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FINAL REPORT
SYSTEM DESIGN OF THE PIONEER VENUS SPACECRAFT

VOLUME 14
TEST PLANNING TRADES

By
C. D. PEDRETTI
ET AL.

July 1973

Prepared Under
Contract No. NAS[redacted]2-7250

By
HUGHES AIRCRAFT COMPANY
EL SEGUNDO, CALIFORNIA

For
AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION

CR-137501

[redacted]

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PREFACE

The Hughes Aircraft Company Pioneer Venus final report is based on study task reports prepared during performance of the "System Design Study of the Pioneer Spacecraft." These task reports were forwarded to Ames Research Center as they were completed during the nine months study phase. The significant results from these task reports, along with study results developed after task report publication dates, are reviewed in this final report to provide complete study documentation. Wherever appropriate, the task reports are cited by referencing a task number and Hughes report reference number. The task reports can be made available to the reader specifically interested in the details omitted in the final report for the sake of brevity.

This Pioneer Venus Study final report describes the following baseline configurations:

- "Thor/Delta Spacecraft Baseline" is the baseline presented at the midterm review on 26 February 1973.
- "Atlas/Centaur Spacecraft Baseline" is the baseline resulting from studies conducted since the midterm, but prior to receipt of the NASA execution phase RFP, and subsequent to decisions to launch both the multiprobe and orbiter missions in 1978 and use the Atlas/Centaur launch vehicle.
- "Atlas/Centaur Spacecraft Midterm Baseline" is the baseline presented at the 26 February 1973 review and is only used in the launch vehicle utilization trade study.

The use of the International System of Units (SI) followed by other units in parentheses implies that the principal measurements or calculations were made in units other than SI. The use of SI units alone implies that the principal measurements or calculations were made in SI units. All conversion factors were obtained or derived from NASA SP-7012 (1969).

The Hughes Aircraft Company final report consists of the following documents:

<u>Volume 1 - Executive Summary</u> - provides a summary of the major issues and decisions reached during the course of the study. A brief description of the Pioneer Venus Atlas/Centaur baseline spacecraft and probes is also presented.

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Volume 2 - Science - reviews science requirements, documents the science-peculiar trade studies and describes the Hughes approach for science implementation.

Volume 3 - Systems Analysis - documents the mission, systems, operations, ground systems, and reliability analysis conducted on the Thor/Delta baseline design.

Volume 4 - Probe Bus and Orbiter Spacecraft Vehicle Studies - presents the configuration, structure, thermal control and cabling studies for the probe bus and orbiter. Thor/Delta and Atlas/Centaur baseline descriptions are also presented.

Volume 5 - Probe Vehicle Studies - presents configuration, aerodynamic and structure studies for the large and small probes pressure vessel modules and deceleration modules. Pressure vessel module thermal control and science integration are discussed. Deceleration module heat shield, parachute and separation/despin are presented. Thor/Delta and Atlas/Centaur baseline descriptions are provided.

Volume 6 - Power Subsystem Studies

Volume 7 - Communication Subsystem Studies

Volume 8 - Command/Data Handling Subsystems Studies

Volume 9 - Altitude Control/Mechanisms Subsystem Studies

Volume 10 - Propulsion/Orbit Insertion Subsystem Studies

Volumes 6 through 10 - discuss the respective subsystems for the probe bus, probes, and orbiter. Each volume presents the subsystem requirements, trade and design studies, Thor/Delta baseline descriptions, and Atlas/Centaur baseline descriptions.

Volume 11 - Launch Vehicle Utilization - provides the comparison between the Pioneer Venus spacecraft system for the two launch vehicles, Thor/Delta and Atlas/Centaur. Cost analysis data is presented also.

Volume 12 - International Cooperation - documents Hughes suggested alternatives to implement a cooperative effort with ESRO for the orbiter mission. Recommendations were formulated prior to the deletion of international cooperation.

Volume 13 - Preliminary Development Plans - provides the development and program management plans.

Volume 14 - Test Planning Trades - documents studies conducted to determine the desirable testing approach for the Thor/Delta spacecraft system. Final Atlas/Centaur test plans are presented in Volume 13.

Volume 15 - Hughes IR&D Documentation - provides Hughes internal documents generated on independent research and development money which relates to some aspects of the Pioneer Venus program. These documents are referenced within the final report and are provided for ready access by the reader.

Data Book - presents the latest Atlas/Centaur Baseline design in an informal tabular and sketch format. The informal approach is used to provide the customer with the most current design with the final report.

CONTENTS

1. SUMMARY

This volume reviews Pioneer Venus system test plans and trade studies which were first published as Study Tasks (References 1 through 5). The plan and trade studies are presented here in a condensed form; greater detail may be found in the referenced study tasks if desired. All significant conclusions and plan outlines of the original studies are, however, presented in this volume.

In general, plans and trades that were applicable only to the Thor/Delta boost vehicle configuration have been omitted, and the contents of this volume are applicable to the Atlas/Centaur boost vehicle configuration. In Section 4, however, the environmental test considerations are presented as they were at the time of the original study task (January 1973). Vibration and acceleration levels are, therefore, Thor/Delta levels. Atlas/Centaur levels have been incorporated into the Pioneer Venus Environmental Test Specification.

2. INTRODUCTION

This volume consists of system test plans (Sections 3, 4 and 5) and trade studies (Sections 6 and 7) as outlined below.

Section 3. Assembly, Test, and Launch Operations Sequences. This section discusses the functional flow of the entire system test and launch operations sequence.

Section 4. Environmental Test Plans. This section presents an outline of the qualification and acceptance test programs to be performed on the units, probes and spacecraft. Actual environmental test levels are listed, and a detail discussion of solar-thermal-vacuum testing is included.

Section 5. Parachute Subsystem Development Test Plan. This section outlines the critical tests to be performed by the General Electric Company to verify the design of the large probe parachute subsystem.

Section 6. Hardware Cost Trades Study. This section summarizes a study that was made of the relative hardware costs of seven different approaches to qualifying, accepting, and launching the two spacecraft.

Section 7. Solar-Thermal-Vacuum/Thermal Vacuum Trades. This section presents the results of a study which examined past Hughes thermal test practices, Pioneer F/G practice, and cost to arrive at the Pioneer Venus solar-thermal-vacuum test plan.

3. ASSEMBLY, TEST, AND LAUNCH OPERATIONS SEQUENCES

Highlights of the Thor/Delta hardware test sequences are as follows:

1) Probe spacecraft development and qualification between go-ahead and 27 months after go-ahead

2) Probe spacecraft acceptance testing between months 18 and 33

3) Orbiter spacecraft development and qualification between months 13 and 44

4) Orbiter spacecraft acceptance between months 45 and 54

Original Thor/Delta flow diagrams were used as the basis for the development of the present system test and operations schedules. The current schedules are presented in the Integrated Test Plan in Volume 15 of the Final Report.

4. ENVIRONMENTAL TEST PLANS

4.1 GENERAL

This section outlines the baseline specification for the qualification and acceptance environmental test programs to be conducted on Pioneer Venus hardware.

The environmental levels of the baseline specifications reflect the test planning early in the program and formed the basis for the current test plan. The present environmental test plan is discussed in the Integrated Test Plan, Volume 15 of the Final Report. Actual environmental test levels presently planned are specified in the Environmental Test Specification.

Environmental test levels are based on the most severe exposure that a given unit or system is required to survive. The test plans for the probe units, pressure vessels, and all up probes are generally more extensive than for the bus units and spacecraft. This is because operation of the probes is required after separation from the bus during the approach to Venus.

In the environmental test level tables of Section 4.2, unit qualification and acceptance tests apply to both probe and bus units. The spacecraft qualification and acceptance tests apply to the full up spacecraft consisting of both bus and probes. The qualification and acceptance tests apply only to the probes and represent conditions encountered by the probes after separation from the bus. Table 4-1 summarizes the environmental tests applicable to the various units and systems.

Before, during, and after any of the environmental tests specified, the test article shall be subjected to comprehensive operational verification tests and records made of all data necessary to determine the proper performance of the test article.

Qualification tests are intended to simulate conditions more severe than those anticipated during ground, launch, and space operations. Satisfactory demonstration of the ability of the assembly to withstand the test environments specified qualifies the assembly, although final acceptance of the design is subject to satisfactory performance of the assembly throughout the complete test program.

TABLE 4-1. ENVIRONMENTAL TEST APPLICABILITY

Environmental Test Type	Test Applicability Qualification (Q) and Acceptance (A)				
	Bus Units	Probe Units	Probe Pressure Vessel*	Probes	Spacecraft (including probe)
Vibration	QA	QA		QA	QA
Acceleration	Q	Q			
Shock	Q	Q			
Spin	Q	Q			Q
Magnetic	QA	QA		A	A
Electromagnetic compatibility					QA
Solar-thermal-vacuum (cruise)					QA
Thermal-vacuum	QA				
Thermal		QA			
Solar-thermal-vacuum (separated probe)				QA	
Pressure temperature			QA		
Deceleration (high g)		QA	QA		

* Probe pressure vessel is the complete probe without deceleration module.

Acceptance tests simulate conditions expected during ground handling, launch, and transit to Venus. Satisfactory demonstration of the assembly to withstand the tests specified herein certifies the units as suitable for integration onto the flight spacecraft and certifies the spacecraft as flightworthy.

4.2 ENVIRONMENTAL TEST LEVELS

Tables 4-2 through 4-7 outline the environmental test levels for qualification and acceptance testing of units, spacecraft, and probes.

The vibration and acceleration levels specified are for the Thor/Delta launch vehicle. All other environments specified are independent of launch vehicle.

TABLE 4-2. ENVIRONMENTAL TEST LEVELS - UNIT [1] QUALIFICATION

TEST	REQUIREMENTS	COMMENTS
Vibration	See below	See below

TEST: Vibration

REQUIREMENTS:

Each assembly shall be vibrated in each of three orthogonal directions with one direction being parallel to the thrust axis. Attachment of the assembly to the test apparatus shall be via a rigid fixture and shall simulate the actual attachment of the assembly to the spacecraft structure. The assembly shall be tested in the operational condition normal for the launch phase of the mission.

a) Sinusoidal Swept Frequency. This portion of the test shall be conducted by sweeping the applied frequency once through each range specified in the schedule. The rate of change of frequency shall be two octaves per minute.

		Bus Units and Probes Acceleration, g, 0-peak							Probe Units Acceleration, g, 0-peak	
	Frequency cps	Large Probe and Pressure Vessel	Small Probes and Pressure Vessel	Shelf Mounted Items	Fuel Tanks	Bicone/ Omni Attachment to Thrust Tube	Bicone/ * Omni Antenna	Frequency cps	Large Probe Shelf Mounted Items (5)	Small Probe Shelf Mounted Items (5)
Thrust Axis	5-15	2.3	2.3	2.9	2.3	2.3	2.3	5-15	2.3	2.3
	15-21	8.0	8.0	10.0	8.0	8.0	8.0	15-21	8.0	8.0
	21-55	4.0	5.0	5.0	4.0	4.0	4.0	21-55	4.0	5.0
	55-85	4.0	25.0	40.0	15.0	6.0	7.5	55-80	4.0	25.0
	85-150	6.0	4.5	10.0	5.5	10.0	11.5	80-150	20.0	25.0
	150-400	4.5	4.5	4.5	4.5	4.5	4.5	150-400	4.5	4.5
	400-2000	7.5	7.5	7.5	7.5	7.5	7.5	400-2000	7.5	7.5
Lateral Axes	5-13	2.5	2.5	2.5	2.5	2.5	2.5	5-13	2.5	2.5
	13-25	5.5	5.5	5.0	4.0	5.5	8.0	13-25	5.5	5.5
	25-55	1.8	1.5	1.5	1.7	1.6	2.7	25-55	1.8	1.5
	55-150	4.0	10.5	8.0	20.0	10.5	17.0	55-80	4.0	10.5
	150-400	4.5	4.5	4.5	4.5	4.5	4.5	80-150	20.0	35.0
	400-2000	7.5	7.5	7.5	7.5	7.5	7.5	150-400	4.5	4.5
								400-2000	7.5	7.5

* Orbiter high gain antenna - TBS
Orbit Insertion Motor - TBS

b) Random Motion Vibration. Gaussian random vibration shall be applied with the "g" peaks clipped at three times the root mean square acceleration specified in the schedule. With the assembly installed, the control accelerometer response shall be equalized such that the specified power spectral density (PSD) values are within ±3 dB throughout the frequency band. Test duration shall be 2 min. per axis.

COMMENTS:

1) Acceleration levels are based on response to notched spacecraft input vibration levels.

2) Thrust axis input notching utilizes a 2900 kg (6400 lb) maximum axial load in the booster attach fitting, based on an 0.83 g maximum input to the spacecraft. The notching is effective in the 63-67 cps frequency range and results from an agreement with NASA concerning Telesat spacecraft design utilizing the 2914 Delta launch vehicle SECO loads.

3) Lateral axis input notching utilizes a (165,000 cm-kg (141,000 in-lb) maximum equivalent bending moment at the spacecraft-booster interface plane based on combined spacecraft ultimate loads of 4.35 g axial and 3.45 g lateral. The notching is effective in the 14-25 cps frequency range.

4) Inputs for shelf-mounted items are at the shelf attach points.

5) Fundamental frequencies of large and small probe shelf-mounted items, including the shelves, shall be greater than 80 cps.

(1) Both bus units and probe units.

TABLE 4-2. ENVIRONMENTAL TEST LEVELS - UNIT QUALIFICATION (continued)

<table>
<tr><th>TEST</th><th>REQUIREMENTS</th><th>COMMENTS</th></tr>
<tr><td>Vibration (continued)</td><td>
<table>
<tr><th colspan="4">UNIT PRELIMINARY QUALIFICATION RANDOM VIBRATION TEST LEVELS</th></tr>
<tr><th>Vibration Axis</th><th>Frequency, Hz</th><th>PSD Level, g^2/Hz</th><th>g_{RMS}</th></tr>
<tr><td>Thrust</td><td>20 - 60
60 - 150
150 - 260
260 - 2000</td><td>13 dB/Oct
1.0
-13 dB/Oct
0.1</td><td>17.8</td></tr>
<tr><td>Lateral</td><td>20 - 150
150 - 300
300 - 2000</td><td>0.20
-3 dB/Oct
0.1</td><td>14.7</td></tr>
</table>
</td><td></td></tr>
<tr><td>Acceleration and Spin</td><td>The assemblies shall be attached to a mounting fixture simulating attachment to the spacecraft structure and attached to the acceleration equipment such that the acceleration is in the direction of the resultant of the thrust and spin accelerations applied to the assemblies. The accelerations are:

a) <u>Thrust.</u> The maximum acceleration(21 g) expected in powered flight.

b) <u>Spin.</u> As obtained at a spin rate of TBD rpm.

The acceleration levels shall be maintained for at least 3 minutes for each assembly. The assembly shall be in the operational configuration for powered flight.</td><td></td></tr>
<tr><td>Thermal-Vacuum</td><td>Thermal-vacuum tests shall be performed with the assembly mounted in a manner thermally simulating the attachment of the assembly to the spacecraft structure. Tests shall be conducted at 10°C (18°F) above the maximum and at 10°C (18°F) below the minimum predicted base plate temperatures (as determined from the thermal model tests or analysis). The assembly shall be sufficiently instrumented to insure measurement of realistic assembly maximum and minimum temperatures.

a) <u>Pump Down.</u> During thermal vacuum chamber pump down, the assembly shall be operated in the condition typical of the launch phase with continuous monitoring for corona effects.

b) <u>Orbital Operation</u>. Tests shall be conducted under stabilized temperature and pressure conditions with the pressure no greater than 1×10^{-5} Torr for a minimum exposure period of 12 h at each of the high and low design temperatures conditions with the assembly operating. For cyclically-operated assemblies, cold turn on capabilities shall be demonstrated at least three times during each 12 h temperature exposure. Performance of the assembly shall be verified throughout and after the exposure period.</td><td>Performed on all units except probe units.

Test sequences should use flight simulative temperature slopes. These will be specified in the Space Simulation section of the Integrated Test Plan (Study Task PL-4).</td></tr>
</table>

TABLE 4-2. ENVIRONMENTAL TEST LEVELS - UNIT QUALIFICATION (Continued)

TEST	REQUIREMENTS	COMMENTS
Thermal	Thermal tests shall be performed on probe units with the assembly mounted in a manner thermally simulating the attachment of the assembly to the spacecraft structure. Tests shall be conducted at 10°C (18°F) above the maximum and at (10°C (18°F) below the minimum predicted base plate temperatures (as determined from the thermal mode tests or analyses). The assembly shall be sufficiently instrumented to ensure measurement of realistic assembly maximum and minimum temperatures. Tests shall be conducted under stabilized temperature conditions for a minimum exposure period of 12 hours at each of the high and low design temperature conditions with the assembly operating. For cyclically-operated assemblies, cold turn on capabilities shall be demonstrated at least three times during each 12-hour temperature exposure. Performance of the assembly shall be verified throughout and after the exposure period.	Performed only on probe units.
Shock	Qualification shock test environment requirements are satisfied by the 100-2000 cps levels specified for unit qualification vibration.	
Magnetic Properties	All assemblies shall be tested for magnetic properties in accordance with subparagraphs (a) through (d) below. Test techniques shall be chosen that will provide information required to calculate the combined magnetic field of all assemblies on the spacecraft at the location of the flight magnetometer sensor. a) Measurement of the radial component of the magnetic field of the item along three orthogonal axes and at a range where the inverse cube law applies. b) The measurements in (a) above shall be performed after the item has been depermed in a field of 50 gauss or more and again after exposure to a momentary field of 25 gauss in each axis. c) Measurements of the radial component at the magnetic field of the item along three orthogonal axis and at a distance where the inverse cube law applies while the item is being operated in its normal flight operating mode. d) Additional tests on any hardware or assemblies that are in close proximity to the flight magnetometer sensors. These tests will be made at close measuring distances 3 to 5 cm (1 to 2 inches) after the items have been magnetized in a 50 gauss field.	

TABLE 4-3. ENVIRONMENTAL TEST LEVELS - UNIT[1] ACCEPTANCE

TEST	REQUIREMENTS	COMMENTS
Vibration	Flight assemblies shall be subjected to vibration tests in the manner described under qualification, except that the sinusoidal sweep rate shall be four octaves per minute and the vibration levels shall be as follows:	

ASSEMBLY ACCEPTANCE SINUSOIDAL VIBRATION SCHEDULE

Vibration Axis	Frequency, Hz	Acceleration, g's (0 - peak)
Thrust	5-2000	Levels for various units are 2/3 of the levels shown for qualification sinusoidal vibration.
Both Lateral	5-2000	

ASSEMBLY ACCEPTANCE RANDOM MOTION VIBRATION SCHEDULE

Vibration Axis	Test Duration, Min. each Axis	Frequency, Hz	Acceleration Levels
All Three Axes	1.0	20-2000	g^2/Hz levels will be 4/9 of unit qualification levels. g_{rms} levels will be 2/3 of unit qualification levels.

[1] Both probe units and bus units

TABLE 4-3. ENVIRONMENTAL TEST LEVELS - UNIT ACCEPTANCE (Continued)

TEST	REQUIREMENTS	COMMENTS
Thermal Vacuum	Thermal vacuum tests shall be performed on each assembly with the assembly mounted in a manner thermally simulating the attachment of the assembly to the spacecraft structure. Tests shall be conducted with the assembly operating in its flight mode at the maximum and minimum predicted assembly temperatures (as determined from the thermal model tests or analysis).	The assembly shall be sufficiently instrumented to insure the measurement of realistic assembly maximum and minimum temperatures. Performance of each assembly shall be verified throughout and after the exposure period.
	a) Pump Down. During thermal vacuum chamber pump down the assembly shall be operated in the condition typical of the launch phase with continuous monitoring for corona effects.	
	b) Orbital Operation. Tests shall be conducted under stabilized temperature and pressure conditions with the pressure no greater than 1×10^{-5} Torr for a minimum exposure of 12 hours at each of the design temperatures. For cyclically-operated assemblies, cold turn-on capabilities shall be demonstrated at least two times during each 12-hour temperature exposure. For assemblies that will not be tested as a part of the system thermal vacuum tests (such assemblies may be solar array panels, appendages and spare assemblies), the minimum exposure period shall be 120 hours at the maximum temperature and 96 hours at the minimum temperature.	Not performed on probe units.
Thermal	Thermal tests shall be performed on each probe assembly with the assembly mounted in a manner thermally simulating the attachment of the assembly to the spacecraft structure. Tests shall be conducted with the assembly operating in its flight mode at the maximum and minimum predicted assembly temperatures (as determined from the thermal model tests or analysis). Tests shall be conducted under stabilized temperature conditions for a minimum exposure of 12 hours at each of the design temperatures. For cyclically-operated assemblies, cold turn-on capabilities shall be demonstrated at least two times during each 12-hour temperature exposure.	The assembly shall be sufficiently instrumented to insure the measurement of realistic assembly maximum and minimum temperatures. Performance of each assembly shall be verified throughout and after the exposure period. Performed on probe units.

TABLE 4-3. ENVIRONMENTAL TEST LEVELS - UNIT ACCEPTANCE (Continued)

TEST	REQUIREMENTS	COMMENTS
Magnetic Properties	All assemblies shall be tested for magnetic properties in accordance with sub-paragraphs (a) through (d) below. Test techniques shall be chosen that will provide information required to calculate the combined magnetic field of all assemblies on the spacecraft at the location of the flight magnetometer sensor. a) Measurement of the radial component of the magnetic field of the item along three orthogonal axes and at a range where the inverse cube law applies. b) The measurements in (a) above shall be performed after the item has been depermed in a field of 50 gauss or more and again after exposure to a momentary field of 25 gauss in each axis. c) Measurements of the radial component of the magnetic field of the item along three orthogonal axis and at a distance where the inverse cube law applies while the item is being operated in its normal flight operating modes. d) Additional tests on any hardware or assemblies that are in close proximity to the flight magnetometer sensors. These tests will be made at close measuring distances 3 to 5 cm (1 to 2 inches) after the items have been magnetized in a 50 gauss field.	

TABLE 4-4. ENVIRONMENTAL TEST LEVELS - SPACECRAFT[1] QUALIFICATION

TEST	REQUIREMENTS	COMMENTS
Spin	The prototype spacecraft shall be subjected to a spin test to demonstrate satisfactory system performance in both the launch and orbital configurations. Visual inspections before and after spin exposure shall be conducted to assess any mechanical damage. The spacecraft in its launch configuration and launch operational mode shall be subjected to a spin rate of TBD rpm.	The duration shall be sufficient to obtain a satisfactory record of spacecraft operation.
Vibration	The prototype spacecraft shall be vibrated in each of three orthogonal directions with one direction being parallel to the thrust axis. The spacecraft shall be attached to the vibrator table via a flight quality interstage structure. Attachment of the spacecraft to the interstage structure shall simulate the attachment for the launch configuration. Vibration excitation shall be applied at the base of the interstage structure where it interfaces with the vibrator table. The control accelerometer for these tests shall be positioned on the interstage structure as close as practical to the interfacing plane between the interstage structure and the vibrator table. The spacecraft shall be tested in the operational condition normal for the launch phase of the mission. a) Sinusoidal Vibration. This portion of the test shall be conducted by sweeping the applied frequency once through each range specified in the schedule below. The rate of change of frequency shall be two octaves per minute.	Any indication of out-of-tolerance performance, degradation, or malfunction of the spacecraft during any phase of the vibration testing shall be recorded and immediately reviewed in cooperation with NASA/ARC Senior Test Representative. Specified performance of all spacecraft equipment shall be verified before and after vibration testing. 1. Thrust axis input will be notched utilizing a 2900 kg (6400 lb) maximum axial load in the booster attach fitting, based on a 0.83 g maximum input to the spacecraft. The notching is effective in the 63-67 cps frequency range and results from an agreement with NASA concerning Telesat spacecraft design utilizing the 2914 Delta launch vehicle SECO loads. 2. Lateral axis input will be notched utilizing a 165,000 cm-kg (141,000 in-lb) maximum equivalent bending moment at the spacecraft-booster interface plane based on combined spacecraft ultimate loads of 4.35 g axial and 3.45 g lateral. The notching is effective in the 14-25 cps frequency range.

Sinusoidal Vibration Test Levels (Unnotched)[1,2]

Qualification - 2 octaves per minute

Thrust Axis		Lateral Axis	
Frequency, Hz	Acceleration, g's (0-P)	Frequency, Hz	Acceleration, g's (0-P)
5-15	2.3	5-14	2.3
5-21	6.8	14-250	1.5
21-250	2.3	250-400	4.5
250-400	4.5	400-2000	7.5
400-2000	7.5		

[1] Spacecraft shall be tested with probes installed.

TABLE 4-4. ENVIRONMENTAL TEST LEVELS - SPACECRAFT QUALIFICATION (Continued)

<table>
<tr><th>TEST</th><th>REQUIREMENTS</th><th>COMMENTS</th></tr>
<tr><td>Vibration (Continued)</td><td>b) Random Motion Vibration. Gaussian random vibration shall be applied with "g" peaks clipped at three times the root mean square acceleration, according to the schedule below. With the spacecraft installed, the control accelerometer response shall be equalized such that the specified power spectral density values are within ± 3 dB everywhere in the frequency band. Test duration shall be 2 min/axis.

Thrust and Lateral Axes

<table>
<tr><th>Frequency Hz</th><th>PSD g^2/Hz</th><th>g_{RMS} Approximate</th></tr>
<tr><td>20-300</td><td>+3 dB/OCT</td><td rowspan="2">9.2</td></tr>
<tr><td>300-2000</td><td>.045</td></tr>
</table>
</td><td></td></tr>
<tr><td>Solar-Thermal-Vacuum</td><td>Thermal-vacuum tests shall be performed with the prototype spacecraft to verify the thermal design and, within the performance capabilities of the thermal-vacuum chamber, to demonstrate performance capability beyond the environmental extremes to be encountered in flight. The spacecraft shall be installed in a vacuum test chamber that has cold wall and solar simulation capability and in such a manner as to simulate space conditions as accurately as possible. Electrical interference shall be minimized to insure accurate transmission of spacecraft performance data. The thermal-vacuum test shall include the following requirements:

a) The spacecraft shall be operated in the modes normal to launch during the chamber evacuation and spacecraft data shall be monitored continuously for detection of corona effects during this period.</td><td>The test facility, test configuration, and test instrumentation shall be subject to approval by the ARC/Project Office.

Failure of any subsystem during the thermal vacuum test that would result in a catastrophic mission failure shall require abortion of the thermal vacuum test and a retest shall be required. Failure of a redundant assembly of a particular subsystem shall not necessarily be sufficient cause for aborting the thermal vacuum test provided the basic mission requirements can be met throughout the entire 17-day thermal vacuum test period. Ultimate authority for abortion of the thermal vacuum test and duration of retest after correction of the malfunction shall be ARC.</td></tr>
</table>

TABLE 4-4. ENVIRONMENTAL TEST LEVELS - SPACECRAFT QUALIFICATION (Continued)

TEST	REQUIREMENTS	COMMENTS
Solar Thermal Vacuum (continued)	b) Simulated orbital flight tests shall be conducted under stabilized pressure conditions at pressures no greater than 1×10^{-5} Torr with a simulated solar illumination of the spacecraft and with the chamber shrouds cooled to -185 ±8°C (-305 ±15°F). At least three intensities of solar illumination shall be applied; that experienced by a spacecraft at 1.0SC, 1.9SC, and at least one intermediate value representative of a reasonable level of illumination between the two extremes. Approximately equal test time shall be allotted for each intensity of solar illumination. The spacecraft shall be operated in all modes of normal operation that are controlled by ground command. The configuration of the spacecraft for the tests shall be its space flight configuration as nearly as practical and external instrumentation to the spacecraft minimized. Ground power supplied to the spacecraft shall simulate that expected from the spacecraft power subsystem throughout the mission. The test duration shall be at least 17 continuous days.	Solar constants, i.e., 1.0 SC and 1.9 SCTBS in Integrated Test Plan (Study Task PL-4).
Electromagnetic Compatibility	Electromagnetic compatibility test of the prototype spacecraft, including scientific instruments, shall be performed to verify that all subsystems are capable of compatible operation without degrading the performance of one subsystem due to the operation of any or all other subsystems. The test shall include a measurement of the spacecraft-generated interference. The test shall also include audio-conducted susceptibility tests and rf-radiated susceptibility tests.	

TABLE 4-5. ENVIRONMENTAL TEST LEVELS - SPACECRAFT[1] ACCEPTANCE

<table>
<tr><th>TEST</th><th>REQUIREMENTS</th><th>COMMENTS</th></tr>
<tr><td>Vibration</td><td>The flight spacecraft shall be subjected to vibration tests in the manner described under qualification except that the sinusoidal sweep rate shall be four octaves per minute and the vibration levels shall be as follows:

<u>Sinusoidal Vibration Test Levels (Unnotched)</u>
Flight Acceptance - 4 octaves per minute

<table>
<tr><th colspan="2">Thrust Axis</th><th colspan="2">Lateral Axis</th></tr>
<tr><th>Frequency, Hz</th><th>Acceleration, g's (0-P)</th><th>Frequency, Hz</th><th>Acceleration, g's (0-P)</th></tr>
<tr><td>5-15</td><td>1.5</td><td>5-14</td><td>1.5</td></tr>
<tr><td>15-21</td><td>4.5</td><td>14-250</td><td>1.0</td></tr>
<tr><td>21-250</td><td>1.5</td><td>250-400</td><td>3.0</td></tr>
<tr><td>250-400</td><td>3.0</td><td>400-2000</td><td>5.0</td></tr>
<tr><td>400-2000</td><td>5.0</td><td></td><td></td></tr>
</table>

<u>Random Vibration Test Levels</u>
Flight Acceptance - 1 minute per axis

<table>
<tr><th colspan="3">Thrust and Lateral Axes</th></tr>
<tr><th>Frequency, Hz</th><th>PSD, g^2/Hz</th><th>RMS, Approximate</th></tr>
<tr><td>20-300</td><td>+3 dB/OCT</td><td rowspan="2">6.1</td></tr>
<tr><td>300-2000</td><td>.02</td></tr>
</table>
</td><td>Vibration levels will be 2/3 of the notched spacecraft qualification levels.</td></tr>
<tr><td>Solar-Thermal-Vacuum</td><td>Each flight spacecraft shall be subjected to thermal vacuum tests in accordance with the qualification section except that the intensity of the solar illumination shall be that experienced by a spacecraft at 1.0 SC, 1.9 SC, and at least one intermediate value representative of a reasonable level of illumination between the two extremes. Approximately equal test time shall be allotted for each intensity of solar illumination. The total test duration shall be at least 14 continuous days.</td><td>Solar Constant, i.e., 1.0 SC and 1.9 SC TBS in Study Task PL-4, Integrated Test Plan.</td></tr>
</table>

[1] Spacecraft shall be tested with probes installed.

TABLE 4-5. ENVIRONMENTAL TEST LEVELS - SPACECRAFT ACCEPTANCE (Continued)

TEST	REQUIREMENTS	COMMENTS
Magnetic Properties	Magnetic measurements of each flight spacecraft shall be made upon completion of the environmental acceptance test phase. Final mass properties determination, final balance, and final spacecraft performance tests after magnetic properties measurements are permissible. If possible, these measurements shall be made on the fully assembled spacecraft in its orbital configuration. Magnetic tests shall be performed with the spacecraft in a non-operative mode and in selected operating modes. These tests shall include measurements to determine the following: a) The change in remanence resulting from a momentary exposure of 25 gauss along each axis. b) The magnitude and direction of remanence after a 50 gauss demagnetization along each axis. c) The magnitude and direction of stray magnetic fields produced by the complete spacecraft in the commandable operating modes.	If suitable magnetic test facilities are not available for testing the fully-assembled spacecraft, separate magnetic measurements of appendages will be acceptable and their results shall be used in determining the magnetic field of a fully-assembled spacecraft. Also performed on probes.
Magnetic Field Measurements	Two types of magnetic field measurements shall be required as follows: a) Spacecraft Remanence. Tests shall be made to permit accurate determination of the dc magnetic fields of the non-operating spacecraft. This test shall include triaxial magnetometer measurements taken simultaneously at three (3) distances, 1.0R, 0.79R, and 0.63R, where R is the distance of the flight magnetometer sensor from the spacecraft spin axis. Measurements shall be taken during continuous rotation of the spacecraft about each of three orthogonal axes. A means shall be provided whereby recordings of the output of the facility magnetometer can be correlated to the instantaneous position of the spacecraft coordinate system. Data shall be recorded for at least five successive rotations through 360° about each axis. b) Spacecraft Stray Fields. Tests shall be made to determine the stray fields associated with the operating modes. Triaxial measurements shall be recorded along an axis normal to the spin axis that passes through the flight magnetometer sensor at the distances defined in (a) above.	Also performed on probes. Stray field tests may be performed in earth's ambient field.

TABLE 4-5. ENVIRONMENTAL TEST LEVELS - SPACECRAFT ACCEPTANCE (Continued)

TEST	REQUIREMENTS	COMMENTS
Magnetic Test Sequence	Magnetic tests shall be performed in the sequence shown. a) Spacecraft Stray Fields b) Spacecraft Remanence (as is) c) Magnetize spacecraft (3 axes) d) Spacecraft Remanence (permed) e) Demagnetize spacecraft f) Spacecraft Remanence (depermed)	Also applies to probes.
Facility Requirements	TBD	Also applies to probes.
Scientific Instrument Fields	In the event that the magnetic field contribution of the scientific instrument payload is not sufficiently defined at the magnetometer sensor location, it may be necessary to make a spacecraft remanence test without the instrument payload to assess the magnetic properties of the spacecraft alone.	ARC will define magnetic properties of the scientific instrument payload. Also applies to probes.
Electromagnetic Compatibility	Electromagnetic compatibility tests shall be conducted on each flight spacecraft in accordance with the requirements outlined under spacecraft qualification.	

TABLE 4-6. ENVIRONMENTAL TEST LEVELS - PROBE QUALIFICATION

TESTS	REQUIREMENTS (5)		COMMENTS
	Large Probe (1)	Small Probe (1)	
Pressure (2)	Simulated Venus descent tests shall be conducted with the pressure vessel module (PVM). The PVM shall be subjected to a coordinated pressure and temperature time profile within a CO_2 gas. PVM temperature will be stabilized at TBD °K for 8 hours. The ambient temperature and pressure will be initialized at 235°K and .081 atm. PVM subsystems will be turned on. Temperature and pressure will be increased to 5.2 atm and 430°K in 40 m and to 120 atm and 765°K in 79 m. PVM subsystem and science experiment operation will be monitored during the test.	Simulated Venus descent tests shall be conducted with the pressure vessel module (PVM). The PVM shall be subjected to a coordinated pressure and temperature time profile within a CO_2 gas. PVM temperature will be stabilized at TBD °K for 8 hours. The ambient temperature and pressure will be initialized at 235°K and .081 atm. PVM subsystems will be turned on. Temperature and pressure will be increased to 120 atm and 765°K in 60 m. PVM subsystem and science experiment operation will be monitored during the test.	
Temperature (3)	"	"	
Deceleration (4)	Critical units and the large probe systems shall be tested to 610 g's. Systems operation verification will be accomplished before and after the acceleration test.	Critical units and the small probe systems shall be tested to 660 g's. Systems operation verification will be accomplished before and after the acceleration test.	

NOTES: (1) Conditions TBS in HAC Specification, HS507-0300-7-1, "Spacecraft Design Environments"

(2) Venus Surface Pressure

(3) Venus Surface Temperature

(4) Entry

(5) These test levels represent conditions encountered by the probes after separation from the bus.

TABLE 4-6. ENVIRONMENTAL TEST LEVELS - PROBE QUALIFICATION (Continued)

TESTS	REQUIREMENTS		COMMENTS
	Large Probe (1)	Small Probe (1)	
Solar-Thermal-Vacuum (5)	Thermal vacuum tests shall be performed with the prototype probe to verify the thermal design and, within the performance capabilities of the thermal-vacuum chamber, to demonstrate performance capability beyond the environmental extremes to be encountered in flight. The probe shall be installed in a vacuum test chamber that has cold wall and solar simulation capability and in such a manner as to simulate space conditions typical of those to be encountered by the probe during the last 20 days of the mission. a) The probe shall be operated in the modes normal to separation, cruise and planetary encounter. b) Simulated near Venus flight tests shall be conducted under stabilized pressure conditions at pressures no greater than 1×10^{-5} Torr with simulated solar illumination of the spacecraft and with the chamber shrouds cooled to $-185 \pm \pm 8^\circ C$ ($-305 \pm 15^\circ F$). At least three intensities of solar illumination shall be applied; that experienced by a probe at E-20 days, E-1 day and at least one intermediate value representative of a reasonable level of illumination between the two extremes. Approximately equal test time shall be allotted for each intensity of solar illumination. The probe shall be operated in all modes of normal operation that are controlled by command. The configuration of the probe for the tests shall be its space flight configuration as nearly as practical and external instrumentation minimized. External power supplied to the probe shall simulate that expected from the battery power. The test duration shall be at least 3 continuous days.	Same as large probe unless noted.	The test facility, test configuration, and test instrumentation shall be subject to approval by the ARC/Project Office. Failure of any subsystem during the thermal-vacuum test that would result in a catastrophic mission failure shall require abortion of the thermal-vacuum test and a retest shall be required. Failure of a redundant assembly of a particular subsystem shall not necessarily be sufficient cause for aborting the thermal-vacuum test provided the basic mission requirements can be met throughout the entire probe thermal-vacuum test period. Ultimate authority for abortion of the thermal-vacuum test and duration of retest after correction of the malfunction shall be ARC. Solar constants to be used for this test will be specified in the Integrated Test Plan (Study Task PL-4).

NOTES: (1) Conditions TBS in HAC Specification, HS507-0300-7-1, "Spacecraft Design Environments"

(2) Venus Surface Pressure

(3) Venus Surface Temperature

(4) Entry

(5) Near Venus Conditions - after separation from bus

TABLE 4-7. ENVIRONMENTAL TEST LEVELS - PROBE ACCEPTANCE

TESTS	REQUIREMENTS[6]		COMMENTS
	Large Probe (1)	Small Probe (1)	
Pressure (2)	The pressure vessel module (PVM) shall be tested to the maximum Venus pressure, 96 atm.	Each small probe shall be tested to the maximum Venus pressure, 96 atm.	
Temperature (3)	The PVM shall be tested to simulate the Venus descent temperature profile. PVM shall be stabilized at TBD $^{\circ}$K for 4 hours. Ambient temperature will be initialized at 235°K. All PVM subsystems and science experiments shall be turned on. Ambient temperature shall be increased to 430°K in 40 m and to 765°K in 79 m. PVM subsystem and science operation will be monitored during the test.	The PVM shall be tested to simulate the Venus descent temperature profile. PVM shall be stabilized at TBD $^{\circ}$K for 4 hours. Ambient temperature will be initialized at 235°K. All PVM subsystems and science experiments shall be turned on. Ambient temperatures shall be increased to 765°K in 60 m. PVM subsystem and science operation will be monitored during the test.	
Deceleration (4)	Critical units and the large probe systems shall be tested to 490 g's. Systems operation verification will be accomplished before and after the acceleration test.	Critical units and each small probe system shall be tested to 530 g's. System operation verification will be accomplished before and after the acceleration test.	
Solar-Thermal-Vacuum (5)	Each flight probe shall be subjected to thermal vacuum tests in accordance with the qualification section except that the intensity of the solar illumination shall be that experienced at TBD SC, TBD SC, and at least one intermediate value representative of a reasonable level of illumination between the two extremes. Approximately equal test time shall be allotted for each intensity of solar illumination. The total test duration shall be at least two continuous days.	Same as large probe unless noted.	

NOTES: (1) Conditions TBS in HAC Specification, HS 507-0300-7-1, "Spacecraft Design Environments"

(2) Venus Surface Pressure

(3) Venus Surface Temperature

(4) Entry

(5) Near Venus Conditions - after separation from bus

(6) These test levels represent condition encountered by the probes after separation from the bus.

4.3 SOLAR-THERMAL-VACUUM TESTING

General

The intent of the solar-thermal-vacuum tests is to duplicate the sequence of events and, where possible, the real-time sequence encountered in an actual mission from prelaunch through entry, in the case of the probe bus, and from prelaunch through orbit, in the case of the orbiter.

Solar-Thermal-Vacuum Test Sequence

Prelaunch Functional Checkout

Prior to the actual start of the mission simulation for both the bus and orbiter spacecraft, a prelaunch functional checkout of the spacecraft is performed to verify spacecraft readiness.

Launch Through Transit

The simulated mission is initiated with launch eclipse and spinup, with the spin rate duplicating the actual mission.

Cruise Mode

At the beginning of cruise mode 1, which marks the start of the transit phase, spacecraft functional performance parameters are verified, the facility solar constant is adjusted to 1.0, and the spin rate and solar aspect angle are adjusted to simulate operational conditions. Cruise mode 1 for the probe bus solar-thermal-vacuum test is in real time; i.e., 5 days.

Midcourse Maneuvers

The first midcourse maneuver is performed in the third day after test start in the case of the orbiter solar-thermal-vacuum test, and at the end of 5 days of testing in the case of the probe bus.

There are three midcourse maneuvers for each spacecraft during the test, and in each case the solar aspect angle will be changed to simulate actual mission conditions. The second and third midcourse corrections occur at 7 and 10 days after test start in the case of the probe bus, and at 4 and 5 days in the case of the orbiter.

Transit Phase to Entry/Orbit Insertion

The test continues with midcourse cruise modes with the appropriate solar aspect angles and solar constants.

Since the objective of the qualification solar-thermal-vacuum test for both the probe bus and the orbiter spacecraft is to demonstrate the performance capability beyond the temperature range encountered in flight, both low and high temperature functional performance verification is planned for the qualification tests. It is planned to impose only high temperature testing for the acceptance tests.

Entry/Orbit Insertion

Entry is simulated for the probe bus just prior to the end of the 17th day of testing. All thermal data will have been obtained at this point in time, and the spacecraft will be transferred to battery power.

Earth occultation is simulated just prior to orbit insertion of the orbiter spacecraft, and the occultation experiments will be checked out during this period.

Orbit insertion simulation for the orbiter solar-thermal-vacuum test occurs 9 days after test start and marks the beginning of extensive functional testing during simulated eclipse and simulated orbit. These tests continue through the end of the 17 day test period.

Shutdown

At the conclusion of the 17 day solar-thermal-vacuum test, warmup is initiated and the spacecraft is removed from the chamber. At this point, a post-solar-thermal-vacuum functional test is performed on the spacecraft to insure no degradation occurred as a result of the environment. The instrumentation is then removed, and the test is terminated.

Instrumentation

The spacecraft is prepared for solar-thermal-vacuum tests by installing approximately 75 thermocouples which are connected to a computerized temperature data monitoring and reduction instrumentation center. These data supplement the flight sensor temperature data.

Following spacecraft instrumentation, the spacecraft and spin fixture are installed in the vacuum chamber. A slip ring assembly permits transfer of power and signals between the rotating spacecraft and the fixture. As the chamber is pumped from ambient laboratory pressure to 1×10^{-4} Torr, the spacecraft is operated in the launch/ascent mode on internal power. The main power bus is monitored for evidence of corona arcing during this test phase.

When chamber pressure of 1×10^{-5} Torr or less has been reached, the shrouds are filled with liquid nitrogen, spinning of the spacecraft is initiated, and the solar simulator is adjusted to a solar constant of 1.0.

Test access to the spacecraft is via rf link, umbilical connector, and special direct access. The test setup is diagrammed in Figure 4-1.

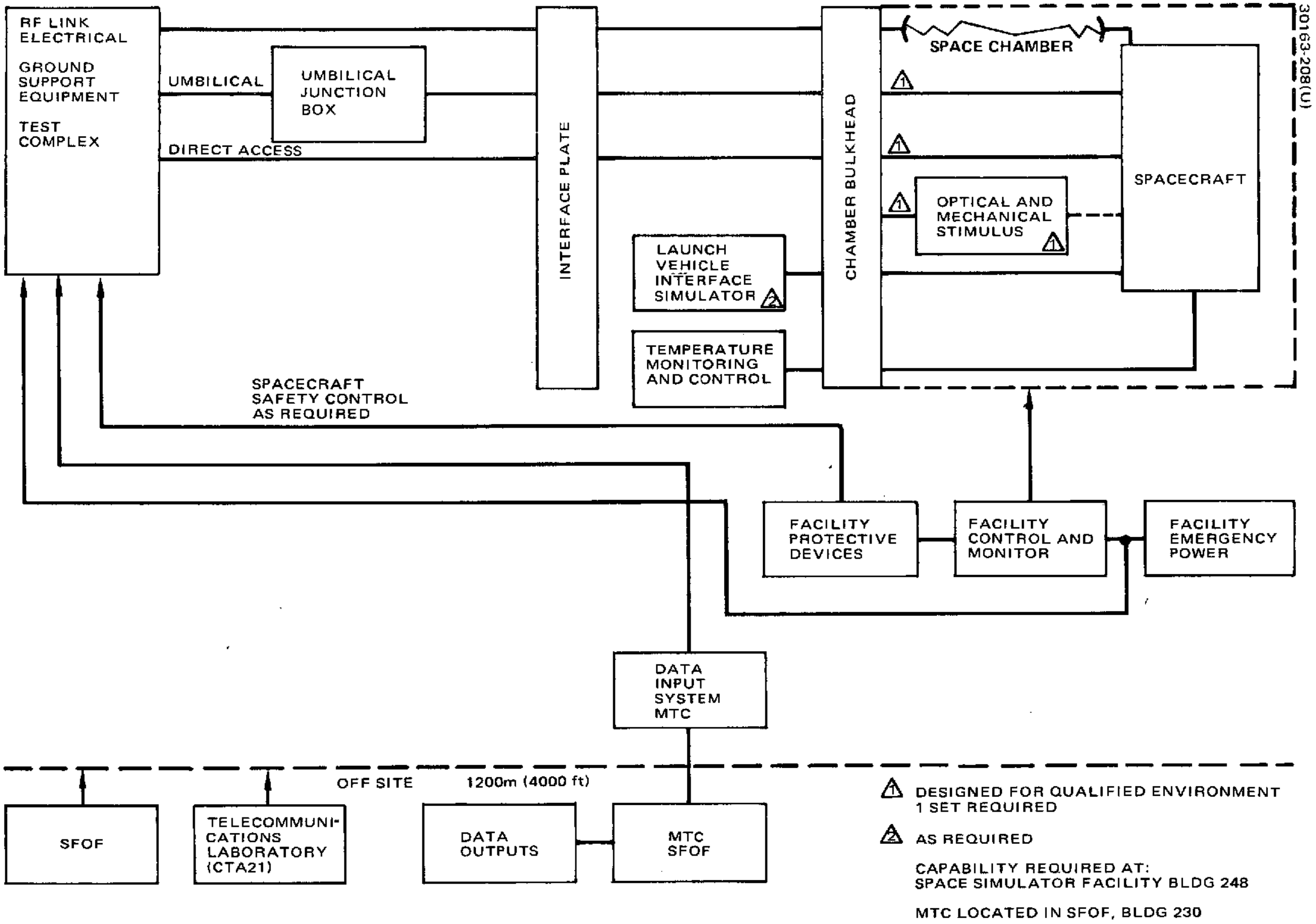


FIGURE 4-1. SOLAR THERMAL VACUUM TEST CONFIGURATION

5. PARACHUTE SUBSYSTEM DEVELOPMENT TEST PLAN

The development testing to demonstrate performance of the large probe parachute subsystem is described in this section. These tests will be performed by the General Electric Company on a subcontract from Hughes.

The development test program is summarized in the four tables included in this section. Table 5-1 shows deployment and pull tests of the pilot chute, main chute, and cords. Table 5-2 shows mortar development tests. Table 5-3 shows parachute proof tests wherein chutes are deployed at high aerodynamic pressure (Q) conditions. Table 5-4 shows the tests of the entire system in aircraft deployment using a boilerplate test probe.

TABLE 5-1. PARACHUTE VENDOR IN-PLANT TESTS

Test	Number Required	Equipment	Comment
A. Pilot Chute Deployment - Static B. Main Chute Deployment - Static C. Pull Tests (Critical Joints, etc.)	2 2 A/R	1 Pilot Chute/Bag Lines, etc. 1 Main Chute/Bag Lines, etc. Samples	Static pull tests to confirm break cord pull loads, orderly extraction of chute from canopy, line strengths, etc.

TABLE 5-2. PARACHUTE EJECTION MORTAR TESTS

Test	Number Required	Equipment	Comment
A. Dummy Load Ejection	6 min.	3 Dummy Pilot Chute Packs; 3 Ejection Mortars; 12 Ejection Charges; 1 Firing Test Fixture Instrumentation for Velocity Measurement Reaction Load Measurement	Live ejection of pilot chute from mortar to confirm ejection velocity, mortar structural integrity, and general firing characteristics.
B. Prototype Parachute Ejection (w/lines simulating stowed configuration)	4 min.	4 Prototype Pilot Chute Assemblies w/Connecting Lines to Main Chute, Covers, etc. 8 Ejection Charges 2 Ejection Mortars 1 Firing Fixture Instrumentation for Velocity Measurement, Load Measurement, High Speed Photo Coverage	Live ejection of pilot chute pack to demonstrate ejection velocity, line deployment, and general mechanics of the prototype ejection system.

TABLE 5-3. PARACHUTE PROOF TESTS

Test	Number Required	Equipment	Comment
A. Pilot Chute Load Tests	2 min.	2 Dummy Bombs (GFE) Fitted with Pilot Chute Pack Compartment 2 Prototype Pilot Chute Packs (Chute, Lines, Bag, etc.) - Onboard Load Measurement Instrumentation - Ground Tracking Stations for Trajectory and Deployment Data	Max Q deployment from aircraft with static line deployment.
B. Main Chute Load Tests	2 min.	2 Dummy Bombs (GFE) with Main Chute Pack Compartment. 2 Prototype Main Chute Packs (Chutes, Lines, Bag, etc.) - Onboard Load Measurement Instrumentation - Ground Tracking Station for Trajectory and Deployment Data	Max Q deployment from aircraft with static line deployment.
C. Chute Sequence Tests	2 min.	2 Dummy Bombs w/Chute Packing Structure 2 Prototype Parachute Systems (Pilot Chute, Main Chute, Ejection Mortar, Lines, Disconnects, etc.) 2 Onboard Programmers for Initiating Chute Deployment 2 Sets Batteries for Instrumentation and Programmer 2 Sets Onboard Load Measurements Instrumentation 2 Onboard Line Cameras - Ground Tracking Station for Trajectory and Deployment Data	Max/min Q deployment from aircraft with programmed ejection of the pilot chute followed by the complete chute sequence.

TABLE 5-4. SYSTEM TESTS (SUBSONIC DEPLOYMENT)

Test	Number Required	Equipment	Comments
A. Parachute System Tests	4 min.	4 Boilerplate Vehicles Simulating Shape, Covers, Attach Points, etc. 8 Onboard Gun Cameras 4 Onboard Programmers 4 Complete Retardation Systems a) Mortars (4) b) Parachutes (pilot and main) (4 each) c) Ejection Mortar Charges (8) d) Release Fittings e) Connecting Lines (4 sets) 4 Onboard Instrumentation Systems (TM or onboard recording of loads, events versus time) 1 Sway Brace Adapter Structure 2 Ground Handling Fixtures for Vehicle Assembly and Check-out 2 Electrical Checkout Consoles (event timing preflight checks) 8 Battery Systems (Onboard) for Programming, Firing Circuits and Camera Operations	Drop from high performance aircraft to check four (4) points on deployment envelope drop from B-58 or equivalent high performance aircraft to obtain near sonic velocity at approximately 40,000 feet altitude. Nominal system requirements are: Mortar: Dynamic pressure — $q = 185\ kg/m^2$ (38 psf) Mach No. — $m = 0.8$ Pilot: Dynamic pressure — $q = 185\ kg/m^2$ (38 psf)* Mach No. — $m = 0.8$ Main: Dynamic pressure — $q = 161\ kg/m^2$ (33 psf)* Mach No. — $m = 0.72$

It should be noted that the full system drop tests will demonstrate not only the parachute performance capability but, in addition, the full separation sequence, including mortar fire of the pilot, aft cover removal, and pressure vessel--forebody separation.

*NOTE: Proof tests will be performed at dynamic pressures 50 percent higher than these values.

6. HARDWARE COST TRADE STUDY

6.1 FLIGHT/QUALIFICATION HARDWARE

The baseline program plan used for initial cost estimates incorporated the following system level flight and qualification models:

2 flight models	1 multiprobe F-1 1 orbiter F-1
1 qualification model	1 multiprobe Y-1 to be refurbished and used as orbiter Y-1

This plan has the advantage of low risk because all flight hardware is new, not reworked qualification hardware. It also allows the option of using the orbiter Y-1 model for qualification and/or flight on the 1980 mission, and it provides for adequate spares for the flight models by utilizing component/subsystem qualification hardware.

Table 6-1 shows the hardware cost trades for seven different approaches to the two-mission test sequence. Of these seven, the first six include the usual separate qualification and acceptance test phases; the seventh (LA-LB) includes a combination qualification/acceptance (protoflight) test phase for each launch.

Of the seven test approaches, the baseline (L2-L3) is the most costly for hardware. The lowest cost approach of Table 6-1, which includes separate acceptance and qualification testing, is L2A-L4. The lowest cost option in Table 6-1 is the combined qualification/acceptance test philosophy of LA-LB. The sequence for this option is as follows:

1) Fabricate probe spacecraft (consisting of a bus, 1 large probe, and 3 small probes)

2) Qualification/acceptance (protoflight) test probe spacecraft

3) Launch multiprobe spacecraft (LA)

TABLE 6-1. PROTOFLIGHT COST TRADES HARDWARE COSTS

Options	Y-1 Fab	Y-1 R/F to F-1	Fab 2 Small Probes	F-1 Fab	Y-1 R/F to Y-1 Orbiter	Y-1 Orbiter Fab	Y-1 Orbiter R/F to F-1 Orbiter	F-1 Orbiter Fab	Total
L1 - L3 (L5 Option)	37.5	3.9	7.0	N/A	N/A	13.0	N/A	4.7	66.1
L1 - L4	37.5	3.9	7.0	N/A	N/A	13.0	1.4	N/A	62.8
L2 - L3 (L5 Option) (Baseline)	37.5	N/A	N/A	20.0	6.8	N/A	N/A	4.7	69.0
L2 - L4	37.5	N/A	N/A	20.0	6.8	N/A	1.4	N/A	65.7
L2 - L4 with R/F Probes	37.5	2.3 (LP, SP only)	7.0	5.2 (Bus only)	6.8	N/A	1.4	N/A	60.2
L2A - L3 (L5 Option)	37.5	2.3 (LP, SP only)	7.0	5.2 (Bus only)	6.8	N/A	N/A	4.7	63.5
LA - LB	44.5	N/A	N/A	N/A	N/A	13.0	N/A	N/A	57.5

L1 - Multiprobe Mission (Protoflight)
L2 - Multiprobe Mission (Fab Y-1 and F-1)
L2A - Multiprobe Mission (Fab Y-1 and F-1 - Fly Y-1 Probes)
L3 - Orbiter Mission (Fab F-1)
L4 - Orbiter Mission (Protoflight)
L5 - 1980 Mission Option (Using Refurbished Orbiter Y-1)
LA - Multiprobe Mission (Combined Qual/Acceptance Test)
LB - Orbiter Mission (Combined Qual/Acceptance Test)

4) Fab orbiter spacecraft

5) Qualification/acceptance (protoflight) test orbiter spacecraft

6) Launch orbiter spacecraft (LB)

These steps are diagrammed in Figure 6-1.

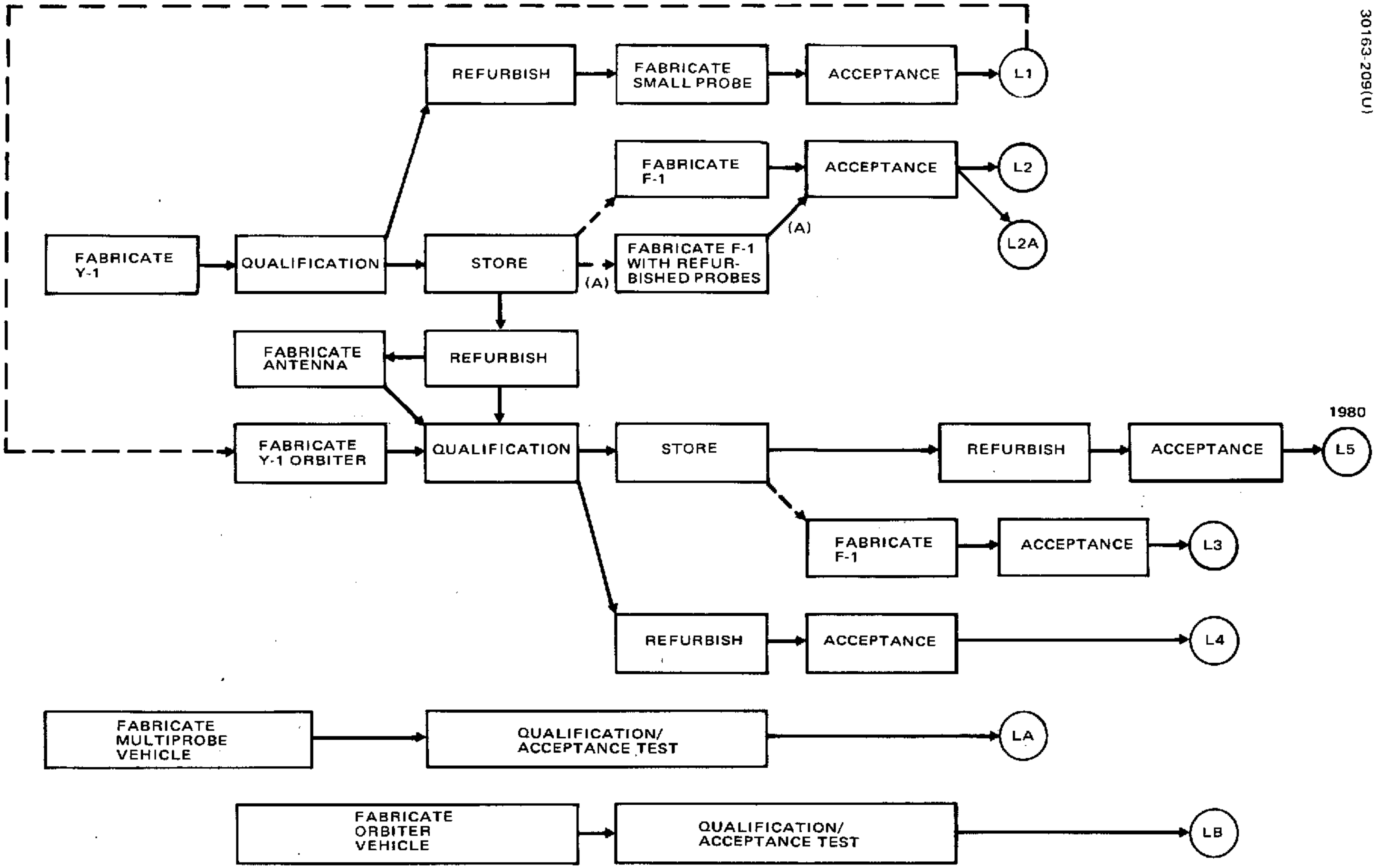
6.2 SPARES HARDWARE

Hughes experience in qualification testing of components designed for space applications indicates that generally little, if anything, is done to the hardware to render it unfit for additional usage. In fact, Hughes experience includes utilization of component qualification hardware in a protoflight spacecraft with no malfunctions traceable to environmental exposures to which the hardware was exposed.

The spares philosophy of option L2-L3 (baseline), to utilize component/subsystem qualification hardware as spares for system test and launch operations, is in keeping with Hughes experience. Since only one of each type of component need be qualified, the spares list would not include all of the hardware required to assemble a spacecraft but rather just one of each type of component regardless of the redundancy carried on the spacecraft.

As spares are expended they may be replaced by either repair and retest of the failed part or new manufacture. Experience on Syncom, Surveyor, Early Bird, ATS, Intelsat II, and TACSAT programs indicates electronic components can be repaired and returned to service far more cost effectively than they can be replaced with new manufacture. After retest, the performance characteristics of the repaired units are indistinguishable from newly manufactured units. Most mechanical parts that fail in environmental test cannot be adequately repaired. Depending on the nature of the failure, mechanical parts for spares replacement will be provided through new manufacture.

The matrix below summarizes the spares philosophy for each of the cost trade options of the previous section. It shows that the low cost protoflight option, LA-LB, would require the manufacture of space units for the orbiter mission.



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LAUNCHES

L1 - MULTIPROBE (PROTOFLIGHT)
L2 - MULTIPROBE (Y-1 AND F-1)
L2A - MULTIPROBE (WITH Y-1 PROBES REFURBISHED)
L3 - ORBITER (F-1)
L4 - ORBITER (PROTOFLIGHT)
L5 - 1980 MISSION
LA - MULTIPROBE } COMBINED QUALIFICATION/
LB - ORBITER } ACCEPTANCE TEST

FIGURE 6-1. FLIGHT AND QUALIFICATION MODEL COST TRADES DIAGRAM

Option	Sparing Philosophy	
	Y-1	F-1
L1	Units	(Units)
L3	(Units)	(Units)
L1	Units	(Units)
L4	(Units)	(Units)
L2 (Baseline)	Kits	(Y-1 Units)
L3	Kits	(Y-1 Units)
L2	Kits	(Y-1 Units)
L4	Units	(Units)
L2A	Kits	(Y-1 Units) and probe kits
L4	Units	(Units)
L2A	Kits	(Y-1 Units) and probe kits
L3	Kits	(Y-1 Units)
LA	N/A	Units
LB	N/A	Units

Parentheses indicate spare units are remainders from previous program phases.

7. SOLAR-THERMAL-VACUUM/THERMAL-VACUUM TRADES

A review of eight commercial, military, and NASA programs, summarized in Table 7-1, showed the average solar-thermal-vacuum activity to be 13 days for spacecraft qualification and 10 days for acceptance. Averages for NASA spacecraft were 16 and 14 days for qualification and acceptance.

It is concluded that the solar-thermal-vacuum durations specified by Ames Research Center (ARC) for Pioneer F/G (Spacecraft Specification PC-210) are appropriate for the Pioneer Venus program. Therefore, the test has been patterned after ARC Spacecraft Specification 210.

On a number of occasions Hughes has substituted thermal-vacuum tests for solar-thermal-vacuum tests and found the results to be quite satisfactory. During this test, incident heat energy is introduced into the spacecraft by means of an IR source or skin heater as opposed to solar intensity lighting. This has a decided cost advantage and is therefore often attractive as a tradeoff (see Table 7-2).

Hughes recommends the following spacecraft thermal-vacuum tests for consideration during the Pioneer Venus test program. All are based upon the same duration and differ only in the recommendations for thermal-vacuum and solar-thermal-vacuum.

<u>Best Technical Approach</u>

Qualification (Y-1) 17 days	Solar-thermal-vacuum
Acceptance (F-1) 14 days	Solar-thermal-vacuum

<u>Technical Option</u>

Qualification (Y-1) 17 days	Solar-thermal-vacuum
Acceptance (F-1) 14 days	Thermal-vacuum

<u>Most Cost Effective</u>

Qualification (Y-1) 17 days	Thermal-vacuum
Acceptance (F-1) 14 days	Thermal-vacuum

TABLE 7-1. PROGRAM COMPARISONS - SOLAR THERMAL VACUUM TESTING

SOLAR THERMAL VACUUM	COMMERCIAL SATELLITES			MILITARY SATELLITES		NASA SPACECRAFT		
	Intelsat IV	Telesat	Western Union	FLTSATCOM	TACSAT	OSO	Pioneer F/G	Mariner Class S/C
Qualification								
Duration	5 days	Part I - 5 days Part II - 5-1/2 days	Same as Telesat	20-1/2 days	10 days	16 days	17 days minimum	Part I - 240 hours Part II - 108 hours
Test Conditions	. Summer soltice - high temp. . Equinox - high temp. . Equinox - low temp. . Equinox - nom. . Winter soltice - low temp.	. Summer soltice - high temp. . Equinox - high temp. . Equinox - low temp. . Winter soltice - low temp. . Part II - add Equinox high and summer soltice high		. Equinox - low temp. . Summer soltice - low temp. . Summer soltice - high temp. . Equinox - high temp. . Equinox - low temp. . Winter soltice - low temp.	. Equinox - low temp. . Equinox - high temp. . Winter soltice - low temp. . Summer soltice - high temp.	Same as acceptance with exposure time increased accordingly.	3 equal segments of test times at solar intensities of 0.9, 5.5 AU and one intermediate value.	Part I - System validation through encounter and playback. Part II - temperature control verification.
Acceptance								
Duration	5 days	5 days	Same as Telesat	11 days	Same as Qual. Test.	13 days	14 days minimum	Part I - 250 hours Part II - 134 hours
Test Conditions	. Summer soltice - high temp. . Equinox - high temp. . Equinox - low temp. . Equinox - nom. . Winter soltice - low temp.	. Summer soltice - high temp . Equinox - high temp. . Equinox - low temp. . Winter soltice - low temp.		. Equinox - high temp. . Equinox - low temp. . Winter soltice - low temp. . Summer soltice - low temp. . Summer soltice - high temp.	Same as Qual. Test.	. 12 hr. hot exposure . 12 hr. cold exposure . 12 hr. hot exposure . 12 hr. cold exposure . 72 hr. hot soak . 72 hr. cold soak . 12 hr. final hot exposure . 12 hr. final cold exposure	3 equal segments of test times at solar intensities of 1.0, 5.5 AU and one intermediate value.	Part I - System validation through encounter and playback. Part II - Temperature control verification.

TABLE 7-2. COST TRADES – SOLAR THERMAL VACUUM/THERMAL VACUUM TESTING

	QUALIFICATION (17 day)						ACCEPTANCE (14 day)					
	Thermal-Vacuum			Solar-Thermal-Vacuum			Thermal-Vacuum			Solar-Thermal-Vacuum		
	Labor	ODC	Total	Labor	ODC	Total	Labor	ODC	Total	Labor	ODC	Total
Test Operations	28 K	-	28 K	28 K	3 K	31 K	23 K	-	23 K	23 K	2 K	25 K
Engineering	10 K	-	10 K	10 K	1 K	11 K	8 K	-	8 K	8 K	1 K	9 K
JPL Chamber	-	-	-	-	289 K	289 K	-	-	-	-	238 K	238 K
Space Simulation Laboratories (HAC)												
• Fixture	71 K	20 K	91 K	71 K	10 K	81 K	-	-	-	-	-	-
• Support	150 K	20 K	170 K	150 K	20 K	170 K	123 K	17 K	140 K	123 K	17 K	140 K
TOTALS	259 K	40 K	299 K	259 K	323 K	582 K	154 K	17 K	171 K	154 K	258 K	412 K

SUMMARY:

	Thermal-Vacuum [1]	Solar-Thermal-Vacuum [2]
Qualification (Y-1)	299 K	582 K
Acceptance (F-1)	171 K	412 K
	470 K	994 K

OPTION:

Qualification (Y-1) (STV)	582 K
Acceptance (F-1) (TV)	171 K [3]
	753 K

[1] Thermal-vacuum tests use IR heat post for heat flux input; however, these tests could be supplemented with solar lighting to a maximum of 1.8 solar constant (SC).

[2] To be done at JPL - Chamber has 2.0 SC capability.

[3] This test would include solar lighting during the thermal-vacuum up to 1.8 SC.

8. CONCLUSIONS

Probe spacecraft development and qualification can be completed in 27 months after go-ahead.

Probe spacecraft acceptance can be accomplished between month 18 and 33.

Orbiter spacecraft development and qualification can be completed between month 13 and 44.

Orbiter spacecraft acceptance can be accomplished between month 45 and month 54.

Unit qualification testing will include vibration, acceleration, spin, magnetic, thermal, and (for the probes) deceleration tests.

Unit acceptance testing will include vibration, magnetic, thermal, and (for the probes) deceleration tests.

Spacecraft qualification testing will include vibration, spin, EMC, and solar-thermal-vacuum tests.

Spacecraft acceptance testing will include vibration, solar-thermal-vacuum, pressure-temperature, and deceleration tests.

Probe acceptance testing will include vibration magnetic, solar-thermal-vacuum, pressure-temperature, and deceleration tests.

Parachute subsystem development testing shall include a minimum of four drop tests of the complete subsystem. These will be from a subsonic aircraft. Boilerplate probe simulators will be used. The complete parachute subsystem sequence will be exercised.

The most cost effective test approach to the two-mission program, from the standpoint of hardware costs, is a protoflight, or combination qualification/acceptance test.

The recommended best technical approach to thermal testing of both spacecraft is a 17 day solar-thermal-vacuum qualification test and a 14 day solar-thermal-vacuum acceptance test. An acceptable technical option is a 17 day solar-thermal-vacuum qualification test and a 14 day thermal-vacuum acceptance test.

REFERENCES

1. "Methods and Sequence of Test and Launch Operations," Hughes Aircraft Company, HS-507-0022-86, Task No. TP-1, dated 5 February 1973.

2. "Environmental Levels Test Requirements," Hughes Aircraft Company, HS-507-0022-74, Task No. TP-2, dated 29 January 1973.

3. "Cost Analysis Test Hardware," Hughes Aircraft Company, HS-507-0022-94, Task No. TP-3, dated 26 March 1973.

4. "Solar Thermal Vacuum Testing Philosophy," Hughes Aircraft Company, HS-507-0022-64, Task No. TP-4, dated 30 January 1973.

5. "Parachute Subsystem Tests," Hughes Aircraft Company, HS-507-0022-64, Task No. TP-5, dated 16 March 1973.