: 750 g half sine for .4 ± .1 ms Estimated Response: 140 g's Assuming f_n = 120 CPS
Acoustic	144 db Overall	147 db Overall
IDCSP/A ENGINE		
Sinusoidal Vibration	In 17-23 CPS Range: 6.0 g's Thrust Direction	None, however, peak acceleration from random vibration = 58.5 g's.
Random Vibration	15.7 g rms 4.0 min/axis (Response Using Q = 10 and Assuming f_n=350 CPS)	19.5 g rms 1 1/2 min/axis Hardmounted Input
Steady Acceleration	14.7 g's Thrust Dir. 3.0 g's Lateral Dir. +10.0 g's Due to Spin = 120 RPM	25 g's Each Direction
Shock	120 g's Response	Input: 750 g half sine for .4 ±.1 ms Estimated Response: 400 g's Assuming f_n = 350 CPS
Acoustic	144 db Overall	147 db Overall

5.11 Contamination Control

The control of contamination in a propulsion subsystem is a major contributor toward improving reliability. This area must be controlled throughout design, assembly and testing phases of the program. The following five basic sources of contaminants can affect the propulsion subsystem:

- Contamination in the hydrazine propellant
- Contamination in the nitrogen pressurant
- Contamination generated by the catalyst
- Internal contamination, such as machining chips and self-generated contamination
- Contamination from external sources such as the atmosphere

In general, the contamination control considerations for the propulsion subsystem, in order to avoid contamination induced failures from the above sources, are as follows:

- Design to minimize contamination sensitivity
- Design to minimize contaminant generation
- Design to provide ease of cleaning and subsequent monitoring
- Cleaning and packaging of all parts to cleaning specifications prior to assembly
- System assembly procedures controlled to minimize introduction of contaminants
- Inspection procedures affirming compliance with cleanliness requirements
- Maintenance and servicing procedures planned to minimize the introduction of contaminants

Downstream of the injector tubes are previously proven propellant diffusers which have 200 x 200 mesh screens capable of filtering 74 micron or large particles. Should catalyst fines migrate into the injector tubes during vibration or handling, they will be flushed clear during the initial firing of the thruster. Those few catalyst particles are 80 percent smaller than the injector tube diameter and are readily flushed clear. The valve is not susceptible to catalyst contamination due to the fact that it is either closed or has a positive flow at all times.

The recommended system filter has a contaminant holding capacity of 100 milligrams which is an order of magnitude greater than the possible quantity of contaminants which may remain in the system upstream of the filter after cleaning. This filter has a rating of 10 microns absolute assuring protection of the downstream components. Contaminant trapped by the filter will be considerably less than the holding capacity of the filter even if it were possible for all of the residual contaminants in the system to migrate to the filter.

5.11 (Continued)

Examples of specific steps taken to avoid contamination include:

- Selection of a propellant valve with no sliding parts
- Selection of materials which are compatible with hydrazine in order to avoid corrosion induced contamination
- Filter all fluids introduced into the subsystem through a 10 micron absolute filter to minimize the contaminant level introduced externally
- Seal all ports and openings with nylon film or protective fixtures when processing in non-controlled facilities
- Positive demonstration by way of fuel compatibility and contamination tests on all systems.

To provide a reliable propulsion subsystem for the Planetary Explorer applications the following cleanliness levels are recommended:

Components Upstream of Filter:

Particulate Size Range (Microns)	No. of Particles Allowed per Sq Ft of Surface Area
0 - 100	Unlimited *
100 - 150	7
150 - 175	1
+175	0

Components Downstream of Filter:

Particulate Size Range (Microns)	No. of Particles Allowed per Sq Ft of Surface Area
0 - 50	Unlimited **
50 - 75	5
75 - 90	1
90 +	0

* "Unlimited" shall be restricted to that quantity of contamination which is not considered excessive silting to the point where discoloration of the membrane filter is observed.

** Same as single asterisk above plus - in addition, no metal particles over 50 microns in size shall be allowed.

5.11 (Continued)

** "Unlimited" shall be restricted to that quantity of contamination which is not considered excessive silting to the point where discoloration of the membrane filter is observed. In addition, no metal particles over 50 microns in size shall be allowed.

More stringent cleanliness level are possible, but, are considered unnecessary and costly, the basic problem being of inherent need and maintenance of cleanliness rather than achievement for short periods of time. Previous program experience (IDCSP/A) has indicated reliable operation of the thrust chamber and propellant control valve at 75 micron maximum particle size cleanliness level which is less rigorous than what is recommended.

5.12 C.G. Tolerances

The candidate subsystems are all inherently unbalanced because of the modularized component panel, and the unsymmetrical engine locations on two of the engine placement concepts. Since weight is critical on the Planetary Explorer spacecraft it appears impractical to balance the propulsion subsystem for these major unbalances with "dead" weight. Coordination between the propulsion system manufacturer and GSFC can provide a compatible mass interface minimizing the addition of "dead" weight. The unbalance for the non-symmetrical placement of components would require the addition of a counterbalance of approximately 4.75 lbs for the worst case component panel, and approximately 10.0 lbs for the worst case engine and component panel concepts. This counterbalance would be strictly "dead" weight which could be better utilized in any number of areas.

Assuming the component panel and the engines can be counterbalanced by experiment packages, and final spacecraft balancing, the tanks then become the leading contributor the center of mass uncertainties. Table 5.12.0-I summarizes the center of mass deviation with, and without, subsystem balancing. From this table it can be observed that the C.G. deviation with, and without, propellant is slightly greater than the desired deviation of .015 inches when the unsymmetrical masses are counterbalanced. The C.G. deviations were estimated for the nine tank Orbiter arrangement and with engines in symmetrical locations, and are based on a ±.030 inch dimensional location accuracy to the center of mass for each component and a ±5 percent tolerance on the component weights. Before the tanks are assembled to the subsystem their weights will be determined and the tanks will then be distributed to minimize the unbalance for the tank and manifold assembly. The effective amount of unbalance contributed was statistically averaged, and the sum of these averages presented as the maximum net result. The tank and manifold assembly can be balanced before addition of the unsymmetrical components. This method of balancing will minimize the effect propulsion expulsion has on C.G. shift.

Propulsion subsystems with symmetrical engine positions can be balanced by adding a counterweight to the test setup offsetting unbalance caused by the component panel and components. This method can also be used for systems with unsymmetrical engine position. Balancing, at the tankage/manifold level can assure the highest accuracy of the subassembly which then will have the least disturbance effect on the spacecraft during propellant expulsion.

TABLE 5.12–I
CENTER OF MASS RADIAL
INACCURACY

PROPULSION SUBSYSTEM	COUNTERBALANCED	COUNTER–WEIGHT (LBS)	CENTER OF MASS RADIAL INACCURACY (MAX)	
			WITH PROPELLANT	W/O PROPELLANT
P–5, P–7, P–10 (SYMETRICAL ENGINE PLACEMENT)	YES*	4.75*	0.013 IN.	0.017 IN.
	NO		0.424 IN.	1.270 IN.
P–12M, P–13 (UNSYMETRICAL ENGINE PLACEMENT)	YES*	9.87*	0.013 IN.	0.018 IN.
	NO		1.42 IN.	4.33 IN.

* NON SYMETRICAL COMPONENT PLACEMENT COUNTERWEIGHT ASSUMED OFFSET BY SPACECRAFT EXPERIMENT PACKAGES.

APPENDIX A

RATE, RESOLUTION AND ENGINE MODULATION

RATE, RESOLUTION AND ENGINE MODULATION

Table A-1 summarizes the equations which can be used to determine the approximate values of rate and resolution for maneuvering a spin stabilized spacecraft. These equations are not exact since they do not account for engine warm-up which may significantly lower maneuver rate for short engine firings. However, for engine firings with on times in excess of 2 to 3 seconds, they approximate the maneuver rate within 10%. If more accurate results are desired the mission computer program used during this contract computes the actual maneuver rate for any given maneuver by dividing the maneuver size by the actual maneuver time required for the engine to deliver the necessary total impulse.

The most complex engine modulation for a spin stabilized spacecraft occurs when multiple engines are fired to perform a velocity change maneuver while simultaneously maintaining a constant vehicle spin rate and attitude.

The basic constraints listed below apply to the following discussion of this type of maneuver.

- The velocity change maneuver is performed with engines firing perpendicular to the spin axis.
- Engines are fired in a pulse mode over a short portion of the vehicle revolution when their thrust vector lines up approximately with the desired velocity change.
- Once the number and location of engines to be considered for the maneuver have been decided upon, it is desirable to complete the maneuver using minimum propellant and in a minimum time period.
- The center of gravity of the spacecraft is a known function of the mass of propellant consumed.
- The resultant thrust level of each engine is a known function of pulse width pulse number and mass of propellant consumed (Note: the resultant thrust level is the impulse delivered in the desired direction divided by the engine pulse width).

If it is assumed that all the engines to be used for the maneuver are fired with a fixed pulse width once per revolution, it can be seen that variations in c.g. location, engine moment arms and engine resultant thrust can cause distrubances in vehicle attitude and spin rate. These disturbances can be eliminated by pre-programmed modulation of the engines. There are at least three obvious methods of modulation:

(Continued)

1. Vary pulse widths on some engines.
2. Skip firing during some spacecraft revolutions on some engines.
3. Combinations of 1 and 2 above.

Regardless of the method used the following analysis applies. Assume it has been decided to consider "K" engines for performing the maneuver. (Note: The word consider is used since it is possible that the analysis will determine that "K-1" or "K-2", etc., engines are more desirable within the constraints defined above). It is assumed that at least one of the selected engines will be fired at the optimum pulse width (from a propellant consumption standpoint) once per revolution. This engine then becomes a reference and has a modulation constant (β) equal to one. In general, the modulation constant for other engines are given by:

$$\beta_i = \left(\frac{t_{on_i}}{t_{on_{opt}}}\right)\left(1 - \frac{1}{R}\right)$$

β_i = modulation constant of the th engine (none)

t_{on_i} = pulse width of the i^{th} engine (seconds)

$t_{on_{opt}}$ = pulse width of the reference engine fired at optimum on time (seconds)

R = the number of revolutions between skipped pulses

The following analysis allows computation of the values of β_i for each engine when the vehicle c.g., engine moment arms and engine resultant thrust level for a optimum pulse width are known. If any of these values change as a known function of propellant consumption then the values of β_i can be updated and determined as a function of time during the maneuver using this analysis.

Step 1

Referring to Table A-1, calculate the values of the coefficients of β_i in the following two series. Each term represents an effective moment of the k^{th} engine about the c.g. The terms in the first equation cause attitude errors. The terms in the second equation cause spin errors. Be consistent when selecting signs for each moment.

(Continued)

$$\sum_{k=1}^{K} r_{\alpha k} \quad F_k \quad \beta_k \quad = M_\alpha$$

$$\sum_{k=1}^{K} r_{Nk} \quad F_k \quad \beta_k \quad = M_N$$

K	=	number of engines considered for firing
k	=	subscript which number engines and then associated moment arms and modulation constants
F_k	=	magnitude of resultant thrust of k^{th} engine firing optimum pulse width (pounds)
$r_{\alpha k}$	=	moment arm of the k^{th} engine as defined in Table A-1 (ft)
r_{Nk}	=	moment arm of the k^{th} engine as defined in Table A-1 (ft)
M_α	=	net attitude moment error (ft-lb)
M_N	=	net spin moment error (ft-lb)

Step 2

From each equation select the term which has the largest coefficient. If these two terms have different subscripts for β then these β's are the trial modulation constants. If these two terms have the same subscript for β then the term with the next smallest coefficient in the spin moment equation is determined. The three terms determined above have two β's associated with them and they become the trial modulation constants.

Step 3

Set the attitude and spin moment equations equal to zero. Set all values of β except the trial modulation β's and any previously determined zero β's equal to one. Solve the two equations for the trial values of β.

(Continued)

If both trial values of β are positive the analysis is complete. If either or both are negative, the negative β's should be set equal to zero and the analysis repeated starting with Setp one.

Step 4

To determine overall maneuver modulation constant average the non-zero modulation constants. To determine the effective number of firing engines sum the non-zero modulation constants.

The overall maneuver modulation constant can be used in approximating the maneuver rate by substituting its value in the appropriate equation in Table A-1. The effective number of firing engines can be used in the propellant consumption computer program as a means for estimating the effects of modulation on propellant consumption.

TABLE A-1

RATE AND RESOLUTION EQUATIONS FOR MANEUVERING A SPIN STABILIZED SPACECRAFT

Maneuver	Rate (Units/Sec)	Resolution (Units)
(Velocity) "ΔV" (M/Sec)	$\frac{n \; F \; \eta \; \beta}{.1019 \; M} \left(\frac{N \; t_{on}}{60} \right)$	$\frac{n \; F \; \eta \; t_{on}}{.1019 \; M}$
(Spin) "ΔV" (RPM)	$\frac{n \; F \; r_s}{2.257 \times 10^{-5} \; I_{zz}}$	$\frac{n \; F \; r_s \; t_{on}}{2.257 \times 10^{-5} \; I_{zz}}$
(Attitude) "Δ α" (Degrees)	$\frac{n \; F \; \eta \; r_\alpha}{.3939 \times 10^{-6} \; N \; I_{zz}} \left(\frac{N \; t_{on}}{60} \right)$	$\frac{n \; F \; \eta \; r_\alpha \; t_{on}}{.3939 \times 10^{-6} \; N \; I_{zz}}$

Symbol	Units	Description
β	None	Modulation constant which is a function of C.G. location. If the sum of the force mements about the C.G. of all engines firing is zero it has a value of one.
η	None	Impulse effectiveness ratio of impulse delivered in the desired direction to the total impulse delivered by engines.
F	Lbs	Engine Thrust Level
I_{zz}	Lb In^2	Vehicle Interia about the Spin Axis
M	Lbs	Vehicle Weight
N	RPM	Vehicle Spin Rate
n	None	Number of Engines Firing
r_α	Ft	Moment Arm for Attitude Control Engines
r_s	Ft	Moment Arm for Spin Control Engines
t_{on}	Sec	Engine on Time

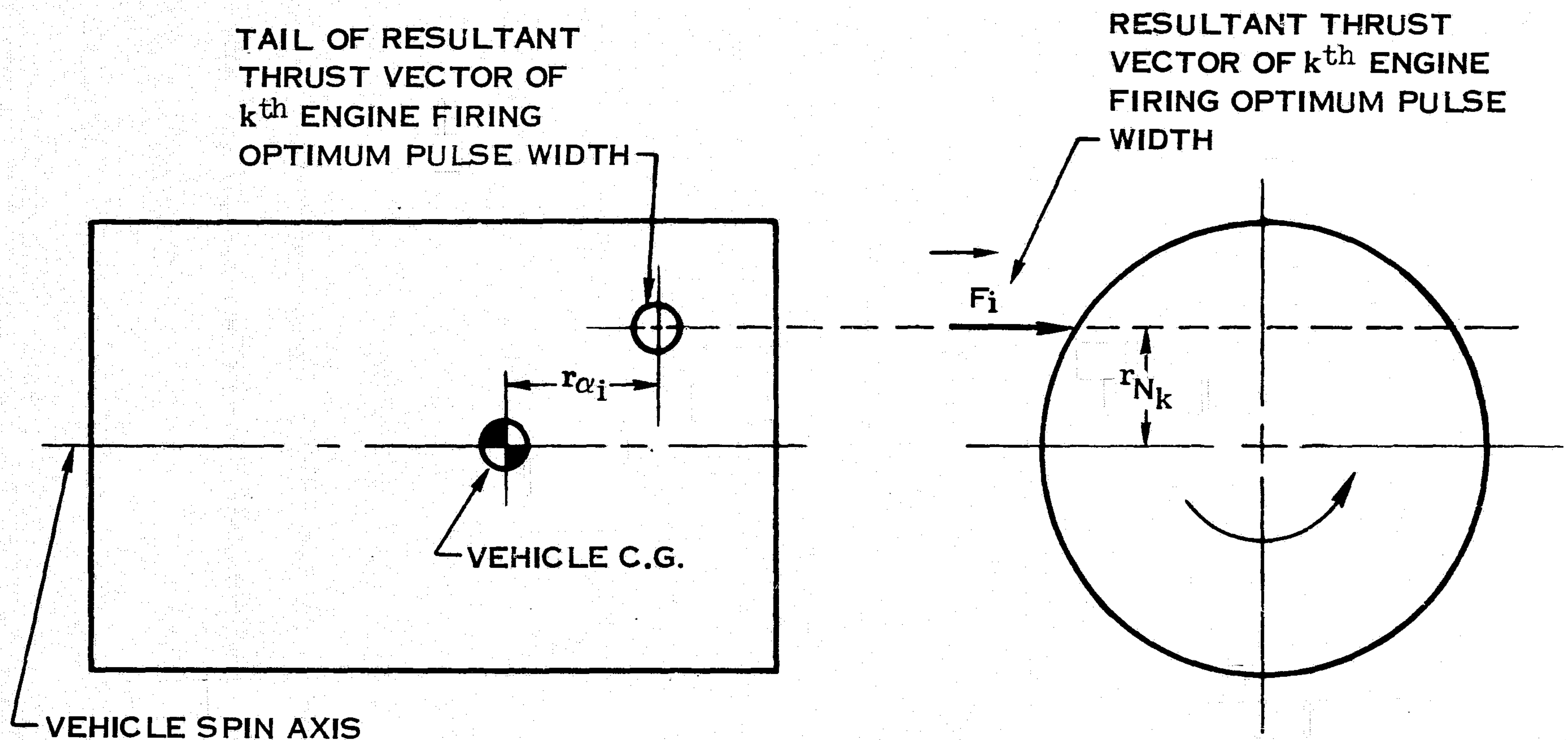
TAIL OF RESULTANT THRUST VECTOR OF k^{th} ENGINE FIRING OPTIMUM PULSE WIDTH
RESULTANT THRUST VECTOR OF k^{th} ENGINE FIRING OPTIMUM PULSE WIDTH
F_i
r_{α_i}
r_{N_k}
VEHICLE C.G.
VEHICLE SPIN AXIS

FIGURE A–1